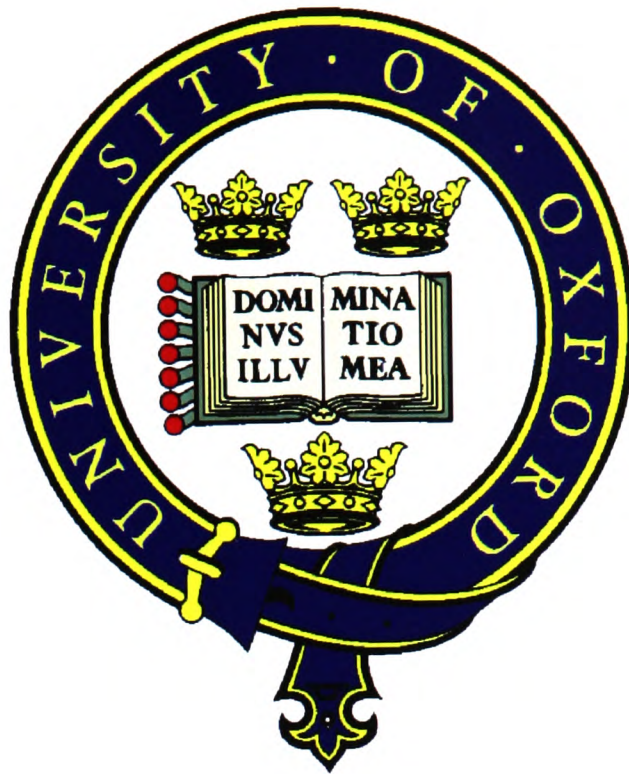
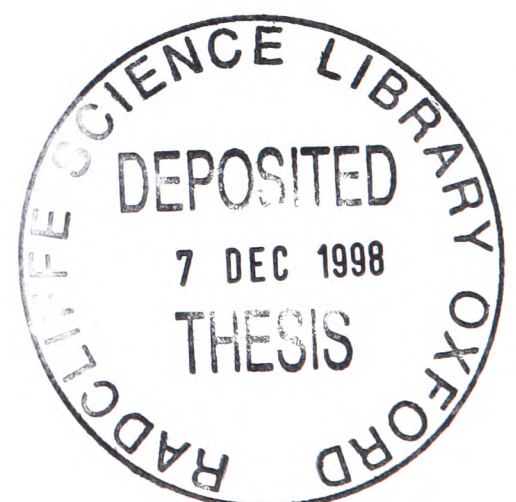




Impact Damage Tolerance of Carbon Fibre Reinforced Plastics

A thesis submitted to the Department of
Engineering Science of the University of Oxford
in partial fulfilment of the requirements
for the degree of Doctor of Philosophy

Digby Duncan Symons
Magdalen College
University of Oxford
Trinity Term, 1998



Abstract

Digby Symons
Magdalen College

D.Phil. Thesis
Trinity Term 1998

Impact Damage Tolerance of Carbon Fibre Reinforced Plastics

Carbon fibre reinforced plastic (CFRP) is a composite composed of very high strength and stiffness carbon fibres within a polymer matrix. Low mass and cost make CFRP attractive for aerospace applications. At present application of CFRP is limited by the poor residual strength and fatigue properties of the material following impact damage, and because of the limited understanding of the problem.

Damage to CFRP results in a combination of fibre fracture and matrix cracking. The strength of CFRP perpendicular to the fibre direction is relatively poor and so the material is generally used as a laminate of unidirectional plies of different fibre orientations. Laminates are particularly prone to delamination, this is matrix cracking which separates plies.

In this thesis techniques for measurement of damage type and severity in CFRP are evaluated. These are used to quantify the distribution of damage resulting from hard body impacts at high, low and quasi-static incident velocities. A composite material damage model in a dynamic finite element analysis computer program (DYNA 3D) was used to predict the experimentally measured impact damage. However the prediction was found to be insufficiently accurate for reliable use by designers. Recommendations are made for improvements in the numerical modelling of impact damage in CFRP.

Mechanical tests that provide data on the effect of impact damage on the residual strength of CFRP specimens are reported. These show that providing impact damage is limited to delamination, with little fibre fracture occurring, the compressive residual strength will be more severely reduced than the tensile property. Strain gauge measurements show that this effect is caused by local compressive buckling of the delaminated plies in the impact damaged area. Equivalent hole sizes, causing the same residual strengths as in the impact damaged specimens, are given.

Cyclic load tests were conducted providing stress-life data for impact damaged specimens of CFRP. The data fit a stress-life diagram divided into three scatter band regions: static fracture; cycle dependent damage growth; and infinite life. Qualitative and quantitative observations are given of the progression of damage in a fatigue after impact damage test. Strain gauge and force/extension measurements show the presence of creep and energy absorption in undamaged and impact damaged CFRP. Much of the observed behaviour may be explained by a viscoelastic model. Because of the presence of viscoelastic creep it is suggested that the fatigue mechanism must depend on interacting cycle dependent and time dependent mechanisms.

Preface

This dissertation is an account of the research carried out by the author in the Department of Engineering Science at the University of Oxford. The work was funded by the Engineering and Physical Sciences Research Council and Rolls Royce Plc. and supervised by Professor Carlos Ruiz. No part of the dissertation has been submitted for a degree at any other university. The research described herein is original, although the work of others has been drawn upon freely, with due acknowledgement in the text.

Acknowledgements

I would like to thank Professor Carlos Ruiz for his help, advice and guidance in completing this project. I also extend my gratitude to Stephen Hallett, Pete Hardy and Richard Duffin for their help with the computational and experimental work and to my parents for their careful proofreading. Support and contributions from Paul Blomerus, David Nowell, David Hills and all of the Solid Mechanics Group are gratefully acknowledged.

Table of Contents

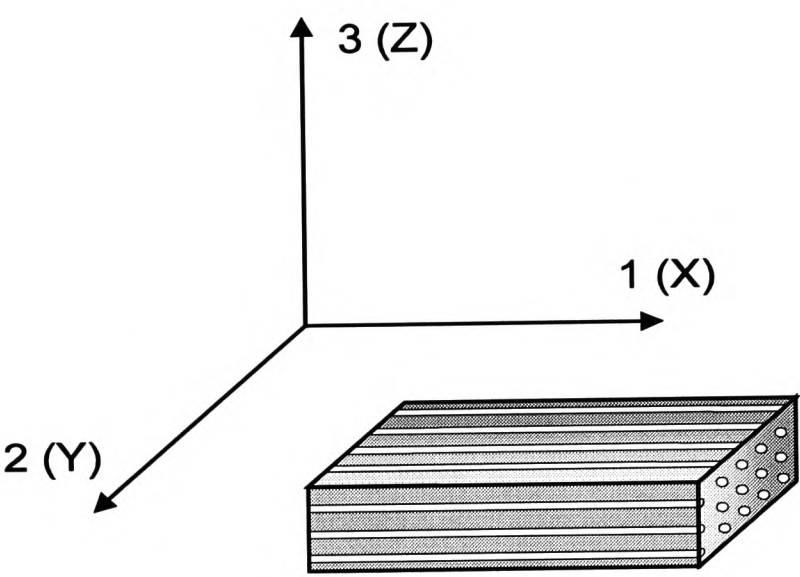
	Nomenclature	7
1	Introduction	9
1.1	Properties of Carbon Fibre Reinforced Plastics	9
1.1.1	Material Comparisons	9
1.1.2	Carbon Fibre	9
1.1.3	Epoxy	10
1.1.4	Laminated Manufacture	10
1.1.5	Failure Mechanisms	11
1.2	Aero-engine Applications of CFRP	11
1.3	Limitations	12
1.3.1	Certification	13
1.3.2	Temperature	13
1.3.3	Reliability	14
1.3.4	Defects in CFRP	14
1.4	Impact Damage	15
1.5	Design Practice	15
1.5.1	Damage Limitation	15
1.5.2	Certification	17
1.5.3	No-Growth Concept	17
1.6	Aims of the Thesis	18
2	Measurement and Characterisation of Impact Damage	19
2.1	Coupons	19
2.1.1	Material Type	19
2.1.2	Manufacture	19
2.1.3	Coupon Design	21
2.2	Structural Support	23
2.3	Low Speed Impact Tests	23
2.3.1	Type of Impact	23
2.3.2	Industry Standard Impact Tests	24
2.3.3	Impactor Design	25
2.3.4	Gas Gun	26
2.3.5	Measurement of Velocity	26
2.3.6	Measurement of Impact Force	28
2.3.7	Number of Tests	28
2.4	High Speed Impact Tests	28
2.4.1	Projectile	28
2.4.2	Gas gun	28
2.4.3	Measurement of Velocity	29
2.4.4	Coupon Support	29
2.4.5	Number of Tests	29
2.5	Quasi-static Indentation Tests	29
2.5.1	Indenter	29
2.5.2	Displacement	29

		4
	2.5.3 Testing Machine	30
	2.5.4 Loading/Unloading Sequences	30
2.6	Evaluation of Damage Measurement techniques	30
	2.6.1 Visual Inspection	30
	2.6.2 Indentation	31
	2.6.3 Ultrasonic C-scan	32
	2.6.4 Ultrasonic B-scan	34
	2.6.5 X-Radiography	35
	2.6.6 X-ray Computer Microtomography	38
	2.6.7 Optical Microscopy	41
	2.6.8 Dye Penetrant Microscopy	44
	2.6.9 Thermography	44
	2.6.10 Acoustic Emission	45
	2.6.11 Eddy Currents	45
	2.6.12 Fibre Optics	45
	2.6.13 Moiré Interferometry	45
	2.6.14 Thermal De-plying	46
	2.6.15 Scanning Electron Microscopy	46
	2.6.16 Conclusions	46
2.7	Low Speed Impact Damage	48
	2.7.1 Description of Damage	48
	2.7.2 Indentation	48
	2.7.3 Delamination Area Measured by C-scan	49
2.8	High Speed Impact Damage	53
	2.8.1 21 ply Coupons	53
	2.8.2 29 ply Coupon	55
2.9	Quasi-static Indentation Damage	57
	2.9.1 Continuous Indentation	57
	2.9.2 Repeated Indentation, 29 ply Coupon	58
	2.9.3 Repeated Indentation, 21 ply Coupon	60
2.10	Conclusions	64
3	Review of Composite Modelling Techniques	65
3.1	Approximate Methods	66
	3.1.1 Hertzian Contact Stiffness	66
	3.1.2 Plate Bending (Isotropic Assumption)	67
	3.1.3 Delamination Prediction	68
3.2	Failure Criteria	69
	3.2.1 Single Ply Failure Criteria	70
	3.2.2 Multiply Laminate Failure Criteria	71
3.3	Impact Energy	73
3.4	Prediction of Delamination by Fracture Mechanics	75
3.5	Numerical Analysis Incorporating Coupon Tests	78
3.6	Numerical Analysis with Stress Based Failure Criteria	79
3.7	Conclusions	81
4	Numerical Analysis of Impact	83
	4.1 DYNA 3D	83

4.1.1	Finite Element Method	83
4.1.2	Explicit Time Integration	84
4.1.3	Element Formulation	86
4.1.4	Hourglassing	86
4.1.5	Contact Surfaces	87
4.1.6	Composite Failure Criteria and Material Type	87
4.1.7	Description of Model	90
4.1.8	Material Data	91
4.1.9	Strain Rate Dependency	93
4.2	Low Speed Impact	93
4.2.1	Impactor	93
4.2.2	Indentation Stiffness	94
4.2.3	Force Time History	95
4.2.4	Damage	98
4.3	High Speed Impact	103
4.3.1	Projectile	103
4.3.2	Force/Displacement	103
4.3.3	Damage	105
4.4	Conclusions	105
5	Effect of Impact Damage on Residual Strength	111
5.1	Design of Test Rigs	111
5.1.1	Industry Standard Tests	111
5.1.2	Undamaged Compression Strength Test	113
5.1.3	Pin Ended Compression Test Rig	113
5.1.4	Tension/Compression/Fatigue Jaws	115
5.2	Testing Machines	116
5.2.1	Denison	116
5.2.2	Dartec	116
5.3	Data Acquisition	117
5.3.1	Strain Gauges	117
5.3.2	Displacement Transducer	117
5.3.3	Data Logging	117
5.4	Undamaged Tension	118
5.5	Tension After Impact	119
5.5.1	Experimental Results	119
5.5.2	Predictive Models for Residual Tensile Strength	120
5.5.3	Equivalent Hole	122
5.6	Undamaged Compression	124
5.6.1	Material Compression Strength Coupon	124
5.6.2	Models for Compression Strength	125
5.6.3	50 x 50 mm, 29 ply Coupon	128
5.6.4	140 x 85 mm, 21 ply Coupon	128
5.6.5	Structural Buckling	128
5.7	Compression After Impact	130
5.7.1	Literature Results	130
5.7.2	Experimental Results	132
5.7.3	Delamination Buckling	136
5.7.4	Damage Zone Model	138

5.7.5	Equivalent Hole	138
5.8	Strain Distribution at Impact Damage Zone	140
5.9	Conclusions	142
6	Fatigue Behaviour	144
6.1	Introduction to Fatigue in Composites	144
6.2	Effect of Damage on Fatigue Life	145
6.2.1	Literature Results	146
6.2.2	Damage Level	148
6.2.3	Load Ratio	148
6.2.4	Frequency	148
6.2.5	Testing Machine	149
6.2.6	Failure Mode	149
6.2.7	Fatigue Test Results	151
6.2.8	Fatigue Life Models	154
6.3	Growth of Damage During Fatigue Lifetime	159
6.3.1	Literature Results	159
6.3.2	External Damage Development	160
6.3.3	Progressive C-Scans	161
6.3.4	Change in Coupon Modulus	161
6.3.5	Interpretation of Results	164
6.4	Hysteresis in Impact Damaged Coupons	165
6.4.1	Literature Results	165
6.4.2	Creep Under Constant Compressive Load	166
6.4.3	Hysteresis in Quasi-static Cyclic Loading	169
6.4.4	Hysteresis Under Dynamic Loading	171
6.4.5	Hysteresis in Undamaged CFRP	174
6.4.6	Viscoelastic Model	177
6.4.7	Discussion	181
6.5	Conclusions	182
7	Conclusions and Recommendations for Further Work	184
7.1	Measurement and Characterisation of Impact Damage	184
7.2	Numerical Modelling of Impact	185
7.3	Effect of Impact Damage on Residual Strength	185
7.4	Effect of Impact Damage on Fatigue Life	186
	References	188

Nomenclature



Local (Global) Coordinate System

Coordinate system

The coordinate axes 123 refer to a unidirectional ply within a laminate. 3 is normal to the laminate, 1 is the fibre longitudinal direction, 2 is transverse to the fibres, within the plane of the laminate ply.

The axes XYZ are global for a multi-ply laminate. Z is normal to the laminate, X and Y are in the laminate plane.

Symbols

ρ	Density
E	Young's Modulus
σ_y	Yield stress (metals)
ϵ_f	Elongation to failure
ϵ_{ft}	Tensile failure strain
ϵ_{fc}	Compressive failure strain
ϵ_{mt}	Matrix tensile failure strain
ϵ_{mc}	Matrix compressive failure strain
σ_f	Failure stress
V_f	Fibre volume fraction
E_1	Elastic modulus longitudinal direction
E_2	Elastic modulus transverse direction
E_3	Elastic modulus normal direction
E_m	Matrix elastic modulus
G_{12}	Shear modulus (in plane)
G_{23}	Shear modulus (transverse/normal)
G_{31}	Shear modulus (normal/longitudinal)
G_m	Matrix shear modulus
ν_{12}	Poisson's ratio in-plane
ν_{23}	Poisson's ratio transverse/normal

v_{31}	Poisson's ratio normal/longitudinal
σ_1	Longitudinal Stress
σ_2	Transverse Stress
σ_3	Normal Stress
τ_{12}	Shear Stress (in plane)
τ_{23}	Shear Stress (transverse/normal)
τ_{31}	Shear Stress (normal/longitudinal)
ϵ_2	Transverse strain
γ_{12}	In plane shear strain
γ_{23}	Transverse/normal shear strain
T_1	Longitudinal tensile strength (axial fibre failure)
T_2	Transverse tensile strength (matrix failure)
T_3	Normal tensile strength (matrix failure)
S_{12}	In plane (longitudinal/transverse) shear strength
S_{23}	Transverse/normal shear strength
S_{31}	Normal/longitudinal shear strength
C_1	Longitudinal compressive strength
C_2	Transverse compressive strength (matrix crushing)
K_f	Bulk modulus of failed material
v_0	Incident impact velocity
v_1	Rebound velocity
c	Speed of sound
m	Mass of impactor
U_0	Incident impact energy
U_1	Rebound impactor kinetic energy
U_d	Absorbed energy
U_{bvid}	Incident impact energy to cause barely visible impact damage
U_{sf}	Static threshold energy for permanent damage
U_{df}	Dynamic threshold energy
k_{indent}	Indentation stiffness
P	Contact force
P_{thres}	Threshold force for permanent damage
P_{max}	Maximum force
R	Tip radius
a	Radius of simple support
h	Plate thickness
δ	Displacement
G	Energy release rate
K_{IC}	Mode I fracture toughness
U_t	Strain energy density
t	Time
l	Length
R	Load ratio (minimum load / maximum load)
N	Number of cycles

Chapter 1

Introduction

1.1 Properties of Carbon Fibre Reinforced Plastics

1.1.1 Material Comparisons

Carbon fibre reinforced plastic (CFRP) is a composite of carbon fibres within a polymer matrix. The fibres provide high strength and stiffness while the matrix gives the composite high toughness. Both matrix and fibres have relatively low densities. The result is a material which is strong, stiff and very light. This makes CFRP a preferred material in structural applications where weight is critical.

Polymer matrix composites (PMCs) have been in limited use in aerospace applications since the 1940s. However, it was not until the early 1970s that large-scale carbon fibre production made these composites serious alternatives to lightweight metal alloys for complex lightweight structures.

In Table 1 the material properties of a CFRP laminate are compared with three alternative structural metals [1]. It can be seen that the specific mechanical properties of CFRP are very much superior to those of the metals. CFRP is an expensive material but, when complete manufacturing costs are considered, CFRP aerospace components often turn out cheaper than their titanium alternatives.

1.1.2 Carbon Fibre

Carbon is a material with a high density of covalent bonds which gives it a very high stiffness. However the fracture toughness is low and in its bulk form the brittle material has a low tensile strength. In fibrous form the defects which initiate fracture are largely eliminated and therefore a high tensile strength is achieved. Fracture toughness of the composite as a whole is further improved by the combined action of the matrix binder material and the fibres. A crack from the fracture of a single fibre is blunted by the surrounding tougher matrix and does not propagate as easily. Cracks in the matrix are restrained by the bridging of fibres which must be pulled out in order for the crack to propagate.

Carbon fibres have high strength and high modulus compared with other fibres used in composites. They have good fibre to matrix bonding characteristics and give good resistance to flame spread. However carbon fibre composites have low abrasion resistance, low breaking strain, and are costly.

1.1.3 Epoxy

The most common matrix material used in carbon fibre composites is thermosetting epoxy. Epoxy generally has average properties when compared to other resin systems used in composites. It has average tensile strength, temperature resistance, resistance to flame spread and chemical resistance. Epoxy is a medium cost resin system. Epoxy does however have low mould shrinkage which is a significant advantage in the manufacture of precision components.

Material Data Comparison

Material			Aluminium 2024	Steel	Titanium- 6A1-4V	CFRP 0°/90° lay-up
Density	ρ	kg/m ³	2800	7900	4420	1590
Young's Modulus	E	GPa	72	210	115	74
Tensile Strength	σ_y	MPa	395	1600	900	850
Strain to Failure	ϵ_f	%	10	10	10	1.1
Specific Stiffness	E / ρ	MNm/kg	25.7	26.6	26.0	46.5
Specific Strength	σ_y / ρ	MNm/kg	0.141	0.203	0.203	0.535

Table 1.1

1.1.4 Laminated Manufacture

Along the direction of the fibres the mechanical properties of a composite are best. The transverse stiffness and strength are given by the matrix, however, and so are much lower. Consequently a practical composite component must have its fibre directions distributed in proportion to the anticipated applied loads. In the most common applications this is done by laying-up a laminated sheet of several plies of unidirectional fibres in different directions. A

quasi-isotropic lay-up will give approximately uniform mechanical properties in all in-plane (two-dimensional) directions.

The transverse properties of a laminate will still be poor, being only that of the matrix, and are typically only 10% of the in-plane strength and stiffness. This transverse weakness makes laminates prone to delamination, resulting from in-plane cracking between plies.

The number of plies, and the number of interfaces prone to delamination, may be reduced by the use of woven fibre plies.

1.1.5 Failure Mechanisms

Fibre-reinforced composites can be considered to be made up of three components: the fibres, the matrix and the interphase region or interface. The differing mechanical properties of these components and the complex geometry of a laminate mean that a composite exhibits a number of interacting failure mechanisms.

Cantwell and Morton [2] reviewed the following mechanisms in detail. They are listed in order of increasing complexity. Fibre fracture results from tensile failure of the fibre. Fibre microbuckling is a compressive failure of the fibres resulting from shear deformation of the surrounding matrix; this is normally followed immediately by fracture at the points of maximum bending. Fibre-matrix debonding is a failure of the interphase region and will allow fibre pull-out with energy being dissipated in friction between the fibre and matrix. Intralaminar matrix cracking may be localised, such as shear cracks through transverse fibre plies, or extensive, in the form of splitting resulting from long cracks parallel to the fibres. Plastic deformation of the matrix can also occur, particularly in ductile matrices such as PEEK (polyetheretherketone). Delamination is a particular form of matrix cracking which often runs through the interphase region as well. A delamination is an in-plane crack between plies of different fibre orientations.

1.2 Aero-engine Applications of CFRP

The aero gas turbine was developed in the 1940s for military applications. Civil applications followed in the 1950s. Cold war military requirements drove the development of engine

technology with civil engines benefiting from the military development. At the present time, however, the economics are very different. The military market is stabilising and the civil engine market is fiercely competitive. Three main engine builders (General Electric, Pratt and Whitney, and Rolls-Royce) compete for market share.

The early incentives for the use of PMCs (polymer matrix composites) in aero-engines were weight reduction and performance improvement. PMC components typically provide a 20-30% weight saving [3] over their metallic equivalents. Engine weight is important; it has an impact on fuel consumption and, particularly in military applications, aircraft agility.

Engine choice (product discrimination) is now much more influenced by economics. Factors include initial procurement cost and cost of ownership. New technology is required to meet cost targets as well as performance targets.

Temperature requirements limit the use of PMCs in aero-engines. However, most of the nacelle (outer housing) of a modern aero gas turbine is PMC, a component which accounts for 25% of the weight and 20% of the cost of the powerplant.

Figure 2 shows a modern turbofan aero-engine. There are several components of the engine in which CFRP could be used. These include non structural components such as nose cone spinners, cowlings, bypass ducts and core engine fairings. There is also potential for application of the material to structural and aerodynamically loaded components such as fan blades, outlet guide vanes, annulus fillers and variable guide vane rings as well as core engine load paths. The use of CFRP for the large static structure immediately behind the fan on a civil engine could potentially yield the largest benefits in terms of cost and weight reduction.

1.3 Limitations

Rolls-Royce made their first major use of composites in the application of CFRP to the Hyfil fan blade for the RB211 engine (used on the Boeing 747). However problems with manufacturing repeatability and bird impact led to a switch to titanium. The wasted development time was very costly to the company.

It is therefore vital that any limitations of a “new” material, such as CFRP are carefully assessed in advance.

Cutaway of Turbofan Aero-engine

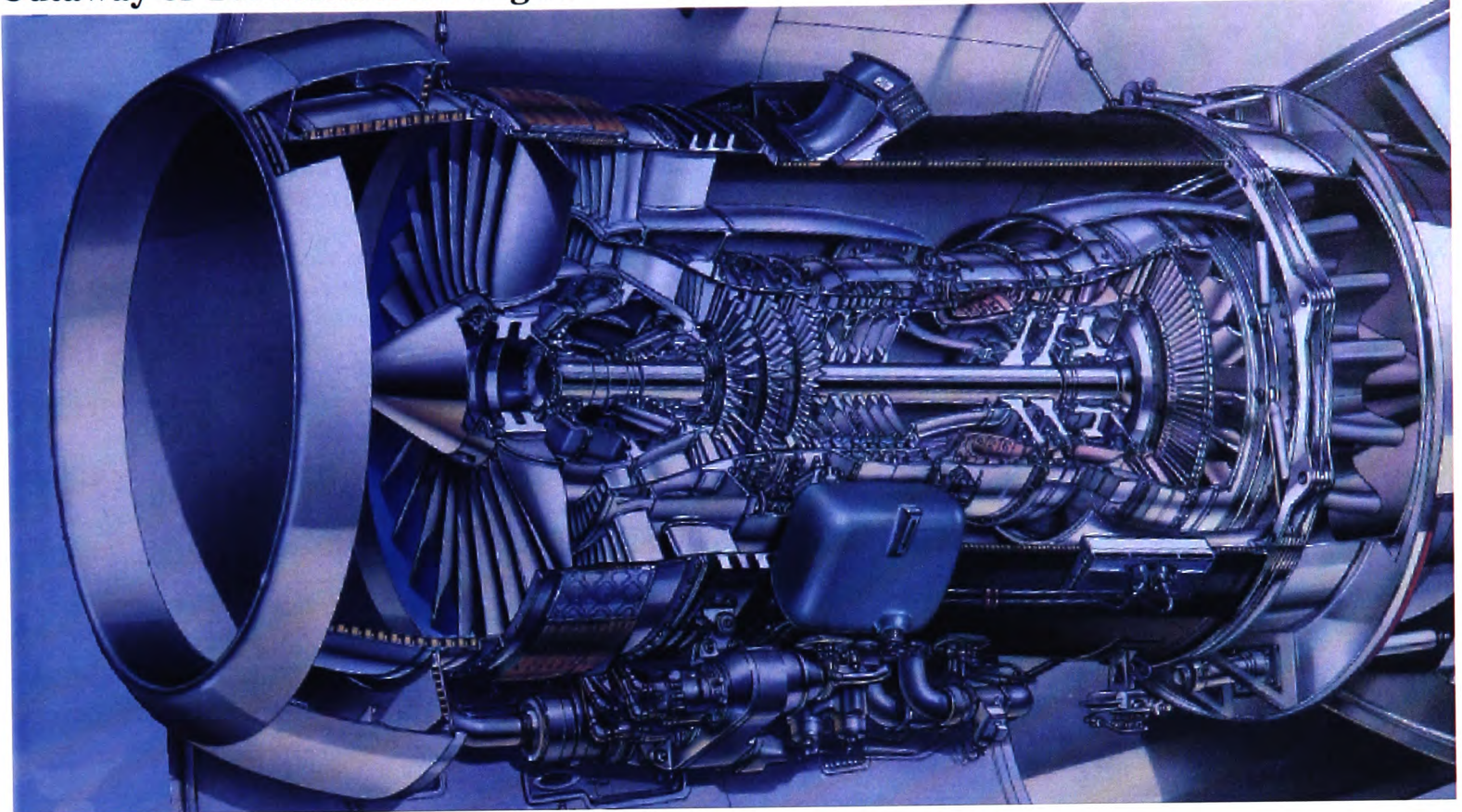


Figure 1.1

1.3.1 Certification

The introduction of new material components often requires the development of an entirely new engine because the certification costs are too high for a single component. For example, certification of a CFRP outlet guide vane requires a 150 hour ‘modification approval’ engine test which costs more than £1M.

Since entirely new engines are infrequently developed, the introduction of advanced materials into aero-engines has been slower than originally anticipated. In fact advanced composite materials have been most rapidly adopted in the world of sports and athletics where certification and testing costs are much lower.

1.3.2 Temperature

Temperature requirements limit the application of CFRP in aero-engines. Carbon fibres oxidize in air at 300 to 400°C. Some matrix types can be used up to a maximum of 300°C.

Performance of composites below these temperatures is a function of the matrix. The maximum continuous service temperature for epoxy polymers is usually between 80 and 180°C.

The air temperature at the high pressure exit of an aero-engine compressor is about 400°C. This is then followed by a flame temperature of 2000°C in the combustion chamber. Applications of CFRP are therefore limited to cooler sections of the engine, such as bypass areas in a turbofan or the first, low pressure, stages of the compressor.

1.3.3 Reliability

Metals have been used since the invention of the aero-engine and therefore the lifeing methodologies are now well established. This is not the case for composites whose material behaviour is significantly different to metals. Table 1.2 [4] summarises these differences.

Although advanced composite materials have been in existence for some time they have not yet proved themselves suitable for many applications where reliability is paramount.

Summary of Composite Material Behaviour

Condition		Composite Behaviour Relative to Metals
Load-Strain Relationship		More linear strain to failure
Notch Sensitivity	Static	Greater sensitivity
	Fatigue	Less sensitivity
Transverse Properties		Weaker
Mechanical Properties Variability		Higher
Sensitivity to Aircraft Hygrothermal Environment		Greater
Damage Growth Mechanism		In-plane delamination instead of through-thickness cracks

Table 1.2

1.3.4 Defects in CFRP

The strength of a laminate can be reduced by the presence of defects. Defects may arise during manufacture or in service.

Defects that may occur during manufacture include: resin rich areas; fibre rich areas; delamination; ply drop-off; and ply kinking. All these defects may be avoided by strict quality control measures at all stages of manufacture. In service defects may occur as the result of impact events.

1.4 Impact Damage

In the aerospace industry impact causes may be divided into “accidental handling damage” and “foreign object damage”. Accidental handling damage will occur during manufacture or maintenance and can include dropped tools and vehicle collisions. Such impacts are generally low velocity events. Foreign object damage is a result of impacts at aircraft take-off or flying speeds and can include runway debris, hailstones, bird strikes and, in the case of military aircraft, ballistics.

Impacts will cause damage to the laminate involving a distribution of fibre fracture and matrix cracking. This may leave a permanent indentation or, if the impact energy is high enough, the projectile will perforate the laminate. Although perforation and, to a lesser extent, indentation are visible to visual inspection any impact event may also cause extensive internal delamination. This damage is not externally visible and consequently may be safety critical.

Figure 1.2 shows an experimental CFRP outlet guide vane with “soft body” impact damage. This test represents the type of damage resulting from a bird strike.

1.5 Design Practice

The occurrence of impact damage in service must be accepted by aerospace designers. Whether the damage results from foreign objects or accidental handling, the design must be damage tolerant.

1.5.1 Damage Limitation

Various techniques have been considered for improving the damage tolerance of composites. These normally involve reducing the severity of damage resulting from a particular impact.

One approach is to improve the fracture toughness of the composite. This has generally meant improving the toughness of the matrix and the quality of the matrix/fibre bond. Various tougher matrix materials have been developed and some success has been demonstrated using thermoplastic polymers (such as PEEK), rather than thermoset polymers (such as epoxy). Some manufacturers have also experimented with particular additives to improve toughness. Cantwell and Morton [5] have conducted a comprehensive review of the impact resistance of composite materials.

CFRP Outlet Guide Vane with Impact Damage

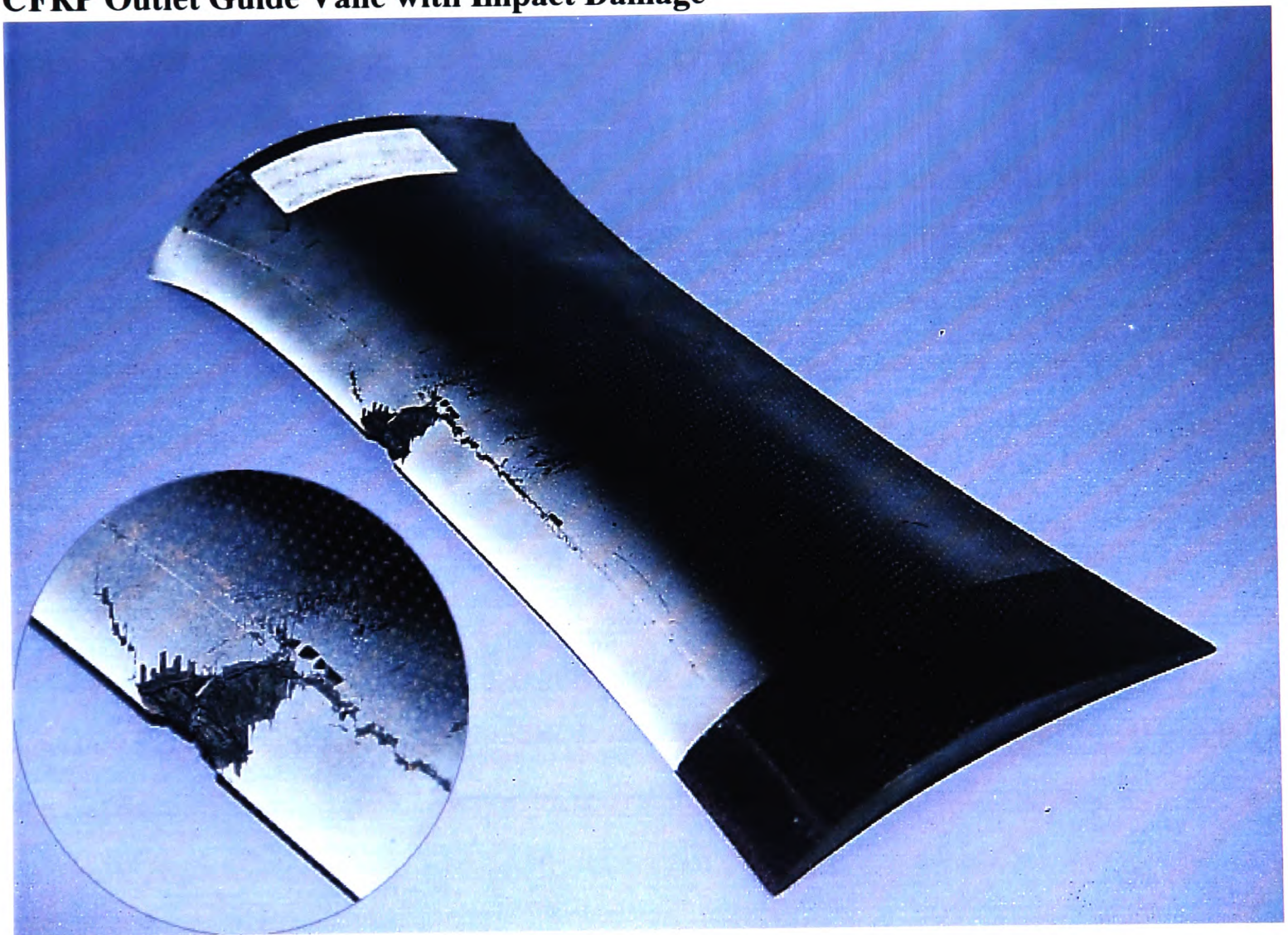


Figure 1.2

Because delamination cracking has been shown to be one of the dominant damage modes resulting from impact, much research attention has focused on through thickness reinforcement. The fibres in laminated composites lie normally only within the plane of one ply, whether the ply is unidirectional or woven. Using fibres in the through thickness direction, by stitching after lay-up, has been shown to reduce delamination.

Increasing the flexibility of a composite structure is another way of reducing the level of damage. Increased flexibility will mean that the structure can absorb more strain energy before failure and therefore will withstand higher impact energies before damage occurs. Flexibility can be increased in a number of ways. Using $\pm 45^\circ$ plies on the outside of the laminate can reduce the flexural (bending) stiffness. Substituting glass or kevlar reinforcement for some of the carbon fibres can also improve the flexibility.

Since PMCs are essentially brittle in nature it is necessary to add metallic protection if impact energy is to be absorbed by plastic deformation. An example of such protection is the use of a titanium leading edge on CFRP fan blades to improve bird impact resistance.

All of the methods described above require an element of compromise. One problem is that a technique used to reduce the magnitude of damage (e.g. the size of delaminations) resulting from an impact may also reduce the residual properties. It may therefore be counter productive. Such a technique would not in fact have improved the damage tolerance. Other methods, while effective in improving damage tolerance, may increase the weight or cost of the component so that it is no longer competitive.

1.5.2 Certification

Civil aviation regulations require that any safety critical aerospace component must undergo a rigorous certification procedure before it enters service. For newly introduced composite components it is necessary to demonstrate that the structure is damage tolerant.

Because it is impractical to conduct a very large number of full scale tests on large structures like an aircraft wing, coupon (small specimen) tests are used to demonstrate the suitability of the material for the application. Such coupon tests include residual strength and fatigue life tests on coupons subjected to impact damage.

1.5.3 The No-Growth Concept

The sensitivity of composite laminates to impact damage, and the possibility that this damage may grow under cyclic loading, has led to the development of the “no-growth” concept.

Because there is a limit to the level of damage visible to external inspection by the naked eye during maintenance, the designer of a component must assume that the maximum level of undetectable damage is present. This level of damage is termed “barely visible impact damage” or BVID. Certification of a component requires that there must be no growth of the damage under the cyclic loads the component will experience between inspection intervals.

A typical design rule applying the no-growth concept is that for components subject to handling damage, the design strain should be limited to 0.4%. This is approximately one third of the material failure strain and is a very conservative design rule.

1.6 Aims of the Thesis

This research aims to develop understanding of the impact damage tolerance of CFRP.

The work aims to contribute to further cost and weight savings by extending the utilisation of CFRP to primary structural components in Rolls Royce aero-engines. At present composites are limited to secondary structural components such as the nose-spinner and the honeycomb sandwich panel cowlings. However, there is potential for application of CFRP in the first stage fan-blades, outlet guide vanes and various engine structural components such as thrust bearing housings and vane control rings.

By experimental investigations, using a typical aerospace CFRP material, and by reference to the literature, the work has the following particular aims:

- i) Determine the most appropriate techniques for the measurement and characterisation of impact damage.
- ii) Identify the type, and quantify the extent, of damage resulting from impact.
- iii) Investigate the suitability of a dynamic finite element analysis package, DYNA 3D, for predicting impact damage.
- iv) Quantify, and identify the mechanism, of the reduction in residual strength after impact damage.
- v) Quantify the effect of impact damage on fatigue lifetime.
- vi) Investigate the mechanism of fatigue degradation after impact damage.

Chapter 2

Measurement and Characterisation of Impact Damage

Impact damage in CFRP results in a complex distribution of fibre and matrix cracks. In this chapter techniques for characterising and measuring the severity of such damage are investigated. Appropriate techniques are applied to experimental impact tests.

2.1 Coupons

2.1.1 Material Type

The material selected for the test coupons was the T300/914 system: Torayca 300B high tensile carbon fibres in a Fibredux 914 epoxy resin matrix.

Carbon fibres are usually produced by subjecting organic precursor fibres such as polyacrylonitrile (PAN) or rayon to a sequence of heat treatments, so that the precursor is converted to carbon by pyrolysis. Torayca T300B high strength carbon fibres have the following mechanical properties (manufacturer’s data) [6]:

Young’s modulus	$E = 235 \text{ GPa}$
Tensile failure stress	$\sigma_f = 3.53 \text{ GPa}$
Tensile failure strain	$\epsilon_f = 1.5 \%$

Fibredux 914 is a modified epoxy resin that is pre-impregnated with unidirectional fibres of carbon, glass or aramid, producing “pre-pregs” (pre-impregnated plies) which are then cured to form a composite component. The material is manufactured by Hexcel Composites (formerly Ciba-Geigy).

T300/914 has a recommended operational temperature range of -60°C to +180°C. The fibre volume fraction $V_f = 60\%$. Typical design data for the material are presented in Table 2.1. The values are for unidirectionally reinforced T300/914 CFRP.

2.1.2 Manufacture

Coupons were manufactured by the author from unidirectional pre-preg T300/914 tape. Pre-preg tape consists of a thin layer of unidirectional carbon fibres pre-impregnated with uncured

epoxy. Tape is stored in a freezer before manufacture to ensure its shelf life. The lay-up used for the coupon laminates was alternating 0°/90°. Autoclave curing under combined temperature and pressure was used to thermoset the epoxy resin. During curing it was necessary to use bleed cloth to remove excess resin so that the correct fibre volume fraction was achieved.

Curing was conducted under a combined top pressure of 100 psi (6.5 bar) gauge and bottom vacuum of 0.15 inches Hg absolute pressure. This gave a total pressure of approximately 8 bar.

For Fibredux 914 (MSRR9411) lay-ups with a thickness of less than 10mm the recommended autoclave cure cycle requires 1 hour at 175°C with a gradual increase to and decrease from this hold temperature. The actual temperature cycle used is shown in Figure 2.1. The initial step brings the lay-up to just above ambient temp. This is required since the pre-preg is stored in a freezer before manufacture. The main cure cycle then starts with a ramp up to 175°C. The ramp rate is slowed before reaching the maximum temperature so there is no overshoot. There is then a hold at 175°C for 1 hour before a smooth drop back to ambient temperature.

After autoclave curing the component is removed from the autoclave and from its mould. Resin bleed cloth is also removed at this stage. The final manufacturing stage is the post cure. This is necessary to relax residual stresses. The post cure used for specimen preparation was four hours at 120°C at atmospheric pressure.

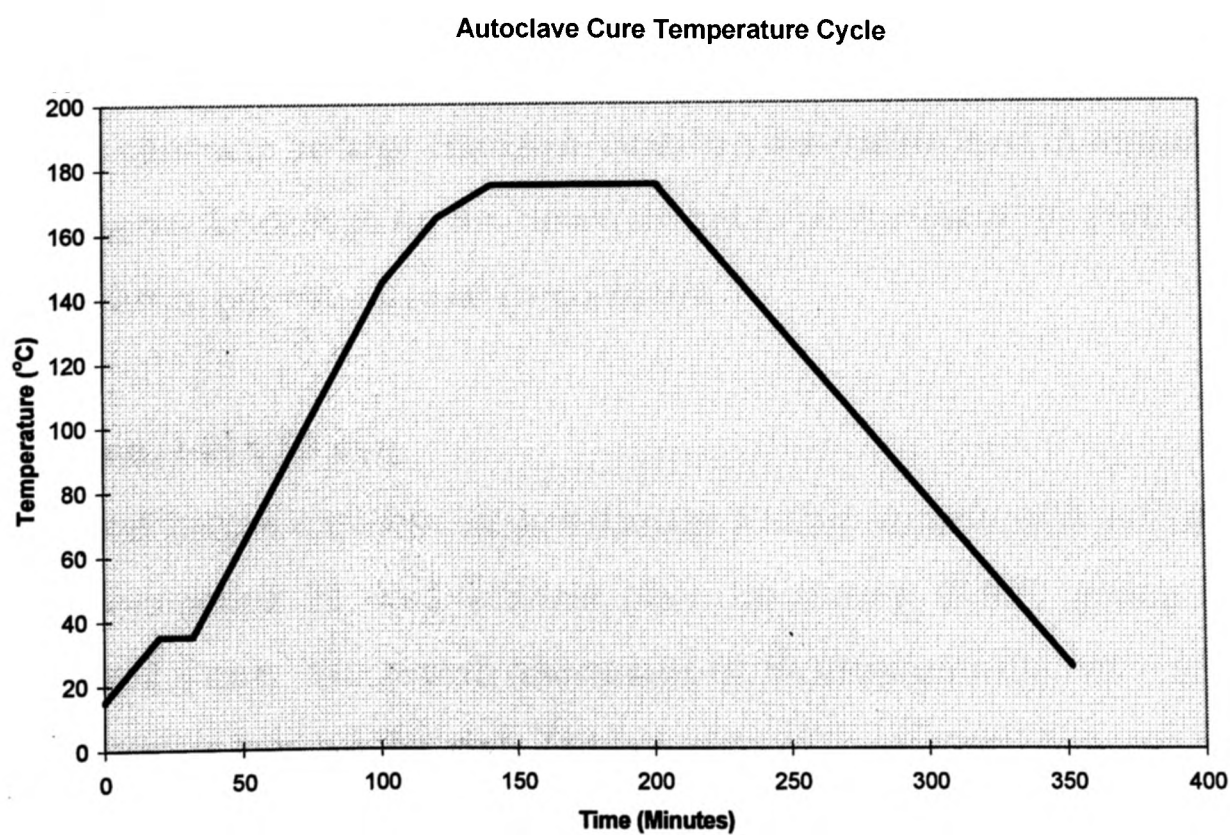


Figure 2.1

Rolls-Royce Data: High Tensile Carbon/Epoxy CFRP 100% 0° fibres

Density	ρ	1590 kg/m ³
Longitudinal modulus	E_1	139 GPa
Transverse modulus	E_2	9.4 GPa
Shear modulus	G_{12}	4.5 GPa
Poisson's ratio	ν_{12}	0.0209
Longitudinal tensile strength	T_1	1700 MPa
Transverse tensile strength	T_2	74 MPa
Longitudinal compressive strength	C_1	1000 MPa
Transverse compressive strength	C_2	237 MPa
In-plane shear strength	S_{12}	86 MPa

Table 2.1

2.1.3 Coupon Design

Two coupon designs were used for testing: a laminate of 29 plies; and one of 21. The coupon designs are summarised in Table 2.2 and illustrated in Figure 2.2. Dimensions are in mm.

29 ply Coupon, 50 x 50 mm

For compression after impact testing plain 50 mm square coupons were used. These had a lay-up of 29 unidirectional plies, alternating 0°/90°, giving a thickness of approximately 3.6 mm.

This size was chosen to be large enough to contain a reasonable level of impact damage while still remaining stocky enough for the behaviour not to be dominated by global buckling when tested under direct, pin-ended, axial compression.

21 ply Coupon, 140 x 85 mm

For tension and fatigue tests after impact damage a larger coupon with end tabs was needed. The lay-up comprised 21 unidirectional plies, alternating 0°/90°, giving a thickness of approximately 2.8 mm. The overall dimensions of the coupons were 140 x 85 mm with 37.5 x 85 mm aluminium tabs at either end. The unsupported gauge length between tabs was 65mm. The lay-up was such that the outside, 0°, plies on front and back were parallel to the long axis and in the same direction as the applied loading.

The two coupon designs were chosen to have almost the same slenderness when considering the fixity of the large coupon tabbed ends and the pin-ended testing of the small coupons. Thus under compressive loading the behaviour of the two coupons was expected to be very similar.

The large (140 x 85 mm) coupon was gripped by tabs glued to the ends. Aluminium plate of 0.1 inch (2.55 mm) thickness was used for the tabs. Before glueing the coupon and tab surfaces were abraded and de-greased. The glue used was a two part epoxy: Eccobond 45 with Catalyst 15. The glue was mixed in the ratio 3 parts Eccobond to 2 parts catalyst to give a formulation of medium rigidity. Five grammes of glue was used for each coupon. The glue was allowed to cure at room temperature for 24 hours.

Coupon Designs

Number of Plies	29	21
Laminate Lay-up	0°/90°	0°/90°
Dimensions	50 x 50 mm	140 x 85 mm
Thickness	3.6 mm	2.8 mm
Mass	14.1 g	53.15 g
Density	1565 kg/m ³	1595 kg/m ³

Table 2.2

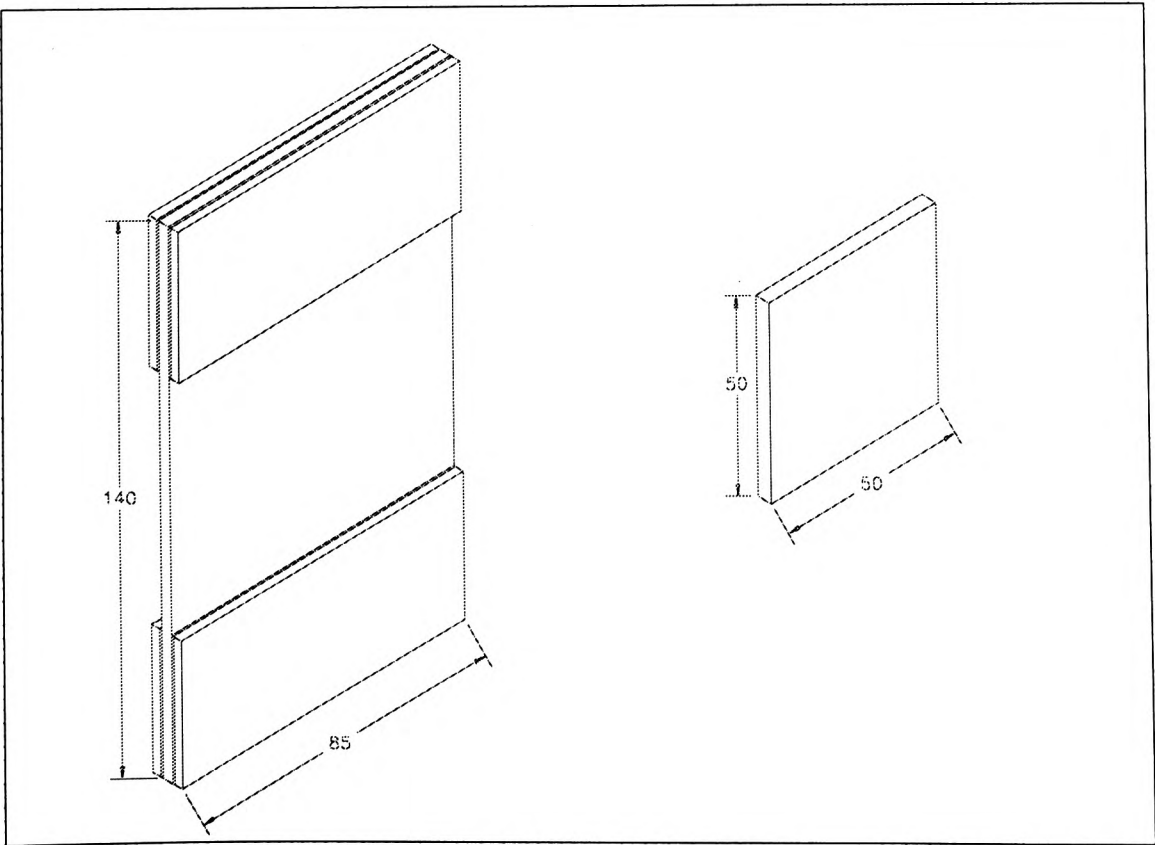


Figure 2.2

2.2 Structural Support

For impact testing the coupons were held loosely over a support block containing a drilled out circular window and struck centrally. The support block dimensions were 75 x 100 mm, depth 25 mm. The drilled out window was of diameter 45 mm with an edge radius of approximately 1mm. The support conditions for the coupons can be approximated as a circular simple support of 45 mm diameter. Figure 2.3 shows how the 140 x 85 mm, 21 ply coupons were positioned over the support for impact. The 50 x 50 mm, 29 ply coupons were held in place by reference pins fitted to the same support block.

Structural Support for Impact Testing

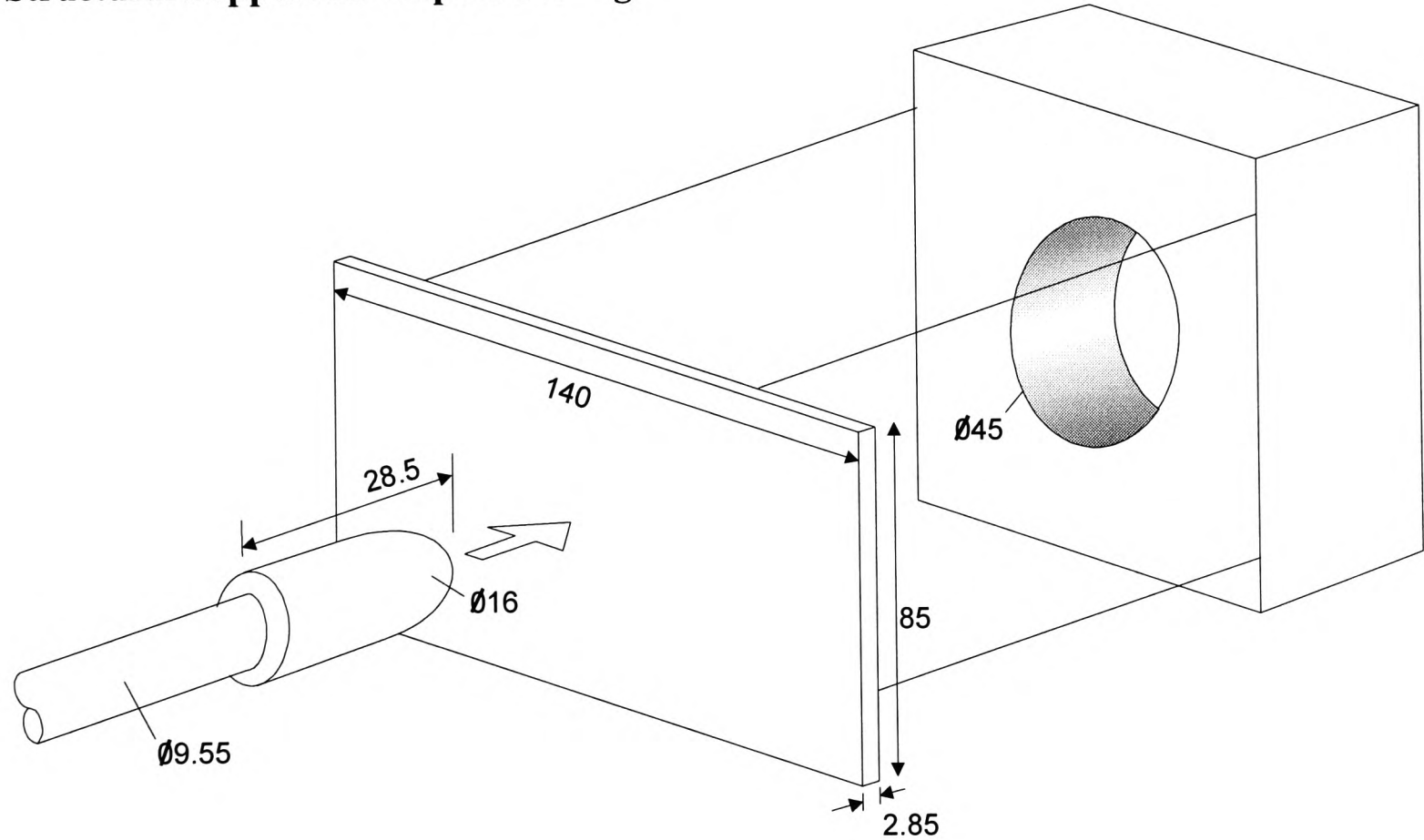


Figure 2.3

2.3 Low Speed Impact Tests

2.3.1 Type of Impact

The type of impact used to produce damage in the coupons was a low speed, blunt nosed, hard body impact with an angle of incidence of 90°, i.e. perpendicular to the coupons.

This type of impact is typical of standard industry tests and represents a “tool drop”. The damage resulting may also be typical of other, higher speed, hard body aero-engine impacts such as pebbles or hailstones.

To differentiate between high and low speed impact damage and quasi-static indentation damage further tests were needed. These tests used the same coupons and support as the low speed impact tests and are described in sections 2.4 and 2.5.

It is important to distinguish the hard body impact damage investigated here from the soft body (bird strike) type of damage. Bird strikes are high speed events, with high energy and high momentum (up to 7 kJ for a 4 lb bird at 200 mph). However, the local peak stresses are lower than those occurring in hard body impacts because the projectile itself (the bird) has almost no structural strength.

Robinson and Davies [7] conducted an investigation into the effects of impactor mass and specimen geometry on the low velocity impact of laminated composites. They defined a low velocity impact as one in which stress waves play no significant part in the stress distribution during impact. Stress waves propagate at the speed of sound c in the specimen. This is approximately 2000 ms^{-1} for common epoxy composites. The compressive strain ε under the impactor therefore may be estimated from the impact velocity v_0 as:

$$\varepsilon = v_0 / c \quad 2.1$$

With typical failure strains between 0.5 and 1% it can be calculated that the transition to a stress wave dominated impact occurs at impact velocities of between 10 and 20 ms^{-1} .

2.3.2 Industry Standard Impact Tests

Typical industry standard impact damage tests are detailed in Table 2.3. These tests are used for comparison of materials. They allow the relative tolerance of composite materials to impact damage to be measured.

In the Airbus Industrie Test Method 1.0010 coupons of the sample material are submitted to a range of impact energies using a drop weight [8]. Maximum impact velocity is 5 m/s (50 J with a 6kg drop weight impactor). After each test the permanent indentation is measured and this is used to determine the minimum energy to cause *barely visible impact damage (BVID)*, E_{bvid} . BVID is defined in this test method as a permanent indentation of 0.3mm. Coupons are then taken on for “compression after impact” (CAI) testing.

The Boeing Advanced Composite Compression Test [9] uses a coupon size and impact support window very similar to that used in the Airbus Industrie test. One difference is that the drop weight is of fixed mass and is used to strike an initially stationary impactor.

Royal Aerospace Establishment CRAG Methods [10] use a coupon which is clamped between two circular rings of 100 mm internal diameter for impact. This is a considerably more rigid mounting than the previous two tests described. The impactor is also more penetrating, with a tip radius of only 5 mm.

Industry Standard Impact Tests

	Airbus Industrie Test Method 1.0010	Boeing Advanced Composite Compression Test	Royal Aerospace Establishment - CRAG Methods
Coupon Size (mm)	150 x 100 x 4	152 x 102	180 x 50 x 3
Impact Support Window Size (mm)	125 x 75 simply supported	127 x 76.2 simply supported	100 diameter clamped
Drop Weight Mass (kg)	1 to 3 or 4 to 6 (4 to 50 J)	5.44 + 0.79 impactor	unspecified
Impactor Tip Radius (mm)	8	7.9	5

Table 2.3

The support condition used for the impact tests conducted for this thesis is more rigid than the three industrial tests examined here. Although the CRAG methods support is clamped at the edges, the 100 mm diameter window (compared to 45 mm diameter) makes it approximately twice as flexible.

2.3.3 Impactor Design

Three different impactor designs were used for the coupon impact tests. Impactor 1 was used for the 50 x 50 mm, 29 ply coupons. Impactor 2 was identical to impactor 1 except for the addition of a 16 mm diameter tip, and was used for the 140 x 85 mm, 21 ply coupons. The larger tip diameter made the impact damage less penetrating. Impactor 3 was used for tests on the 21 ply coupons where the impact force/time history was recorded by the use of strain gauges bonded to the impactor. Table 2.4 details the impactor specifications. The “time period” is the length of time for a stress wave to travel, and return, over the impactor length.

Impactor Specifications

Impactor	1	2	3
Material	Steel	Steel	Titanium
Wavespeed (m/s)	5032	5032	5100 m/s (calculated) 4880 m/s (measured)
Length (mm)	329	336	500
Time Period (ms)	0.131	0.133	0.205
Diameter (mm)	10	10	10
Tip Radius (mm)	5	8	8
Mass (g)	208	236	256

Table 2.4

2.3.4 Gas Gun

The impactors were mounted on “oilite” bearings and given a velocity by being struck by a gas gun projectile. Figure 2.4 shows the arrangement. Impact velocity and therefore energy was controlled by the monitoring of the gas gun firing pressure with a digital pressure gauge. Firing was controlled by manually opening the firing valve. A latch device on the impactor prevented secondary impacts occurring on rebound.

2.3.5 Measurement of Velocity

The impactor velocity was measured by the use of LED/LDR (light emitting diode/light dependent resistor) gates, separated by 50 mm, connected to a digital counter/timer. The timer gave the elapsed time for the impactor to pass through the gates and thus the velocity could be calculated. The gate, coupon support and impactor 1 are illustrated in Figure 2.5.

The incident impact kinetic energy, E , was calculated directly from the incident impact velocity, v_0 , and the known mass of the impactor, m .

$$E=\frac{1}{2}mv_0^2$$

2.2

Gas Gun Schematic

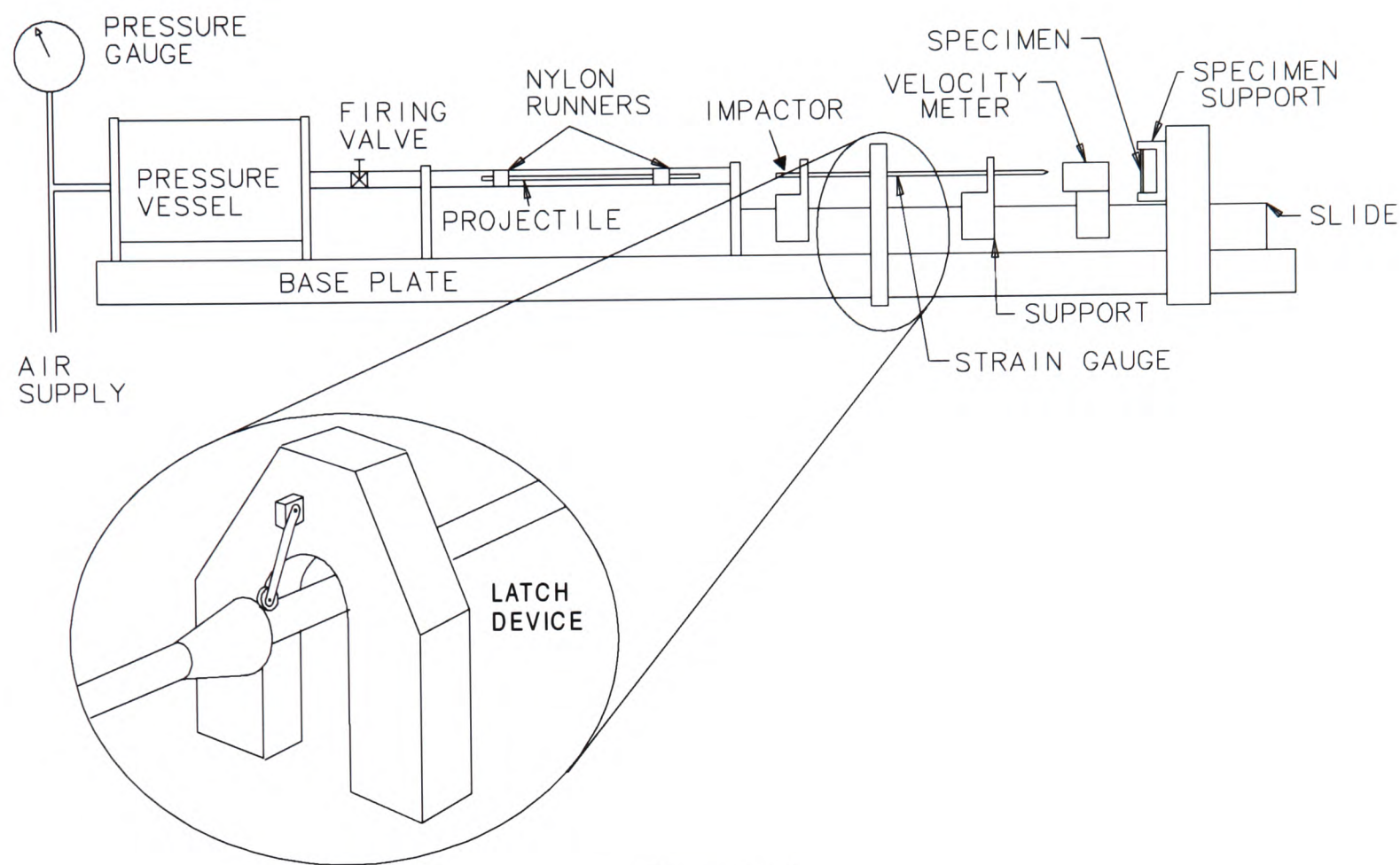


Figure 2.4

Impactor, Support Block and Velocity Gate

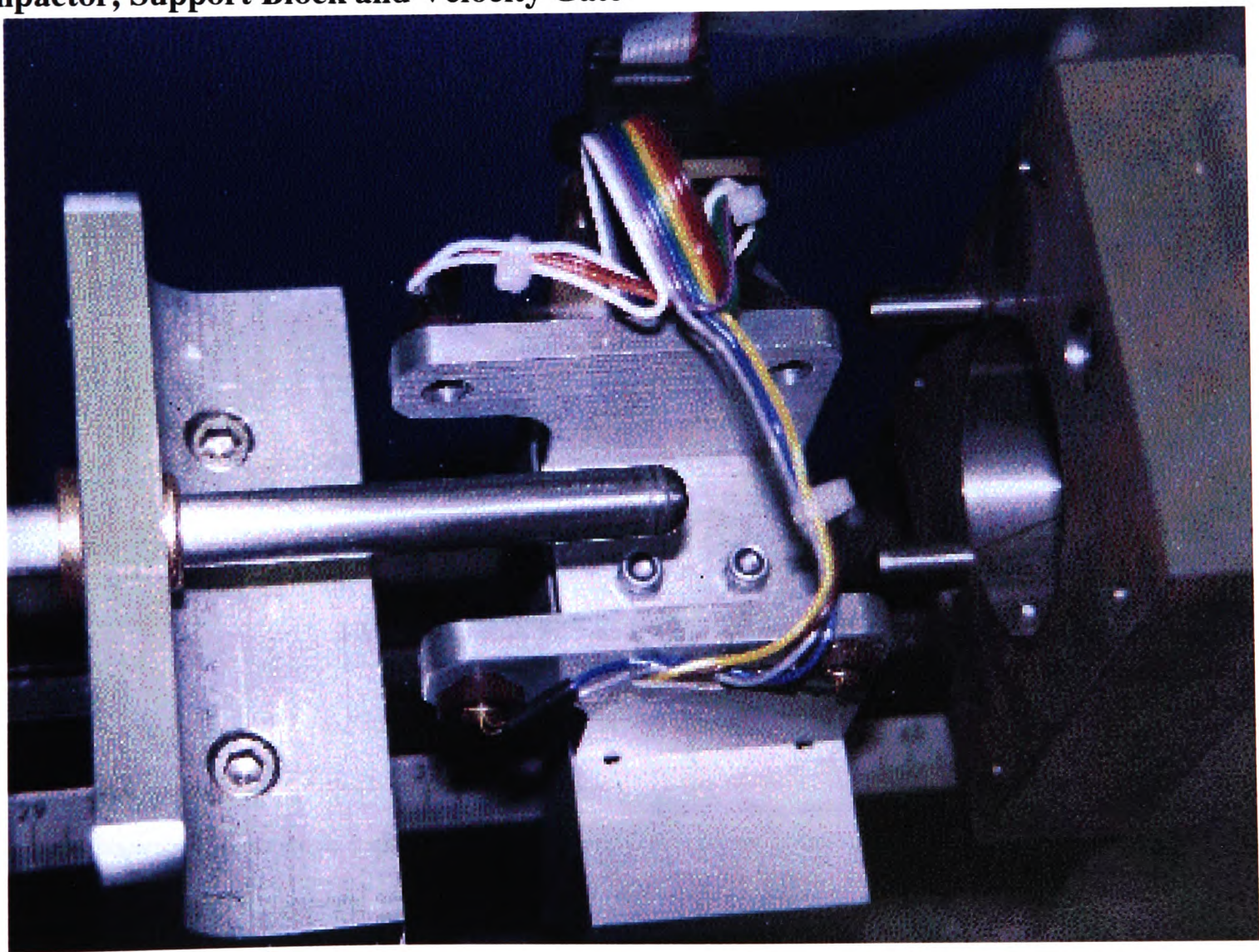


Figure 2.5

2.3.6 Measurement of Impact Force

Force/time histories for the impact event were measured by the use of impactor 3 which was fitted with strain gauges at mid-length. The strain gauge signals were amplified and recorded on a high-speed “Rapid Systems” data-acquisition system. The sampling rate was 2 MHz. Triggering for the initiation of data recording was taken from the second LED gate of the velocity measuring gates. The system was statically calibrated on an Instron mechanical testing machine.

2.3.7 Number of Tests

In total 83 low speed impact tests were conducted. Of these:

- 37 were on the 29 ply, 50 x 50 mm coupon using impactor 1
- 40 were on the 21 ply, 140 x 85 mm coupon using impactor 2
- 6 were on the 21 ply, 140 x 85 mm coupon using impactor 3 to measure impact force

2.4 High Speed Impact Tests

High speed impact tests were conducted to compare the type of damage with that resulting from low speed impacts.

2.4.1 Projectile

The projectile used for high speed impact tests was a 6 mm diameter steel ball bearing of mass 0.9 grammes.

2.4.2 Gas Gun

The projectile was fired by mounting it in a sabot which was fired by a compressed gas gun. The gas gun consisted of a reservoir, a firing clamp and a 1.5 m long, 12.7 mm bore barrel.

Reservoir pressure was preset to control the gun velocity, and the sabot was held by gas operated clamps. The operating gas was nitrogen and, upon release of the sabot clamps, this was allowed to expand freely from the gun reservoir to accelerate the sabot and projectile. At the end of the gas gun barrel a sabot separator was used to stop the sabot but allow the projectile to continue unimpeded to strike the target coupon.

2.4.3 Measurement of Velocity

The projectile velocity was measured by LED/LDR timing gates.

2.4.4 Coupon Support

The same support block used for the low speed impacts was also used for the high speed impact tests.

2.4.5 Number of Tests

Six high speed impact tests were conducted: five, at varying velocities, on the 21 ply coupons and one on a 29 ply coupon at a higher velocity.

2.5 Quasi-static Indentation Tests

Quasi-static indentation tests were conducted with the following aims:

- i) Determine the damage produced in order to make comparisons with impact damage.
- ii) Measure the elastic and inelastic stiffness of the coupons.
- iii) Determine peak force and work done at increasing levels of quasi-static indentation damage.

Tests were conducted on both the 29 ply and 21 ply coupons. The support used was the same as for the impact tests.

2.5.1 Indenter

For the 29 ply coupons a ball bearing of 10 mm diameter was used as the indenter. This was chosen to correspond with the 5mm tip radius of impactor 1 used for the low speed impacts on the 29 ply coupons.

For the 21 ply coupons an 8mm tip radius indenter was used. This was the same tip used for impactor 2 and 3 in the low speed impacts on the 21 ply coupons.

2.5.2 Displacement

Displacement of the indenter relative to the coupon support block was measured by use of a linear voltage displacement transducer (LVDT) rigidly fixed to the indenter loading point.

The LVDT (RDP Electronics D2/200A 3999) was calibrated by reference to a Mitutoyo Digimatic RS Digital Vernier Caliper 600-880. It was determined to have an output of 0.1925 V/mm over a 4 mm linear range.

The LVDT power source was a Farnell Stabilised Power Supply L30B supplying $6.00 \text{ V} \pm 0.01 \text{ V}$ via a RDP LVDT splitter junction box. The junction box also connected the LVDT to the testing machine via a 1kHz low-pass filter which removed noise from the signal.

2.5.3 Testing Machine

A Hounsfield Tensometer testing machine was used. This is an electrically powered, screw driven testing machine. Tensometer control software (Servocon Systems Program) was used to control the testing machine and also to log data. The indentation tests were stroke controlled and conducted at an indentation rate of 1 mm/min.

2.5.4 Loading/Unloading Sequences

Four quasi-static indentation tests were conducted. Two were continuous displacement controlled tests to failure on both the 21 and 29 ply coupons. The second two tests involved repeated loading and unloading at increasing maximum indentations. These tests gave the unloading force/displacement curves as well as the reloading curves. After each loading increment the 21 ply coupon was ultrasonically C-scanned (section 2.6.3) to measure the extent of delamination and the permanent indentation was measured.

2.6 Evaluation of Damage Measurement Techniques

Numerous techniques exist for measuring the severity of impact damage in composites. In this section the various techniques are evaluated. In subsections 2.6.1 to 2.6.8 techniques used in the current work are evaluated. Subsections 2.6.9 to 2.6.15 describe further techniques which were considered for use but rejected.

2.6.1 Visual Inspection

Visual observation is the most straightforward technique for detecting impact damage and assessing its seriousness. However it has obvious limitations. The inspector may only be able to see one side of a laminate component, perhaps only from a distance of some metres. Damage is only visible as an indentation or perforation; internal laminate damage such as delamination

is not visible.

Aerospatiale [11] conducted an investigation into the level of damage that could be reliably detected by visual inspection. Three kinds of inspection were checked:

1. Quick visual inspection: for each 1 m² panel, the operator (13 operators participated) had 10 seconds to make a quick visual inspection standing 2 m away from the panels; after this time only the “sure” results were recorded.
2. Long visual inspection: the same approach but without time limitation.
3. Detailed visual inspection: the inspection consisted of a close examination of the panels from a distance of about 50 cm using grazing lighting as required. The examination time was neither limited nor measured.

The first two distant inspections revealed that for an indentation depth between 0.20 and 0.23 mm the detection rate was 95 %. For the third inspection the average detected indentation depth was about 0.1 mm, but there was also an average of two false alarms per square metre which was prejudicial to in-service inspection. Following these results the company chose the following indentation depth values as standards for detailed inspection to detect BVID:

Mean value:	0.3 mm
99 % probability with 95 % confidence:	0.5 mm

In the Airbus Industrie Test Method 1.0010 [8] BVID is simply defined as an indentation of 0.3 mm.

2.6.2 Indentation

The permanent indentation left after a hard body impact is one indication of damage severity. It is the only indication that can be visually assessed and externally measured without the use of non-destructive or destructive evaluation techniques.

For the 140 x 85 mm, 21 ply coupons tested here the permanent indentation after impact was measured with the probe of a digital caliper.

The tip of the probe had a 2 mm x 1 mm flat end and measured the depth between two points separated by 16 mm. This width was sufficient to bridge the most severe indentation considered

in the impact tests.

The Airbus Industrie Test Method 1.0010 [8] standard for measuring permanent indentation uses five depth gauge measuring points. One point is at the deepest part of the indented area and the other four points are taken at 20 mm from the indentation in orthogonal directions. The indented depth is taken as the central depth subtracted from the average depth of the other four points.

2.6.3 Ultrasonic C-Scan

The ultrasonic C-scan is a useful technique for determining the overall extent of delaminations within a damaged composite laminate. It produces a 2-dimensional image of the specimen showing the shape, size and location of the delaminations. It does not however show the depth or shape of individual delaminations in a multi-ply laminate.

The ultrasonic scanner uses a piezoelectric transducer which is a combined emitter and receiver. This delivers a short pulse of ultrasound and then detects the reflected pulses. The ultrasound pulse will be reflected at interfaces of differing acoustic impedance.

For C-scanning the transmission medium used is water. Specimens are placed in a bath of water and the scanning head is also submerged and scans over the specimen surface. A flat glass plate is placed beneath the specimen and the scanning equipment is set up so that it detects reflected pulses from the glass plate which have been transmitted, and retransmitted, through the specimen.

Ultrasonic C-scan was the primary method used for assessing the damage level in impact damaged coupons tested for this thesis.

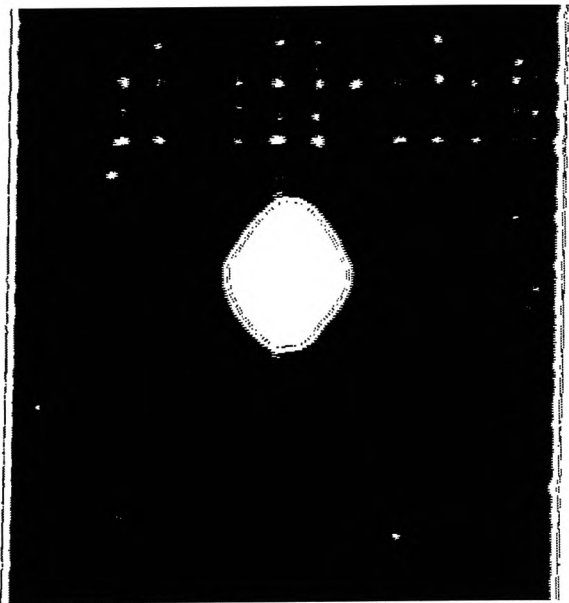
The equipment used for C-scanning the CFRP coupons was manufactured by Ultrasonic Sciences Ltd. It included an Ultrasonic Module UIM.73, a scanning head Motor Drive SMD 03 and a PPA15 Pulser/Pre-Amplifier. The Pulser/Receiver used was a Technisonic ISL-1003-HR operating at 10MHz.

Pre-amp and main gain were set so that the transmitted signal through the undamaged laminate

corresponded to approximately the full (100 %) amplification range. Any delamination damage then showed up as much less than 100% signal.

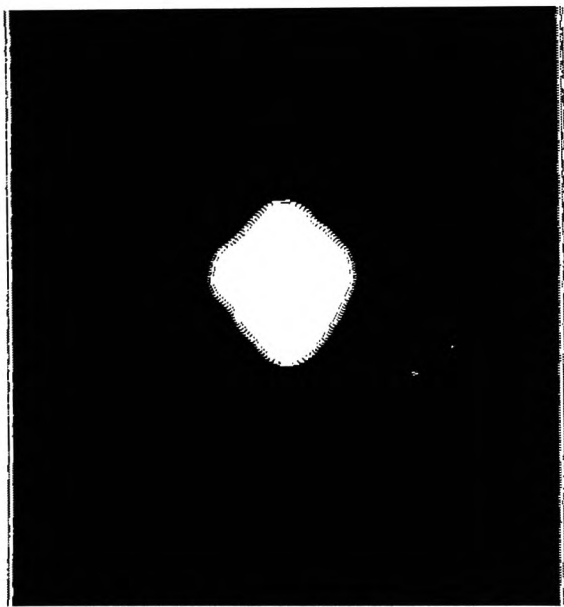
The C-scan image shows a 2 dimensional picture of the overall extent of the delamination. On the image there is a fairly precise cut off between undamaged and delaminated areas. The polygon feature in the USL software measures the area of a polygon constructed on the screen and this was used to estimate the overall area of delamination.

Figures 2.6, 2.7 and 2.8 show C-scans of 21 ply coupons with increasing levels of impact damage. The coupons are referred to as A, B and C.



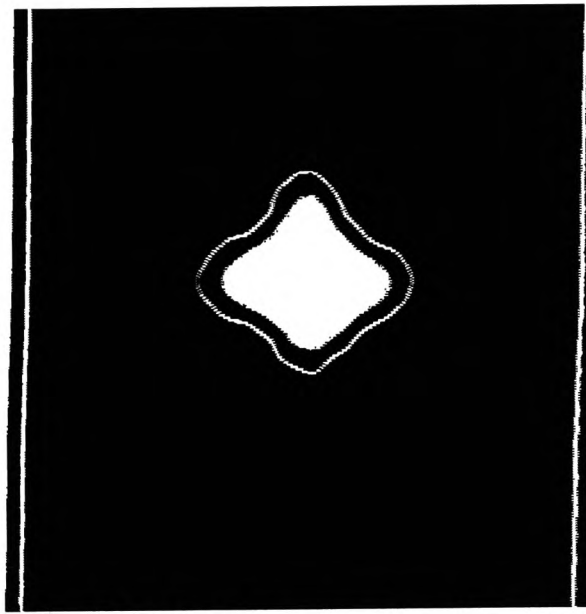
Coupon A, Impact Energy 5.11 J, Delamination Area 295 mm², Indent 0.05 mm

Figure 2.6



Coupon B, Impact Energy 7.35 J, Delamination Area 352 mm², Indent 0.06 mm

Figure 2.7



Coupon C, Impact Energy 9.86 J, Delamination Area 568 mm², Indent 0.16 mm

Figure 2.8

2.6.4 Ultrasonic B-scan

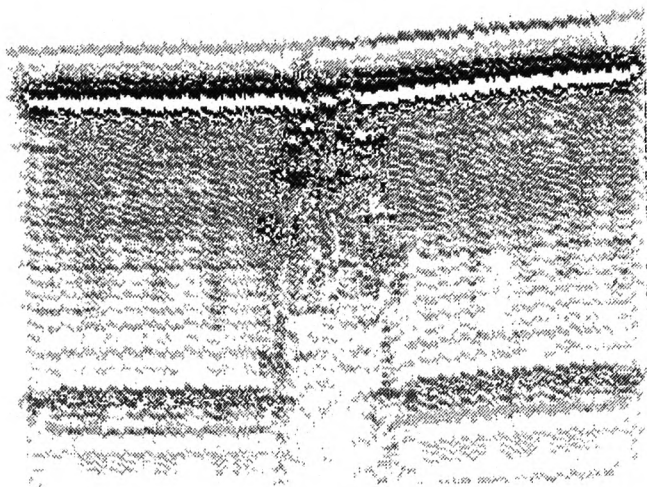
The ultrasonic B-scan is a technique that produces a cross-sectional scan through the specimen. Usually a C-scan is conducted first and the C-scan image is used to determine the line of the B-scan section.

Whereas the C-scan simply records the amplitude of one reflection of the emitted pulse the B-scan records a range of reflection times and therefore depths in order to construct a depth profile.

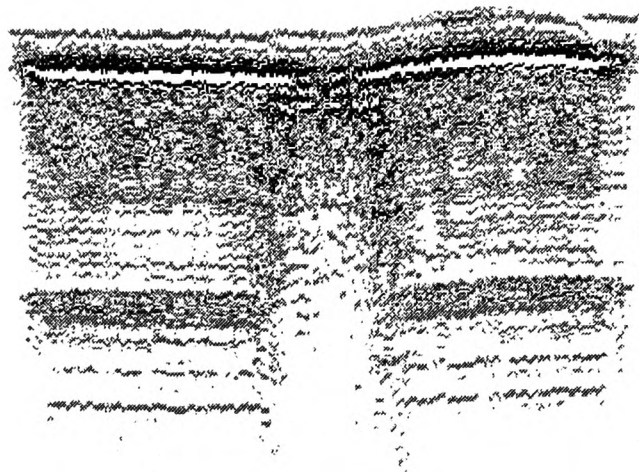
When B-scanning a composite laminate the section of reflected waveform examined ranges from the first reflection from the top of the specimen down to the reflection from the bottom. Any major reflections between these two pulses correspond to changes in acoustic impedance (density) and are therefore due to damage or delamination.

The B-scan shows the cross-sectional extent of the damage below the scanner. However since the damage itself interferes with the signal no more information is available about the material state below a damaged area or delamination.

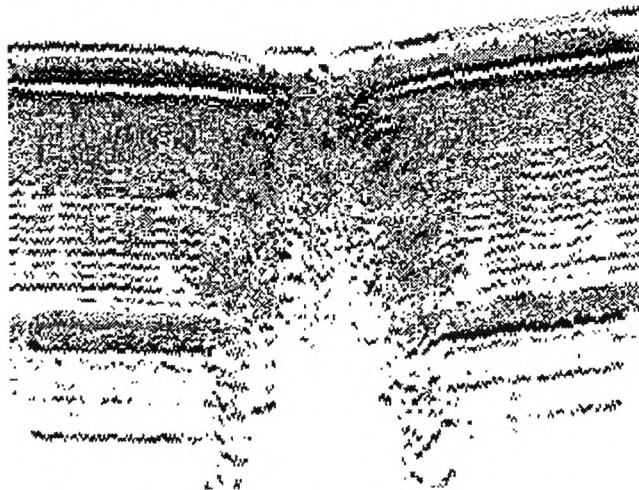
Figure 2.9 shows B-scans conducted across the impact damage zone of coupons A, B and C. The width of the scans corresponds to the 85 mm width of the coupons. The depth to the first major reflection corresponds to the 2.8 mm thickness of the coupons and consequently the B-scan images have differing horizontal and vertical scales.



Coupon A, 5.11 J, 295 mm²



Coupon B, 7.35 J, 352 mm²



Coupon C, 9.86 J, 568 mm²

Ultrasonic B-Scan Cross-Sections of Impact Damaged Coupons

Figure 2.9

2.6.5 X-Radiography

The X-ray radiographic technique does not lend itself to general inspection but is more suited to high-resolution damage characterization. Radiography uses the differential absorption of radiation by the specimen under examination to produce an image on a screen or film underneath. To improve the contrast between defects and undamaged laminate a penetrant such as zinc iodide is generally applied to the specimen before examination. Damage such as matrix

cracking, delamination and extensive fibre fracture may all be visible on the penetrant enhanced X-ray image.

X-radiation is electromagnetic radiation of high quantum energy (greater than 1 keV). It is produced by bombarding a metal target with high-speed electrons which are emitted from a heated cathode by thermionic emission and accelerated towards the positive anode and target. On striking the target the electrons lose most (about 99%) of their energy in low-energy collisions with target atoms resulting in a substantial heating of the target. The rest of the electron energy reappears as X-radiation.

The intensity of an X-ray beam will decrease with distance from source due to divergence. For spherical wavefronts from a point source this will follow the inverse square law. When the X-ray beam traverses a medium the photons may be scattered or absorbed in the medium leading to an overall attenuation of the beam. Variations in attenuation will produce an image on photographic film placed behind the specimen.

Several factors affect the image quality obtainable from X-radiography:

- i) Tube voltage. This should generally be in the range 10-50 kV for good image contrast with carbon epoxy composites which are relatively transparent to X-rays.
- ii) Tube current. Increasing tube current gives an overall increase in intensity of both incident and transmitted beams.
- iii) Exposure time. This determines the density or blackening of the film. Longer exposure times will improve radiographic contrast until overexposure.
- iv) Beam size. Narrow beams improve contrast by reducing scatter, which may be further eliminated using special grids.
- v) Focal spot size. A small focal spot produces sharper images.
- vi) Filtration. This reduces unwanted low-energy radiation and scatter giving some improvement in contrast.

Generally radiography is used to detect features of a component or assembly that exhibit a difference in thickness or physical density as compared to surrounding material. Large differences are more easily detected than small ones. Specifically, radiography can detect those

features that have an appreciable absorption parallel to the radiation beam. Discontinuities such as voids in composites can only be detected if they are of appreciable size in relation to specimen thickness and exhibit an absorption at least 2% different from that of the surrounding material. Certain types of flaws are difficult to detect by radiography. Cracks cannot be detected unless they are parallel to the radiation beam. Tight cracks in thick sections usually cannot be detected at all, even when properly oriented. Delaminations in composites are nearly impossible to detect with radiography since they are generally perpendicular to the X-ray beam and do not produce a difference in absorption.

Conventional radiography therefore fails, due to the lack of contrast, to detect the delaminations and fine translaminar/interlaminar cracks usually caused by impact damage. To be viable as an inspection technique for impact damage detection, subject contrast must therefore be enhanced. This is achieved by the introduction of a contrast medium, usually referred to as a radio-opaque penetrant.

Birt [12] has investigated the applicability of radiography to the inspection of composites and in particular the suitability of various penetrants. A zinc iodide solution is generally regarded as the best penetrant. It has good X-ray opacity, does not affect the composite properties and will rapidly penetrate tight cracks even in the presence of moisture. A typical penetrant solution recipe is:

Zinc Iodide Solution

60g ZnI_2

10 ml propan-2-ol

100 ml water

1 ml Kodak photoflo wetting agent

Radiography with a penetrant will allow identification of the location and size of internal delaminations and matrix cracks so long as these internal cracks are open to the specimen surface and it is possible for the penetrant solution to ingress. To obtain full penetration complete immersion of the specimen in the penetrant solution is preferred. Injection, brushing or spraying are alternative methods of penetrant application. Of these only injection comes close in performance terms to immersion.

2.6.6 X-ray Computer Microtomography

X-ray computer tomography (CT) is a non-destructive testing technique that allows 3-dimensional imaging of the internal structure of a specimen. It is a useful technique for investigating impact damaged CFRP because it reveals the distribution of delaminations and matrix cracks without the need for physical sectioning.

X-ray microtomography is a miniaturised form of CT scanning, used to create 2D and 3D X-ray attenuation maps of specimens up to a few centimetres in size, with a resolution typically of the order of tens of microns. In biology, it is especially useful for the 3D study of hard tissue, including structure and mineral concentration. It is also used in the materials field, particularly to look at composite materials and for the non-destructive imaging of fracture formation.

First generation microtomography systems are similar to the original system developed by Elliott and Dover in 1981, whereby a specimen of up to a few mm diameter is stepped through a narrow (typically 15 micron) x-ray beam. X-ray attenuation is recorded at each step and the specimen rotated slightly and the process repeated. Projections are continued until the specimen has been rotated about 360 degrees, typically using 251 projections with 128 steps in each. A parallel filtered back-projection algorithm is used to reconstruct a 2D X-ray attenuation map from the X-ray attenuation data. Typical data acquisition time is around 12 hours per slice. Full 3D images are made by imaging adjacent slices along the rotation axis, in which case data acquisition can take more than a week. An advantage of this system is that an energy dispersive detector can be used, such that full attenuation spectra are recorded at each step, allowing for elemental analysis of the specimen.

For examining impact damaged coupons for this thesis a fourth generation scanner was used. This was developed in the Department of Biophysics in Relation to Dentistry, Queen Mary and Westfield College, University of London by Davis and Elliott [13]. The scanner is a modified third generation scanner, whereby an area detector is used to record whole projections simultaneously. Data collection is thus orders of magnitudes faster, with complete 3D data sets collectable in less than one hour. The fourth generation scanner uses a time delay integration (TDI) mode CCD readout on the camera, synchronised to a novel scanning stage which moves the specimen through the X-ray beam whilst maintaining the camera aspect (Figure 2.10). This eliminates ring artefacts in the reconstructed image (a common problem with third generation

systems) and allows specimens larger than the CCD imaging area to be scanned without optical de-magnification and consequent loss of efficiency. The system can scan specimens up to 5 cm across and 5cm in depth (in 5 mm increments), with a cubic pixel size of 58 microns. Currently a fan-beam reconstruction algorithm is used, the geometry of the system being such that this does not introduce significant errors.

For the X-ray energies used in CT, the dominant attenuation mechanism is Compton scattering which is proportional to material density. Therefore it is a reasonable approximation to use CT images as a density map of the object.

Impact damaged CFRP coupons A, B and C were investigated by CT. For scanning, a 35 mm diameter disc of material containing the impact damaged zone was cut from each coupon. Scanning and processing produced image data in the form of grey scale pixels. Each pixel represented a volume of 58 x 58 x 58 microns cubed and the database was 610 x 610 x 66 pixels. This represents a cuboid 35.4 x 35.4 x 3.8 mm, which is just large enough to contain the disc.

Figures 2.11, 2.12 and 2.13 show the effect of increasing incident impact energy on the size and distribution of matrix cracks and delaminations within coupons A, B and C.

The images are “slices” through the centre of the impact damaged zone taken at 10 pixel intervals (0.58 mm). To enhance the definition of cracks the greyscale of each pixel has been averaged over a 10 pixel depth (0.58 mm). This was found to give a much sharper image of the damage than presenting a single pixel depth “slice”.

The tomograph images presented here were produced by Dr. Graham Davis of the Department of Biophysics in Relation to Dentistry, Queen Mary and Westfield College, University of London.

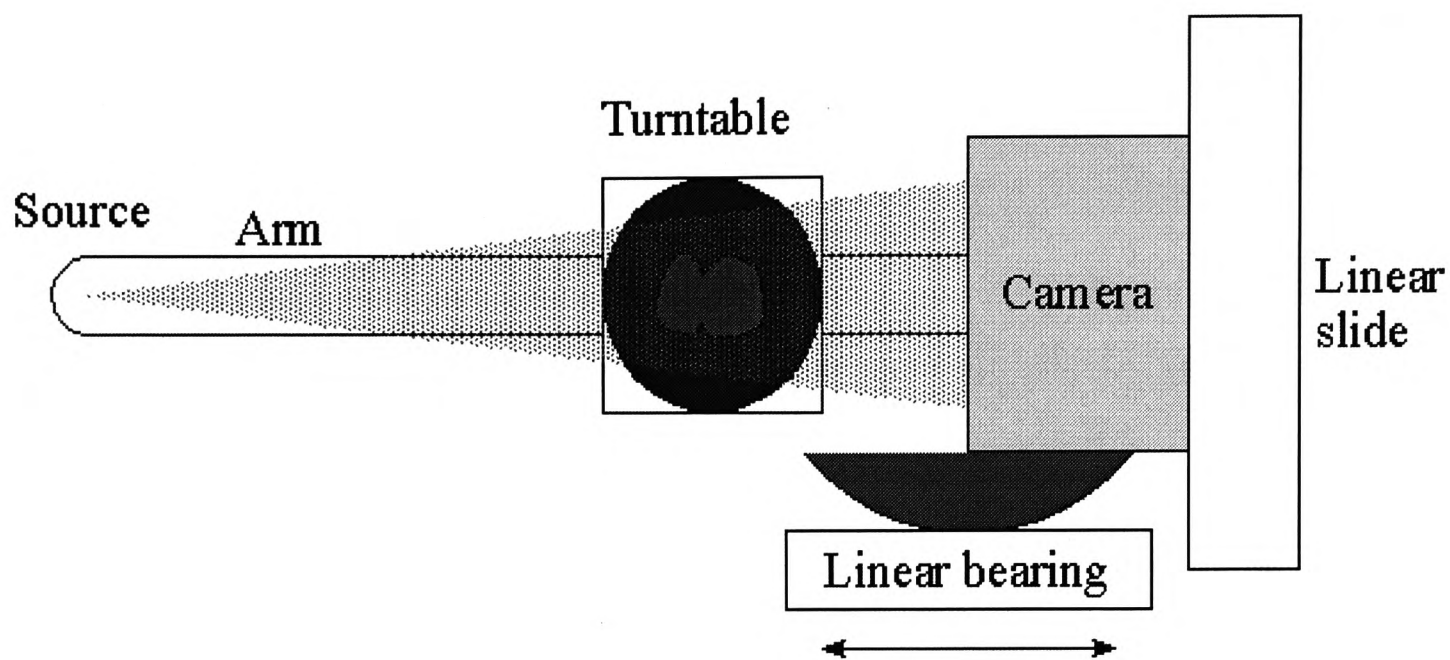
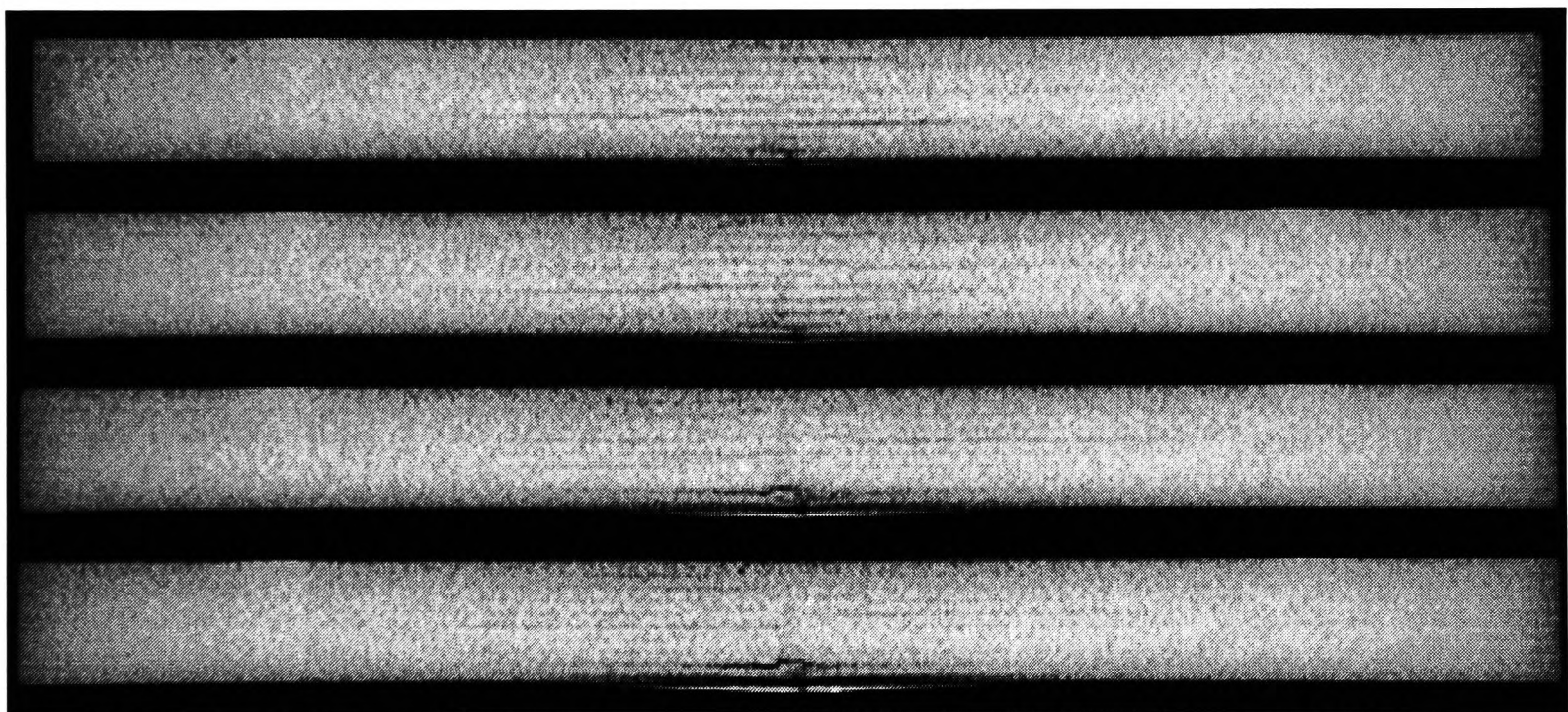
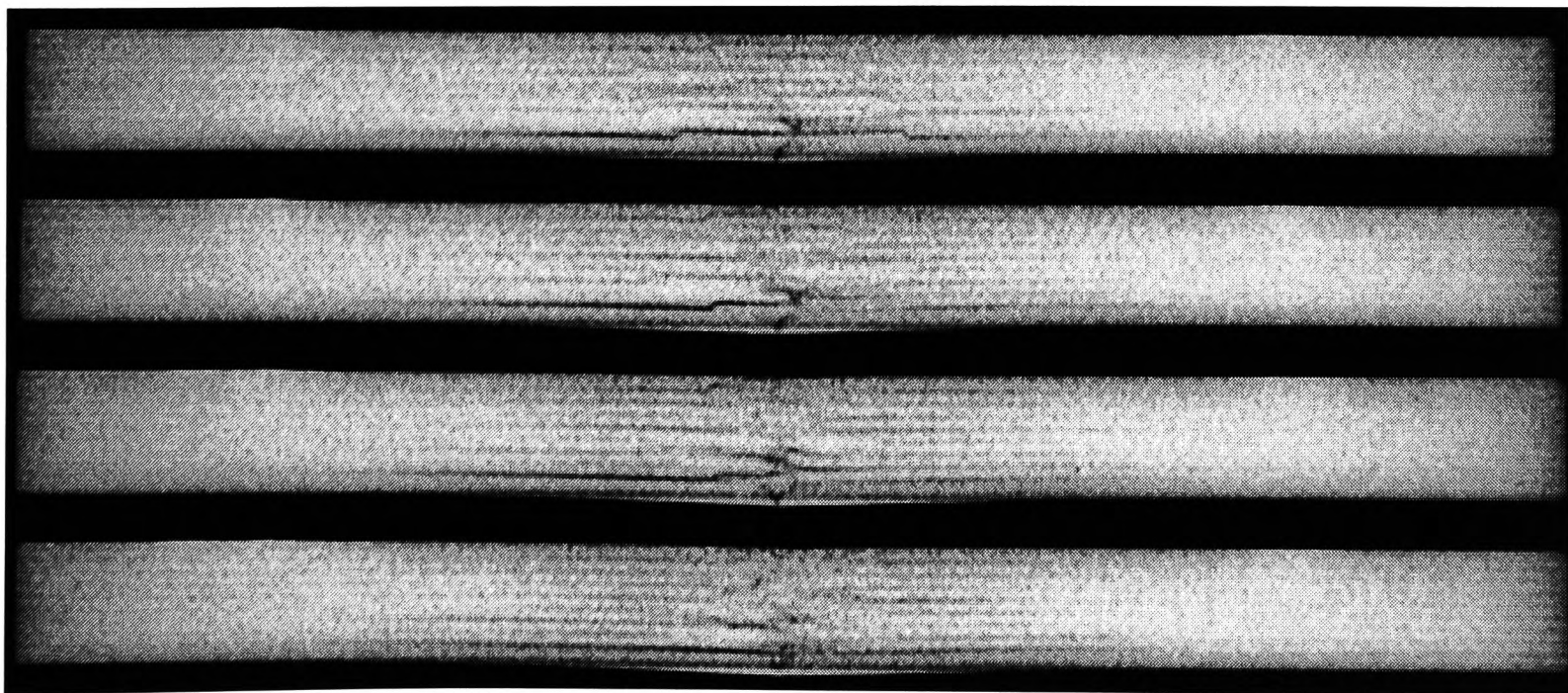


Figure 2.10



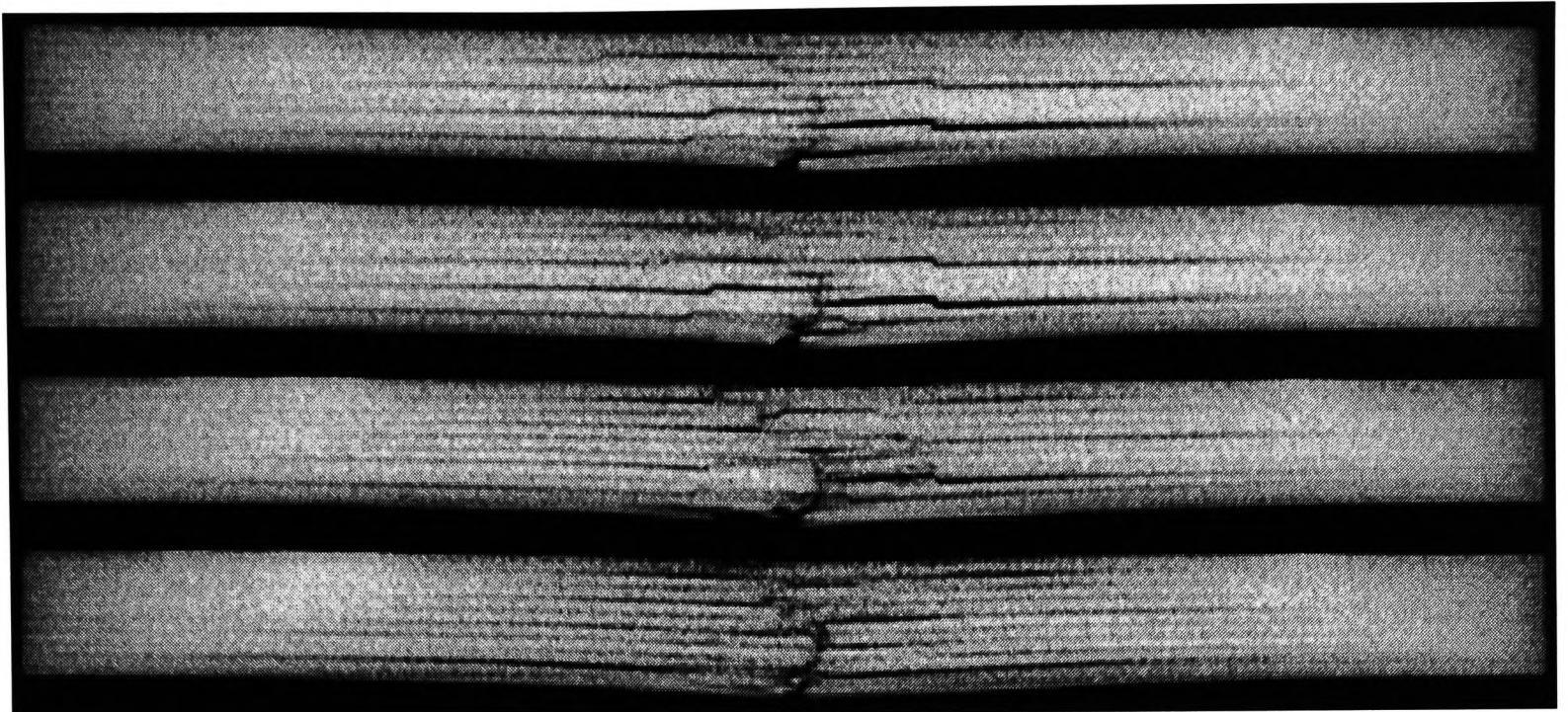
Coupon A, Impact Energy 5.11 J, Delamination Area 295 mm², Indent 0.05 mm

Figure 2.11



Coupon B, Impact Energy 7.35 J, Delamination Area 352 mm², Indent 0.06 mm

Figure 2.12



Coupon C, Impact Energy 9.86 J, Delamination Area 568 mm², Indent 0.16 mm

Figure 2.13

In the CT images shown (Figures 2.11, 2.12 and 2.13) the location and width of delaminations throughout the laminates are quite clear. Fibre breakages are also visible.

X-ray tomography of the resolution used here is currently limited to very small specimens (a 35 mm diameter disc in this case). The technique in general is also unsuitable for objects which have an aspect ratio greater than 15:1. Large, flat, structures are therefore not suitable for X-ray CT.

2.6.7 Optical Microscopy

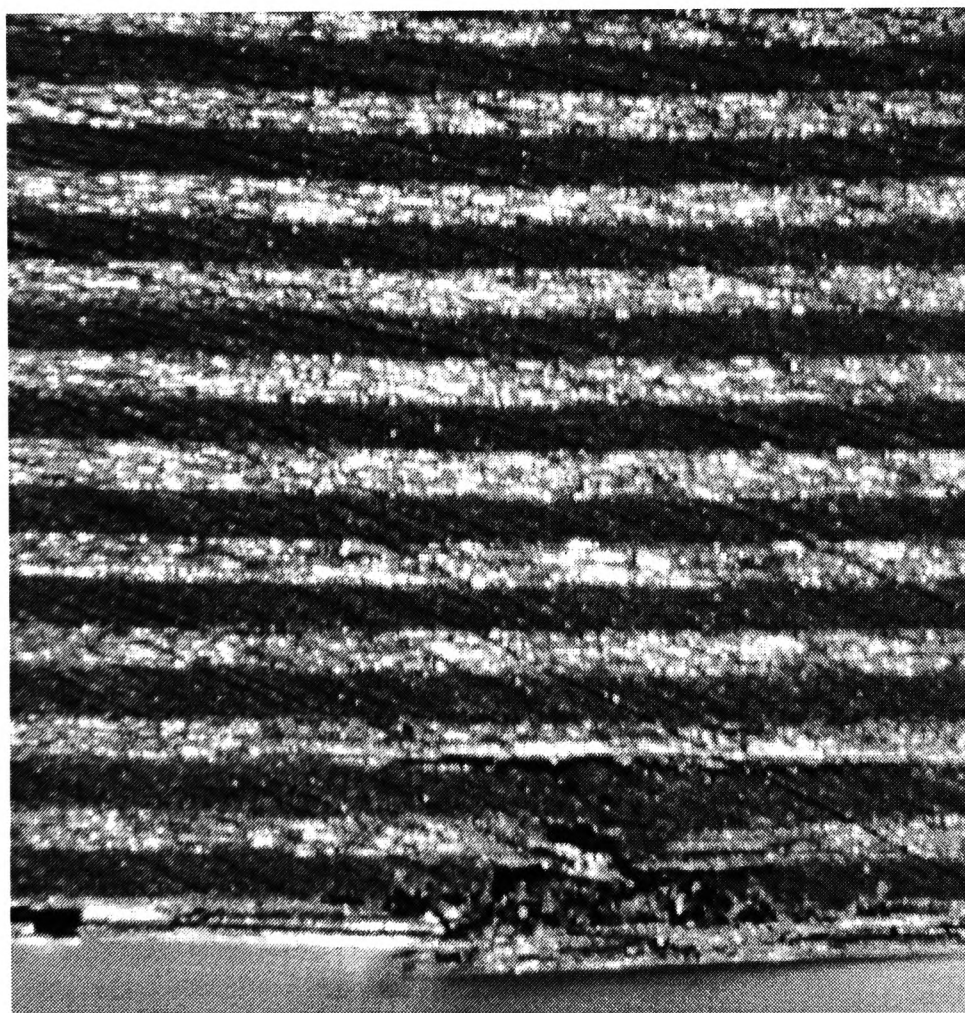
Generally a laminate specimen is sectioned, ground and polished before being viewed under an optical microscope. In the cross-section fibre fracture, matrix cracking and delamination can be readily identified. For certain materials it may be necessary to add a fluorescent dye or to slightly load the specimen so that all cracks are visible.

Optical microscopy is quite simple and relatively cheap to conduct and gives a very good indication of damage type and scale. However physical sectioning is obviously highly destructive and there is the danger that additional damage may be caused by the sectioning process itself. In order to obtain a three-dimensional map of damage several sections must be taken.

A Wild M5 microscope with 5, 12 and 25 times magnification was used for the optical

microscopy conducted for this thesis. Images were captured on an analogue CCD camera and digitised by a VFG-512 frame grabber card on an IBM PC. The resulting image was of 512 x 512, 8 bit (0-255) grey scale pixels. Figures 2.14 to 2.16 demonstrate the results obtained with computerised optical microscopy on coupons A, B and C.

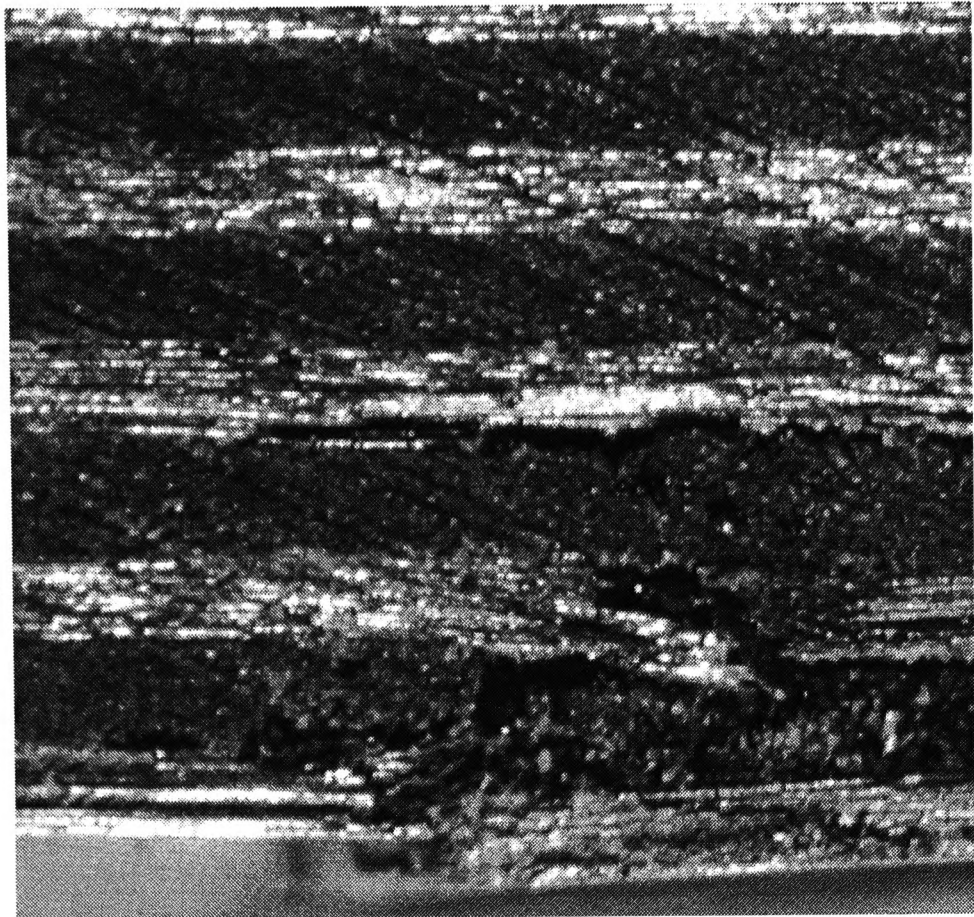
Figures 2.14 and 2.15 show the fibre fracture that has occurred at this low impact energy in the first two plies of the back face of coupon A. The plies that are lighter in colour in the image are 0° plies where the fibre orientation is in the plane of the image. The darker coloured plies are 90° plies where the fibre orientation is perpendicular to the image. The start of a delamination crack initiated from a transverse matrix crack can be seen between the fourth and fifth plies. However the wide distribution of delaminations throughout the laminate depth that can be seen in the X-ray CT image (Figure 2.11) and the C-scan (Figure 2.6) are not visible in Figures 2.14 and 2.15.



Coupon A, Impact Energy 5.11 J, Delamination Area 295 mm², Indent 0.05 mm

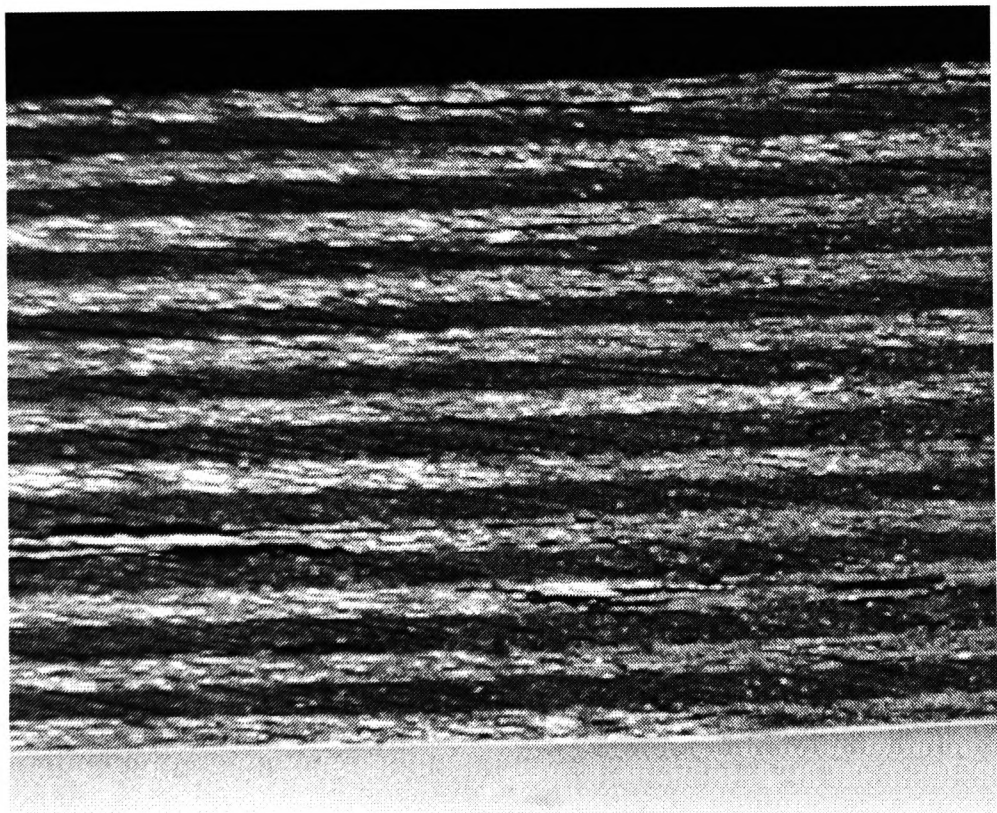
(Scale 34:1)

Figure 2.14



Magnification of Coupon A (*Scale 75:1*)

Figure 2.15

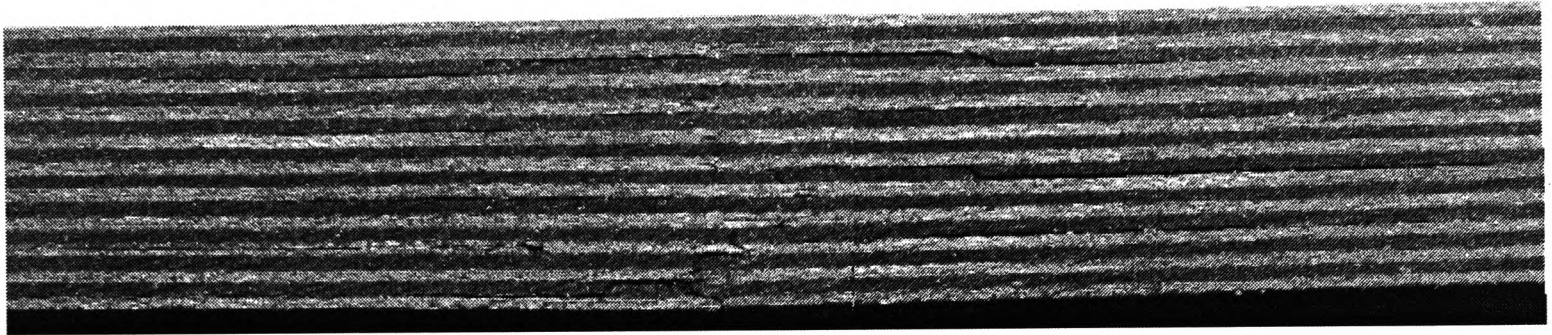


Coupon B, Impact Energy 7.35 J, Delamination Area 352 mm², Indent 0.06 mm
(*Scale 24:1*)

Figure 2.16

Optical microscopy of coupon B (Figure 2.16) does not reveal much of the damage which can be seen in the X-ray CT image (Figure 2.12). There are some delamination cracks visible but it is clear from the C-scan (Figure 2.7) that we cannot see their full extent in this image.

Figure 2.17 is a composite of five microscope images of coupon C. Here the delamination cracks are more distinct. This is because the greater impact energy has left a larger permanent indentation which leaves the delamination cracks slightly open. However delamination and matrix cracks in the microscope image are still not as clear as they are in the X-ray CT (Figure 2.13).



Coupon C, Impact Energy 9.86 J, Delamination Area 568 mm², Indent 0.16 mm
(Scale 10:1)

Figure 2.17

2.6.8 Dye Penetrant Microscopy

Fluorescent dye penetrants are commonly used with metallic components for detecting cracks in welds. The dye penetrant is applied to the weld, excess is cleaned from the surface, and any cracks are then visible when the dye fluoresces under an ultraviolet light. Dye penetrants can be used with composites in the same way. However several attempts with the impact damaged coupons A, B and C revealed that surface damage was so widespread that the dye penetrant smeared the surface and did not reveal cracks any more accurately than the unenhanced optical microscopy.

2.6.9 Thermography

Thermography uses the differential transmission and dissipation of heat in a damaged composite component. Passive thermography relies on an external heating source and measurement of the transmitted heat. However, thermography is less sensitive than the various ultrasonic methods. Because of the poor thermal conductivity of carbon fibre composites the method is best suited to the detection of gross delaminations in large structures such as aircraft wings. Active thermography involves measurement of the heat dissipated in a fatigue loaded structure due to friction and fretting of fracture surfaces in internal damage areas.

2.6.10 Acoustic Emission

Acoustic emission involves detection of the stress waves given off in a composite failing under load. Some studies [2] have shown that it may be possible to detect matrix cracking, fibre breaks and debonding. However friction and rubbing of fracture surfaces may create noise which exceeds the actual fracture process. The technique is further limited by the fact that composite materials tend to attenuate and disperse the stress waves.

2.6.11 Eddy Currents

Eddy currents can be used to detect fibre damage in composites containing electrically conducting fibres. Flaws within the material interrupt the current field of a coil positioned above the component and therefore cause a change of impedance in the coil. The technique is only suitable for detecting fibre fracture however and cannot detect delamination so is best used in conjunction with C-scanning. The introduction of wires into the composite is itself undesirable because it results in a weakening of the structure.

2.6.12 Fibre Optics

Optical fibres may be introduced into a composite during the manufacturing stage and the integrity of the component may then be evaluated from the intensity of light passing through the fibres. When a fibre is damaged or fractured as a result of cracking or localized failure the intensity of transmitted light will drop. Care must be taken in the choice of optical fibre material and position within the laminate in order to achieve optimum detection of damage and to avoid introducing a stress-raiser which could itself initiate delamination.

2.6.13 Moiré Interferometry

Moiré interferometry is an optical technique which provides contour maps or fringe patterns representing the in-plane displacement components of surface deformation. A fine crossed line diffraction grating is applied to the surface of interest and the specimen loaded and viewed in an interferometer. The technique can be used to detect the development of matrix cracking and delamination in composite materials by applying a grating to the edge of a laminate. These forms of damage cause discontinuities in the fringe patterns.

Shadow Moiré is a different technique which can also be used in damage monitoring. Shadow Moiré produces fringes representing out-of-plane displacement but with a sensitivity one or two

orders of magnitude less than Moiré interferometry. The technique may be used to observe the growth of a surface delamination blister in a compressively loaded laminate.

2.6.14 Thermal De-plying

This destructive technique involves heating the laminate to a temperature above that at which the polymeric matrix degrades. Once at room temperature the individual plies may be separated using a razor blade and observed for damage. Fibre fracture can be easily identified and with brittle matrix carbon fibre composites the delamination fracture surface areas are visible.

The technique is however very time consuming and is only suitable for thermoset matrices.

2.6.15 Scanning Electron Microscopy

Scanning electron microscopy (SEM) is an excellent post-failure analysis technique and has been used extensively to determine fracture paths in long fibre composites. Failure mechanisms such as fibre fracture, delamination, shear yielding, fibre pull-out and matrix cracking can be identified.

SEM is employed when it is not possible to obtain sufficient magnification from optical microscopy. In the case of CFRP composites, this is necessary to investigate failure modes at the individual fibre scale. For whole plies optical microscopy gives adequate magnification.

2.6.16 Conclusions

For rapid assessment of impact damage severity on in-service CFRP a visual inspection together with measurement of indentation is the only available method. However the scale of internal damage can only be estimated by this approach.

Detecting and measuring internal delaminations in-service will require the use of hand-held ultrasonic techniques or, perhaps, transmission thermography (for example on aircraft wings). B-scanning is also possible in-situ and can be used to give an indication of delamination depth and extent.

In the laboratory there are many more investigative techniques available. For measurement of the overall extent of delamination, ultrasonic C-scanning is the cheapest and fastest technique.

The C-scan does not, however, reveal matrix cracks and so X-ray projection radiography may be employed. Radiography, though, requires the use of a radio-opaque penetrant and therefore will only reveal cracks that are open to the specimen surface.

The only method for complete three-dimensional assessment of impact damage is X-ray CT. This technology is not currently truly non-destructive because it is necessary to cut out the impact damaged zone for scanning in order to obtain good image resolution. However the quality of section images obtained can be superior to optical microscopy after physical sectioning. Delaminations and matrix cracks can be defined more clearly since the CT section image can be computationally averaged through a depth of virtual section and this increases the image resolution.

B-scanning is an alternative technique for producing a section image through a laminate and has the advantage that it is not limited by specimen size. However the images obtained are not comparable in quality with X-ray CT or optical microscopy. The B-scan will give a guide to the upper limits of delamination but cannot give accurate information on the full distribution of cracks.

Physical sectioning is of course a fully destructive technique. But it is relatively cheap, and with optical microscopy good cross-sectional images of an impact damaged laminate may be obtained. Closed cracks are hard to see without a dye penetrant, however.

If it is necessary to investigate fracture surfaces at the individual fibre scale then SEM must be used.

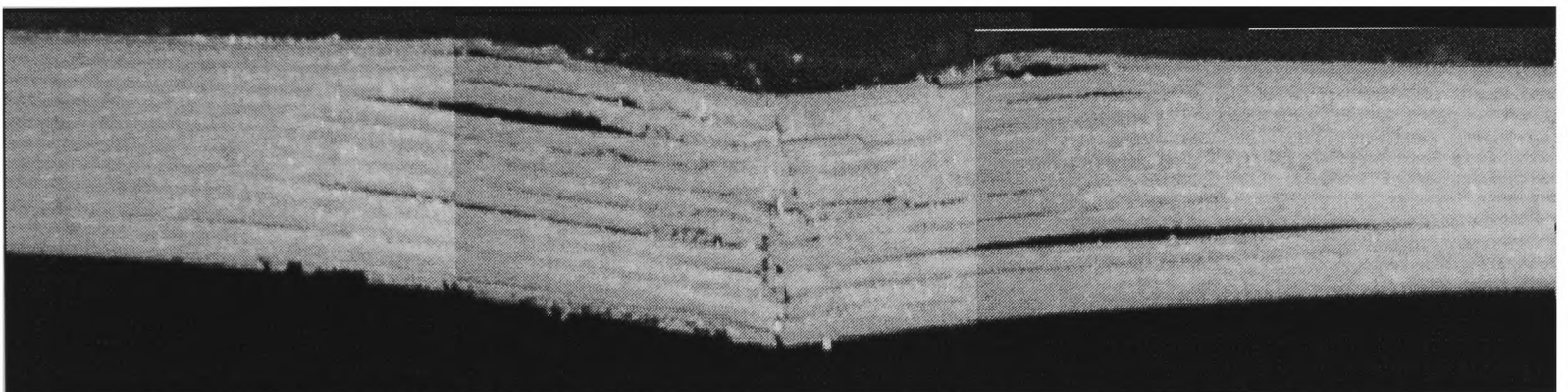
For characterising the impact damage in the tests conducted in this thesis the following techniques were adopted:

- Visual inspection
- Measurement of indentation
- Ultrasonic C-scan to determine extent of delamination
- X-ray computer tomography
- Optical microscopy

2.7 Low Speed Impact Damage

2.7.1 Description of Damage

Figure 2.18 shows a microscope image of typical low speed, hard body impact damage in a 21 ply coupon. Two distinct damage modes are apparent; delamination and fibre fracture. Delamination appears to have occurred at every ply interface. Fibre fracture occurs at the point of impact and has propagated throughout the whole laminate depth. It can be seen in Figures 2.14 and 2.15 that this fibre fracture is initiated on the back face (opposite face to impactor) and we can therefore deduce that the failure is due to tensile bending stresses. It would appear that the delaminations are shear stress failures in the matrix between plies.



Impact Energy 11 J (9 m/s), Delamination Area 816 mm², Indent 0.73 mm

Figure 2.18

Examination of X-ray CT images of damaged coupons (Figures 2.11, 2.12 and 2.13) indicates how the impact damage develops with increasing incident impact energy. Increasing impact energy results in larger and more frequent delaminations together with increasing depth of fibre fracture cracks initiated at the back face. The delamination cracks are larger towards the back face. It can also be seen that the largest delamination cracks are separated by approximately two plies and that there are smaller delaminations on the intermediate ply interface. The largest delaminations in a section are seen on the interface of 90° plies above 0° plies when the section has been cut perpendicular to the 90° plies.

2.7.2 Indentation

After impact testing the 21 ply coupons were measured for permanent indentation. Figure 2.19 shows the measured variation of permanent indentation with changing incident impact energy. There is significant scatter in these measurements.

The depth of permanent indentation shows a general increase with increasing incident impact energy. A best fit (least squares) linear trendline has been plotted through the data. This has been used to predict E_{bvid} :

21 ply coupon $E_{bvid} = 6.5 \text{ J}$

where barely visible impact damage (BVID) is an indentation of 0.3 mm as defined by the Airbus Industrie Test Method 1.0010 [8].

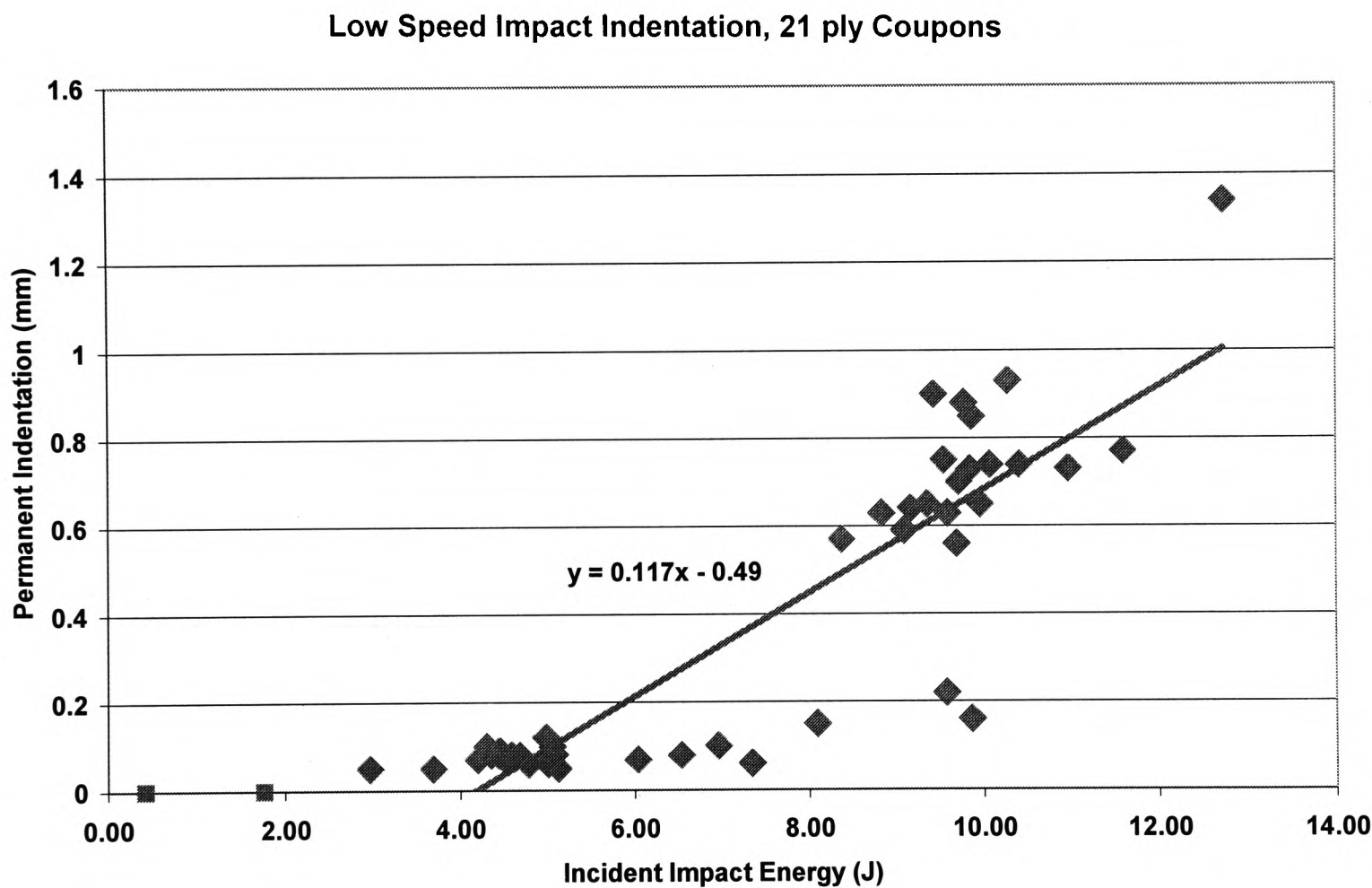


Figure 2.19

2.7.3 Delamination Area Measured by C-scan

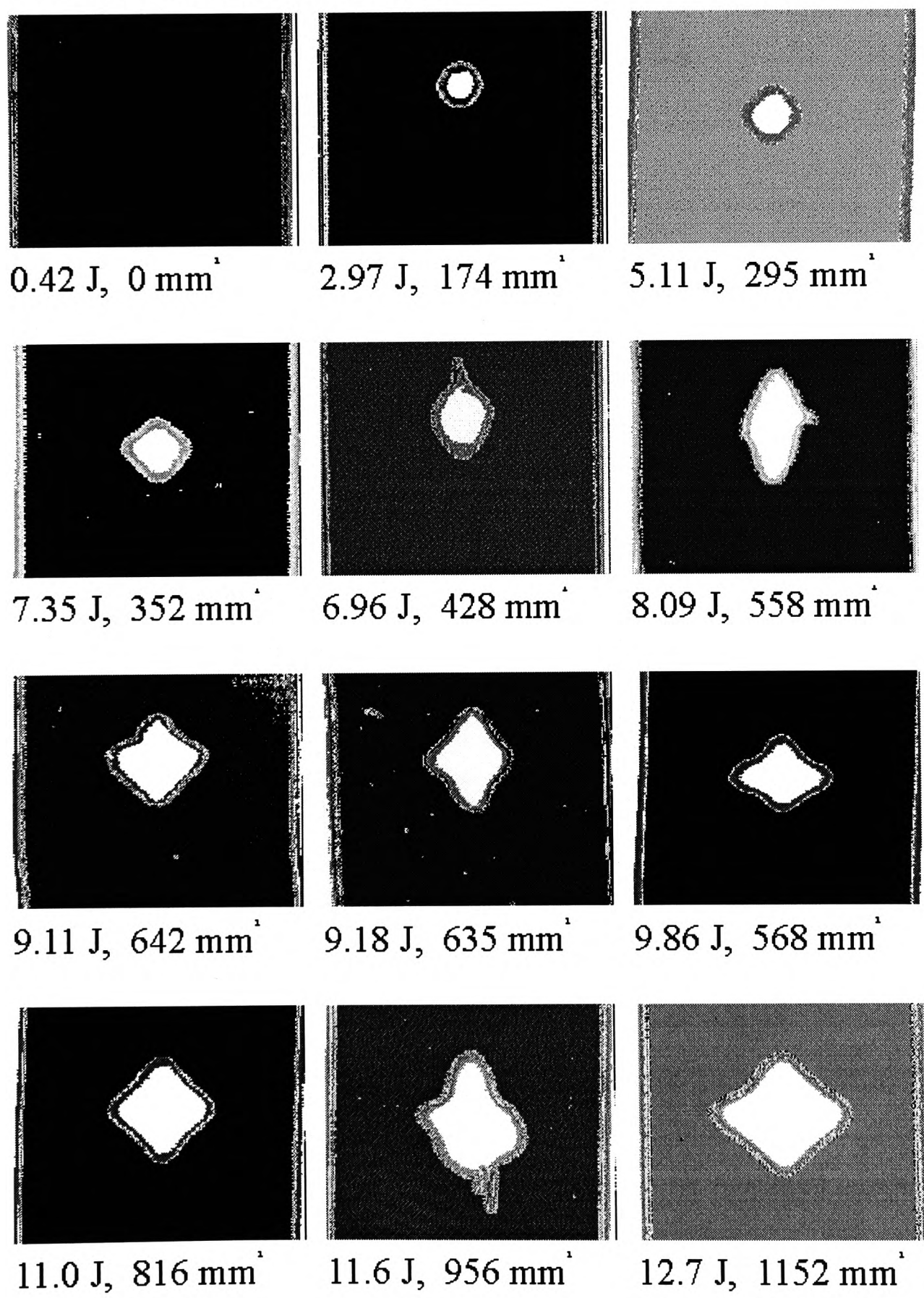
21 ply Coupons

Figure 2.20 shows typical C-scan images for the 21 ply coupons after low speed impact damage. They are presented in order of increasing incident impact energy. The C-scan image shows the overall extent of delamination in the coupon and does not indicate the shape of a delamination area between particular plies. The colour density of the undamaged area in each C-scan is unimportant and arises from slight differences in the magnitude of the reflected signal.

At very low impact energies the overall delamination area is roughly circular but for larger energies a diamond like shape is found. The shape of delamination is oriented with the 0°/90°

lay-up of the laminate plies.

For each coupon C-scanned the area of overall extent of delamination was calculated. In Figure 2.21 this delamination area has been plotted against the incident impact kinetic energy. It can be seen that there is an approximately linear relationship between delamination area and impact energy although the scatter is significant. There is a threshold impact energy for delamination damage to occur and this is approximately 2 J.



Ultrasonic C-Scan Images for 21 ply Coupons after Low Speed Impact Damage
Figure 2.20

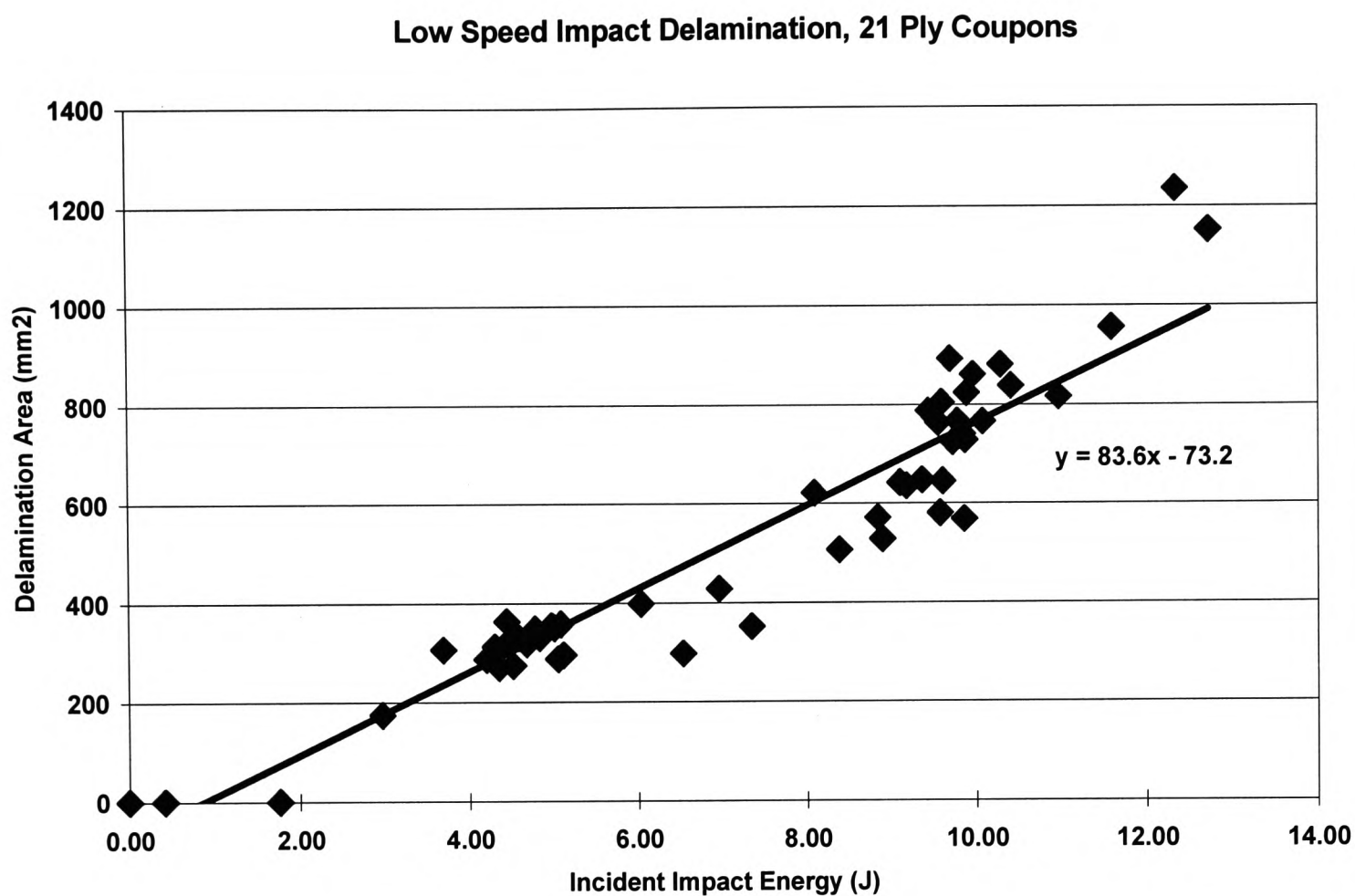


Figure 2.21

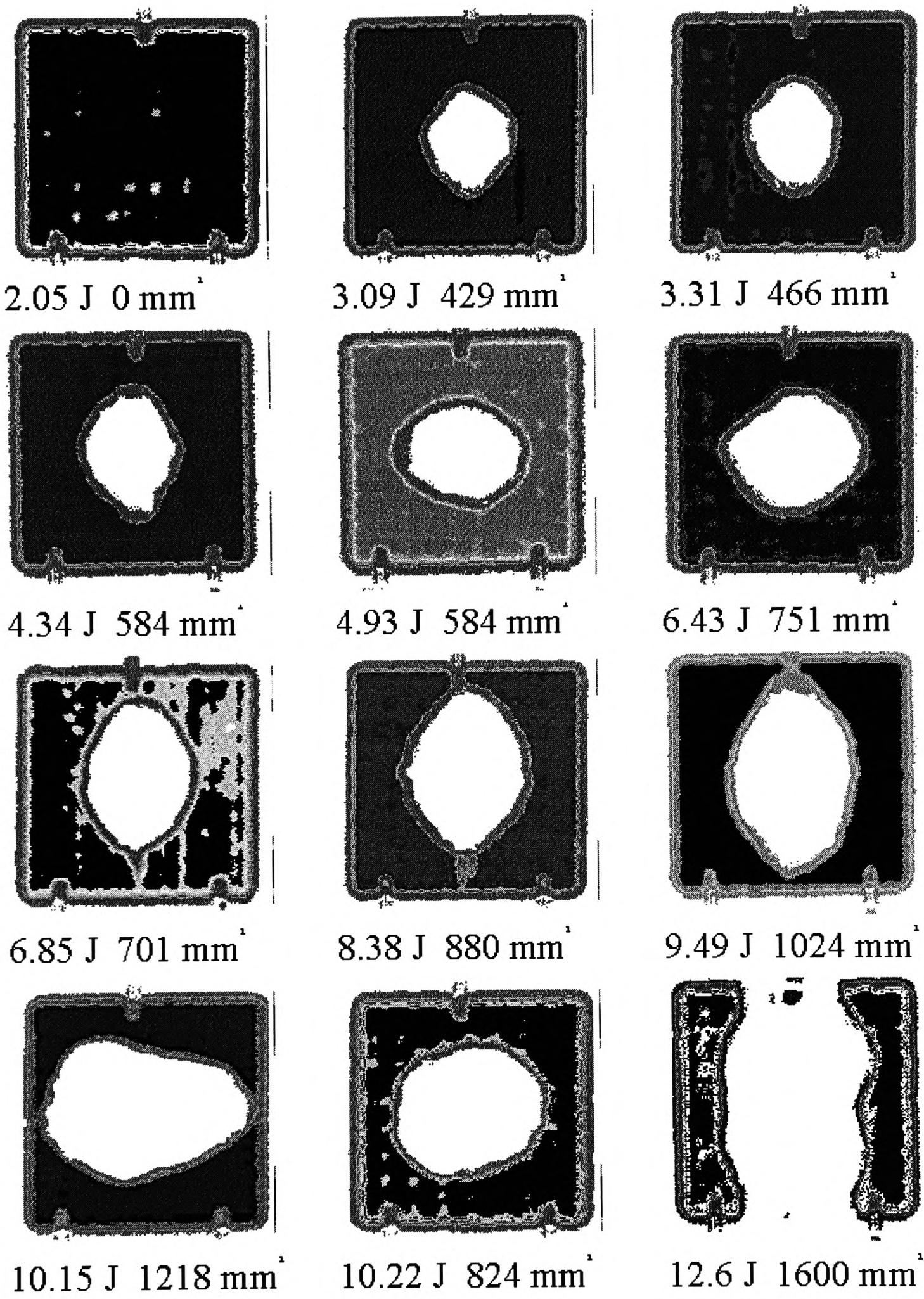
29 ply Coupons

Figure 2.22 shows typical C-scan images for impact damaged 29 ply coupons. The shape of the overall delamination area is generally elliptical for these coupons. Impact energies in excess of 12 J resulted in delamination damage spreading to the coupon edges.

Differences in colour density in the images arise from small variations in the magnitude of the reflected signal from coupon to coupon and are unimportant.

Figure 2.23 shows the distribution of impact delamination area with varying incident impact energy for the 29 ply coupons.

It can be seen that there is an impact energy threshold below which no delamination damage occurs. This is approximately 2.5 Joules. A least squares best fit trendline has been included on the graph.



Ultrasonic C-Scan Images for 29 ply Coupons after Low Speed Impact Damage

Figure 2.22

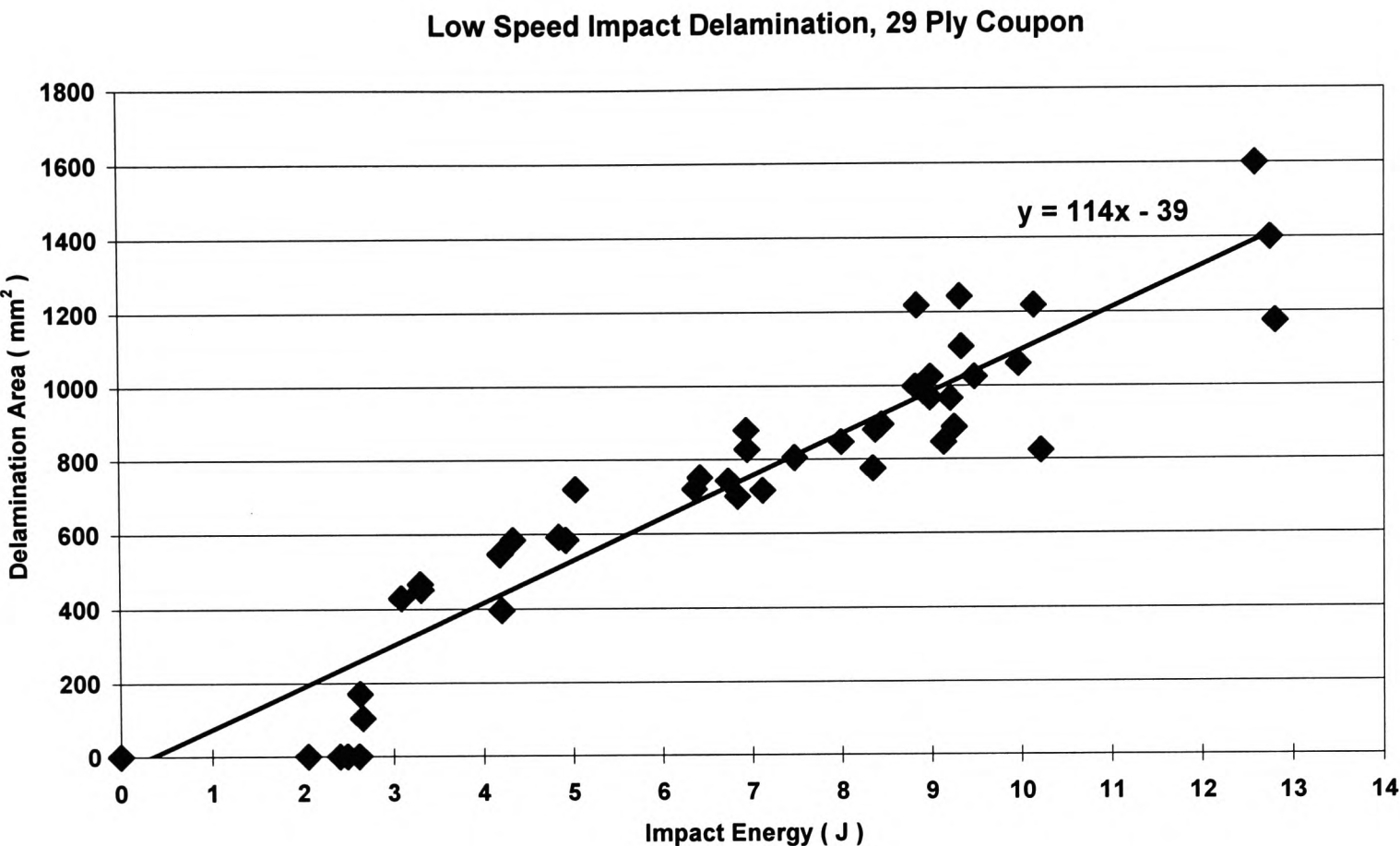


Figure 2.23

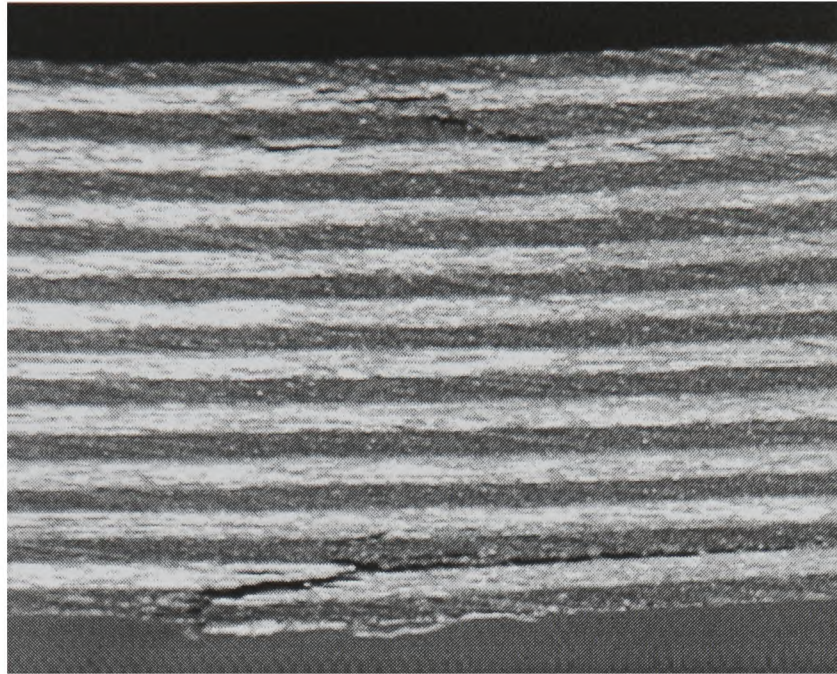
2.8 High Speed Impact Damage

2.8.1 21 ply Coupons

Figures 2.24, 2.26 and 2.27 show optical microscopy images of the impact damage resulting from high speed impact tests on the 21 ply coupons.

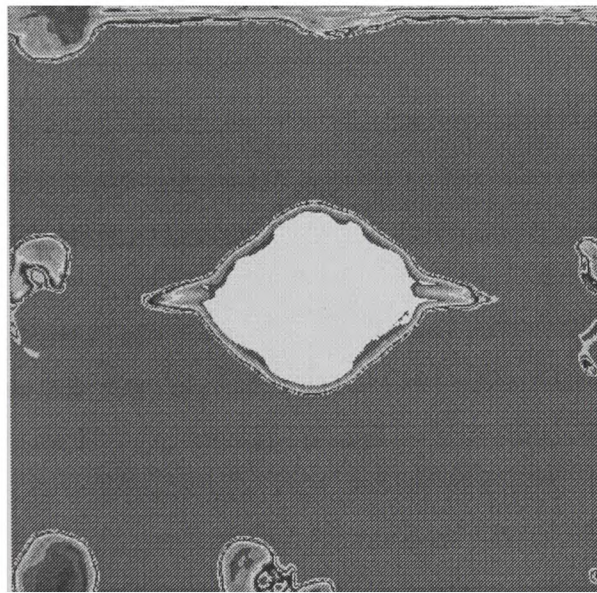
Figure 2.24 shows little visible damage for the low energy impact (3.3 J). There is some fibre fracture and matrix cracking at the back face due to bending stresses and also some cracking close to the top face just beneath the impact site. Although not visible optically there was substantial delamination in this coupon that can be seen in the C-scan image (Figure 2.25).

The delamination areas measured for the four high speed impact tests on 21 ply coupons are plotted against impact energy in Figure 2.28. The delamination areas for the low speed impacts are also plotted for comparison. It can be seen that, for non-penetrating impacts of a particular impact energy, impacts at speeds up to 100 m/s cause approximately the same area of overall delamination as higher momentum impacts at speeds under 11 m/s.



Optical Microscope Cross-section of High-Speed Impact Damage (*Scale 20:1*)
 21 ply Coupon, 6 mm $\varnothing$ Projectile (0.9 g), 86 m/s, 3.3 J, Indent 0.06 mm

Figure 2.24

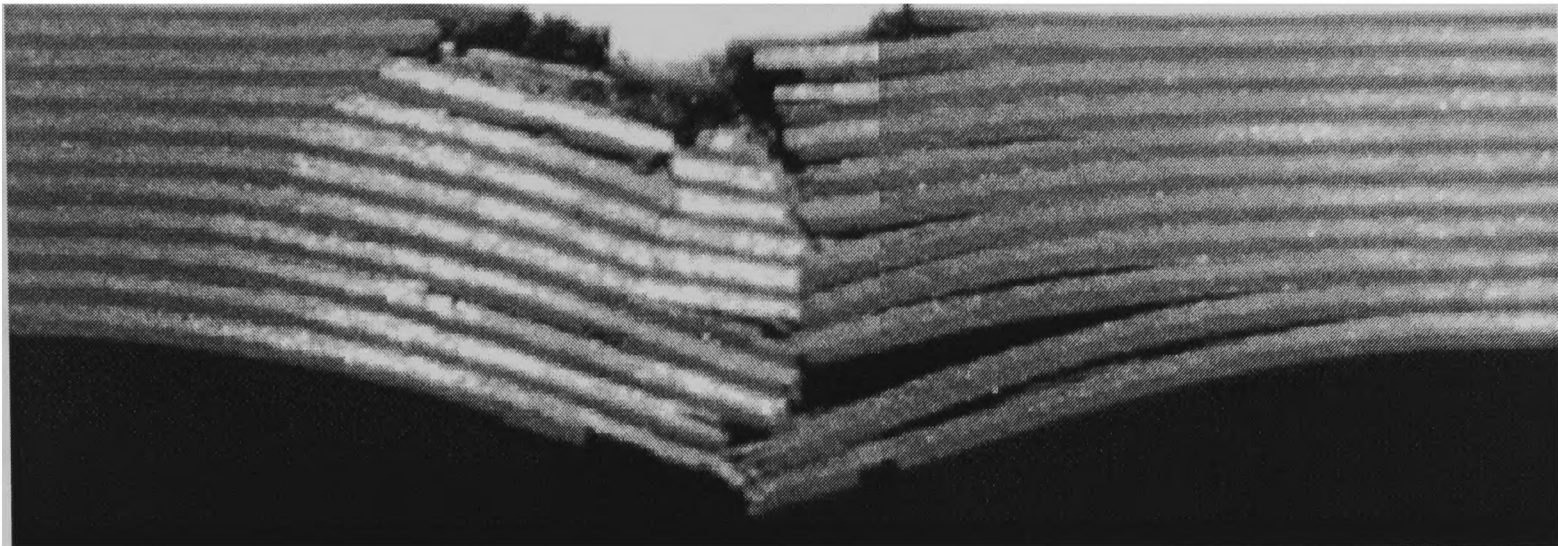


60 x 60 mm C-Scan Image, Delamination Area 284 mm² (*Scale 1:1*)
 21 ply Coupon, 6 mm $\varnothing$ Projectile (0.9 g), 86 m/s, 3.3 J, Indent 0.06 mm

Figure 2.25

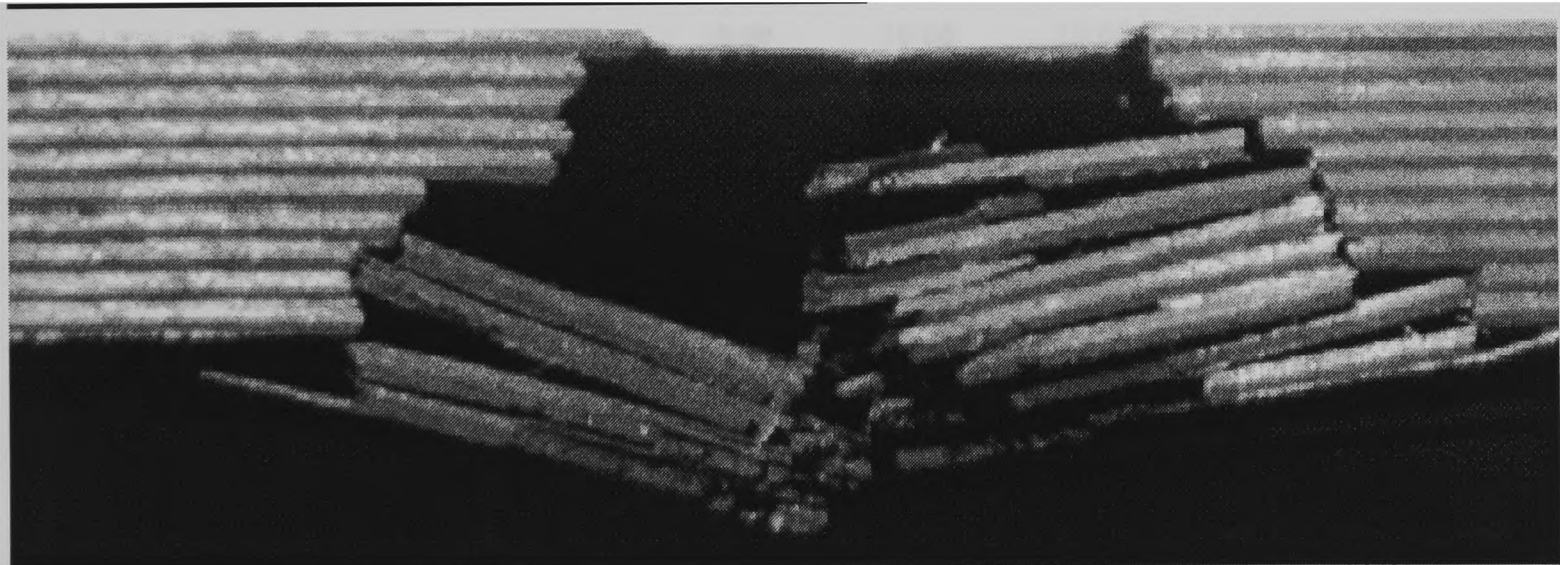
Figure 2.26 shows typical damage for a high speed (109 m/s) impact of 5 J. Although the amount of delamination is limited, the fibre fracture is much more severe than we observe with a low speed 11J impact at only 9 m/s (Figure 2.18).

The tendency of high speed impacts to cause localised damage is further demonstrated by Figure 2.27. This image shows the damage resulting from a 143 m/s impact. The projectile has penetrated the coupon although the incident kinetic energy was only 9 J. The failure mode is tending towards a shear cone and with even higher impact velocities we would expect a clean plug-out failure.



High Speed Impact Damage (*Scale 12:1*)
21 ply Coupon, 6 mm $\varnothing$ Projectile (0.9 g), 109 m/s, 5.4 J, Indent 0.55 mm

Figure 2.26



High Speed Impact Damage (*Scale 12:1*)
21 ply Coupon, 6 mm $\varnothing$ Projectile (0.9 g), 143 m/s, 9.2 J, Penetration

Figure 2.27

2.8.2 29 ply Coupon

Figure 2.29 shows the damage resulting from a 195 m/s impact on a 29 ply coupon. The projectile has directly penetrated the coupon surface and pushed out a shear cone of failed plies.

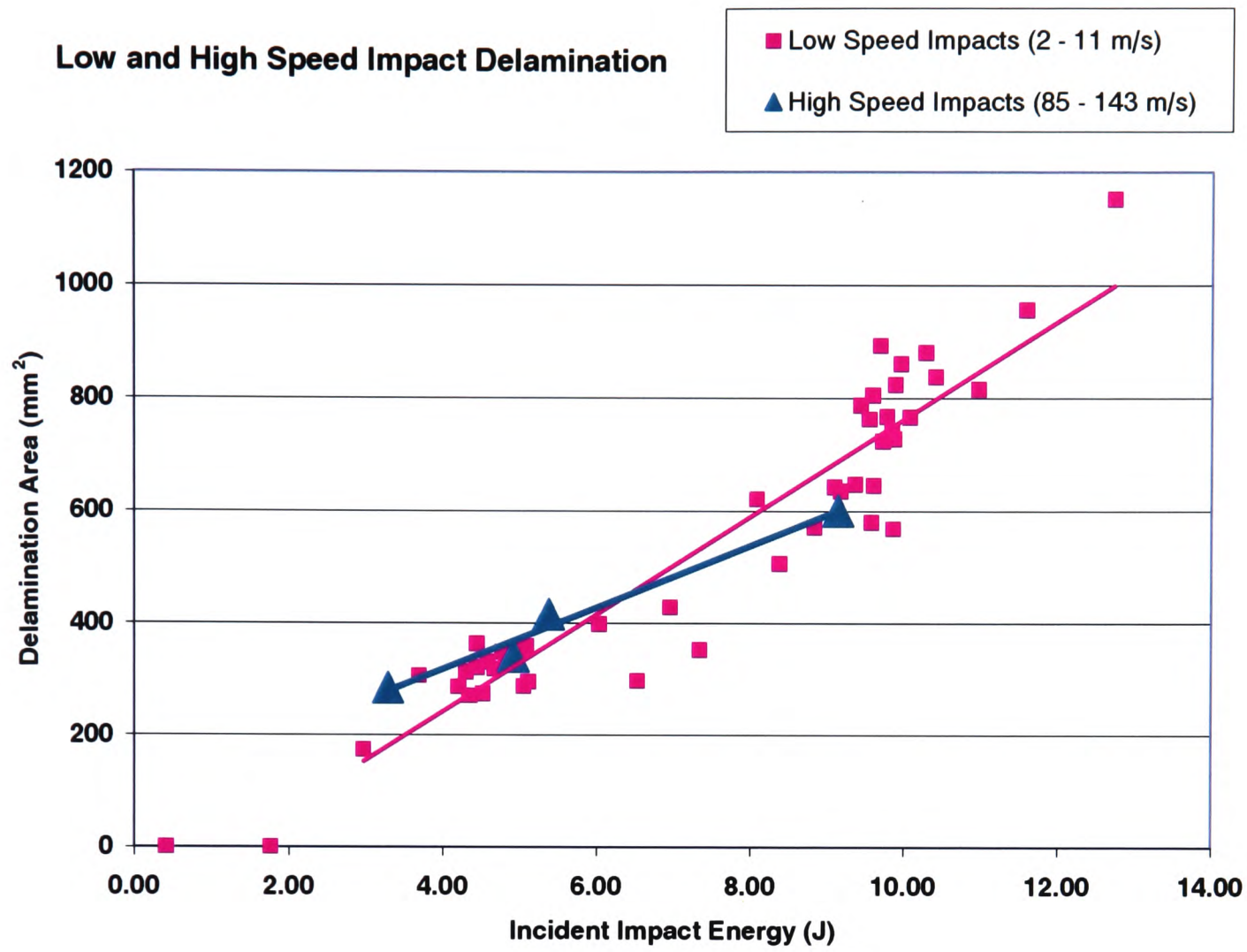
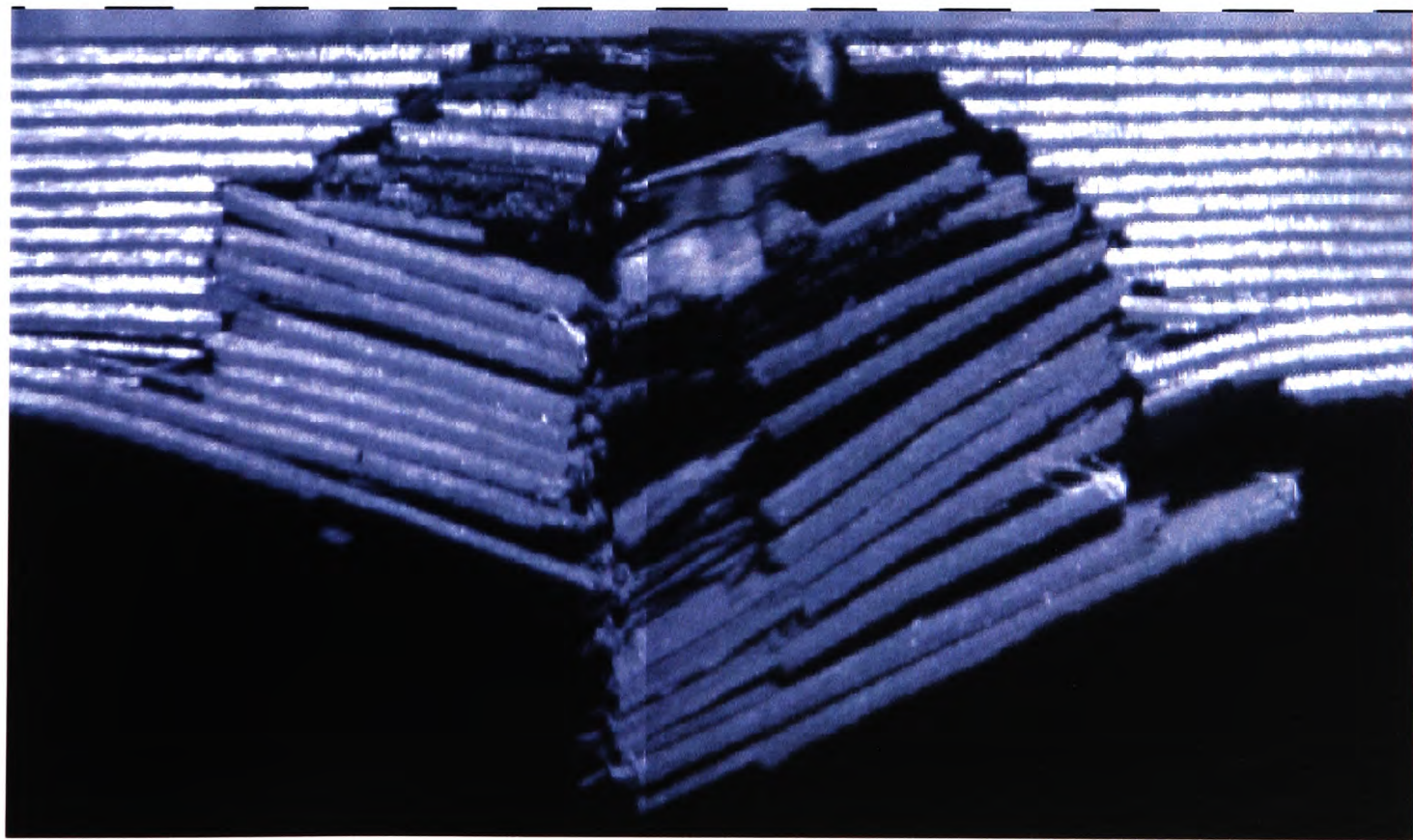


Figure 2.28



High Speed Impact Damage
29 ply Coupon, 6 mm ø Projectile (0.9 g), 195 m/s, 17 J

Figure 2.29

2.9 Quasi-static Indentation Damage

2.9.1 Continuous Indentation

Figure 2.30 shows the loading curves for continuous quasi-static indentation tests on both 29 and 21 ply coupons. The behaviour is elastic up to the point of first failure.

Behaviour after first failure is similar for both coupons. There is a drop in load but then a steady increase with increasing damage up to a second, higher, maximum load. The loss in strength after the second major failure is more extreme for the 21 ply coupon. Continued indentation shows that the severely damaged coupons retain some resistive strength but complete penetration would be expected to follow soon if the tests were continued further.

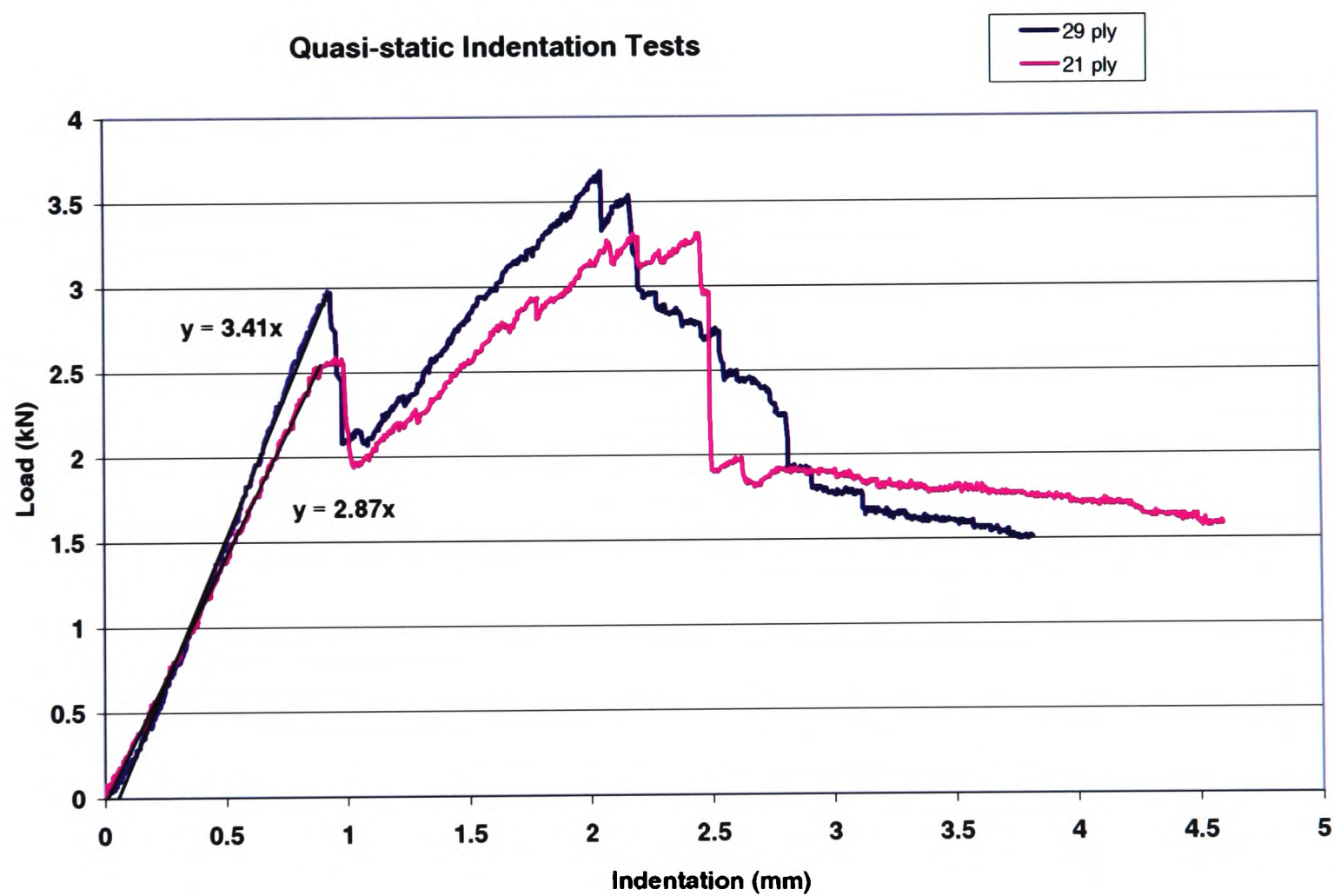


Figure 2.30

Measured elastic indentation stiffness, first failure load and maximum load values from Figure 2.30 are given in Table 2.5. The elastic strain energy absorbed by the coupon at the point of first failure has also been calculated. This is the threshold energy for permanent damage U_{sf} .

Indentation Stiffnesses and Strengths

Coupon	29 plies, 50 x 50 mm	21 plies, 140 x 85 mm
Indentation Stiffness k_{indent}	3.41 kN/mm	2.87 kN/mm
First Failure Indentation Load P_{thres}	2.98 kN	2.51 kN
First Failure Indentation	0.925 mm	0.894 mm
First Failure Energy U_{sf}	1.46 J	1.15 J
Maximum Indentation Load P_{max}	3.68 kN	3.31 kN

Table 2.5

2.9.2 Repeated Indentation, 29 ply Coupon

Figure 2.31 shows loading and unloading curves for a 29 ply coupon. The test involved indenting to a particular depth and then reversing the machine to determine the unloading curve from each point. Upon reloading, the loading curve returns to the continuous indentation envelope.

Since this is a force/displacement graph the area under the loading curve represents the elastic and inelastic work done on the coupon. The area under the unloading curve is the energy regained after deformation.

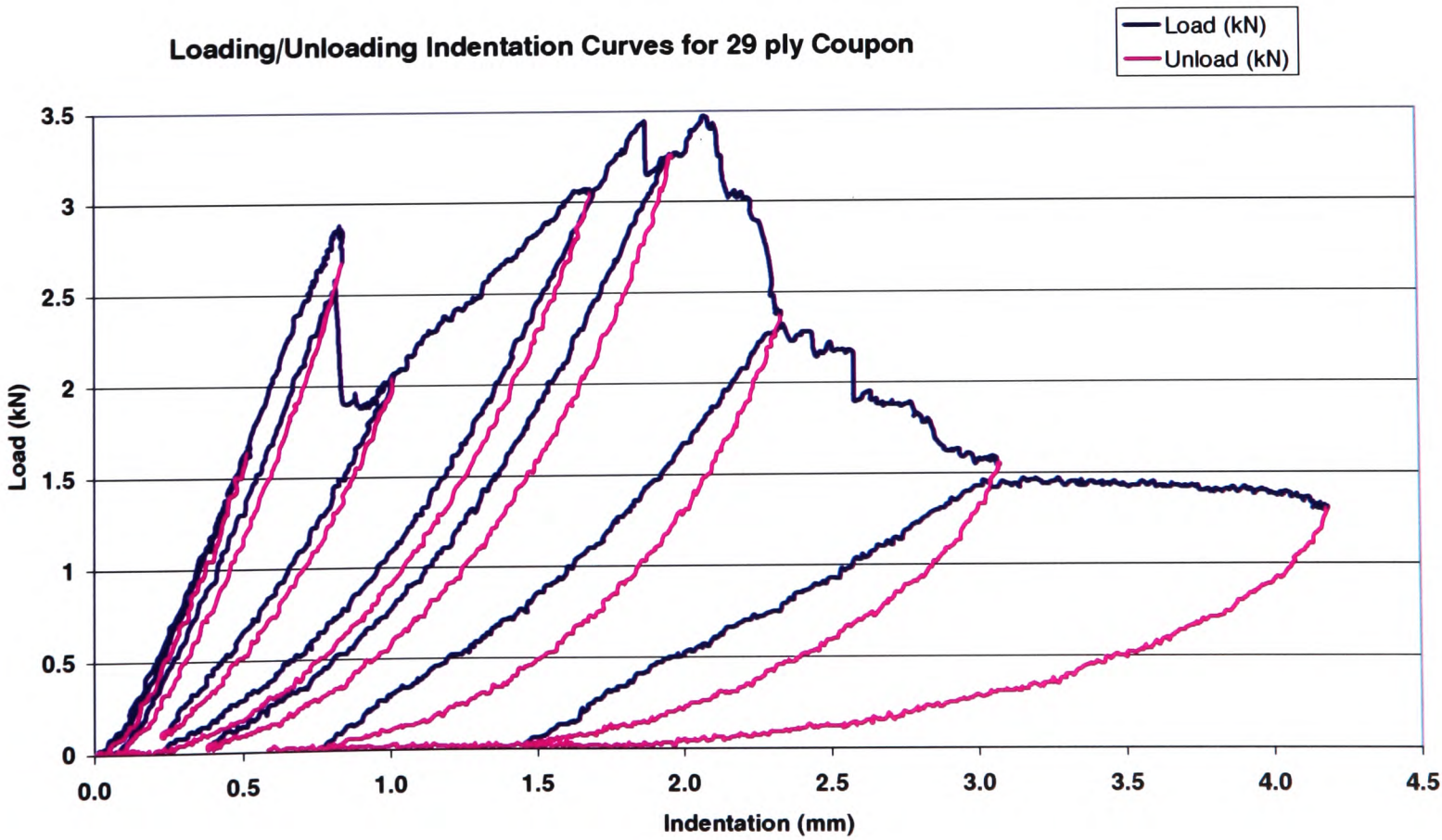


Figure 2.31

It can be seen that there is significant hysteresis between loading and unloading paths. The area between loading and unloading paths represent the absorbed energy that has been dissipated within the coupon.

The tests indicate that the energy absorption within the coupon can be considered to have three components:

- i) Fracture energy (fibre fracture, transverse matrix cracking and delamination)
- ii) Frictional work (between delaminated plies, debonded fibres etc.)
- iii) Inelastic matrix behaviour

The main component is the energy of the damage fracture modes. Fracture gives the biggest drops in load in continuous indentation and consequently the largest proportion of dissipated energy.

However, there is also hysteresis between the reloading and unloading paths where there is no increase in damage. This reloading hysteresis most probably arises from frictional effects within the damaged CFRP laminate. It seems likely that the pushing and pulling out of broken fibres in the matrix and the sliding of delaminated areas over each other may cause this behaviour.

The final, and presumed smallest, component of dissipated energy arises from the inelastic behaviour of the matrix. Within the nominally elastic region of loading very slight hysteresis is observed and although no fibre fracture can be detected a very slight indentation can be observed in the coupon epoxy surface.

Some researchers have measured force and displacement in coupon impact tests and seen the same kind of energy absorption measured here in quasi-static tests.

Yuan and Karger-Kocsis [14] observed energy absorbed in this way in impact indentation tests on CFRP coupons. Their tests used a drop weight at speeds of up to 10 m/s impacting 0/90 UD CFRP lay-up coupons. The impact tip had a radius of 10 mm and the coupon support was a clamped edge 40 mm diameter window.

Delfosse and Poursartip [15] measured the force and displacement of a rebounding impactor

in a CFRP coupon impact test. They measured the energy absorbed in drop weight tests on quasi-isotropic lay-up CFRP coupons. Impact conditions were according to the Boeing standard. It was found that in quasi-static tests and drop weight impacts of up to 4 m/s the coupon stiffness and energy absorption were virtually identical.

2.9.3 Repeated Indentation, 21 ply Coupon

Figure 2.32 shows loading and unloading curves for an indentation test on a 21 ply coupon. The same hysteresis behaviour is seen here. In this test after each loading and unloading cycle the coupon was visually inspected for damage before the permanent indentation was measured and the overall delamination area was assessed by ultrasonic C-scan.

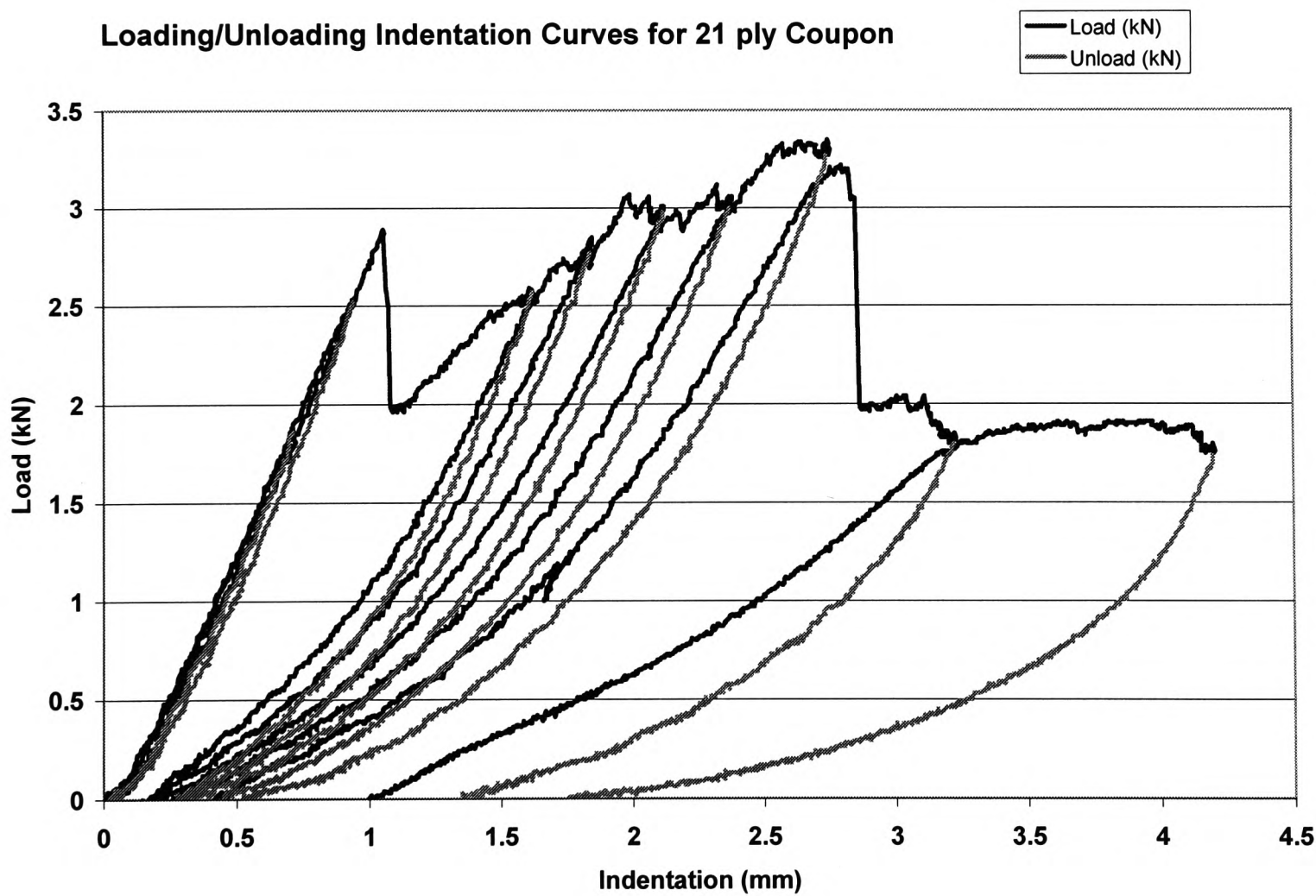


Figure 2.32

The measurements taken and observations made after each load/unload interval are detailed in Table 2.6. *Indentation* represents the limit of indenter displacement in the corresponding load interval; *permanent indentation* was measured after unloading in that interval. *Work done* was calculated as the work done by the indenter on the coupon up to the limit of indentation. It has been taken as the integral of load with respect to indentation displacement. *Delamination area*

was measured by C-scan. The progression of C-scans taken during the test are shown in Figure 2.33.

Although no fibre damage occurred in the first two load/unload intervals there was some permanent matrix deformation. In the first interval this was indicated by a permanent indentation of 2.5 mm diameter, less than 0.01 mm deep, that was just observable under close visual inspection. The second load interval was also in the nominally “elastic” range and again there was no detected fibre breakage or delamination. However there was further permanent matrix deformation. This included a very slight bulge visible on the back face in addition to the increased front face permanent deformation. The energy absorbed in this matrix deformation can be inferred from the slight hysteresis of the first two “elastic” intervals in Figure 2.32.

The third load interval caused fibre fracture and delamination. Cracks visible on the front face were two compression cracks, approximately 4 mm long, on either side of the permanent indentation and perpendicular to the 0° fibres. On the back face a 20 mm longitudinal split parallel to the 0° fibres was visible. Delamination was observed and measured by C-scan.

Further load intervals increased the permanent deformation, the delamination area and the length of the back face longitudinal split. Fracture processes were audible during each loading interval beyond the previous maximum indentation.

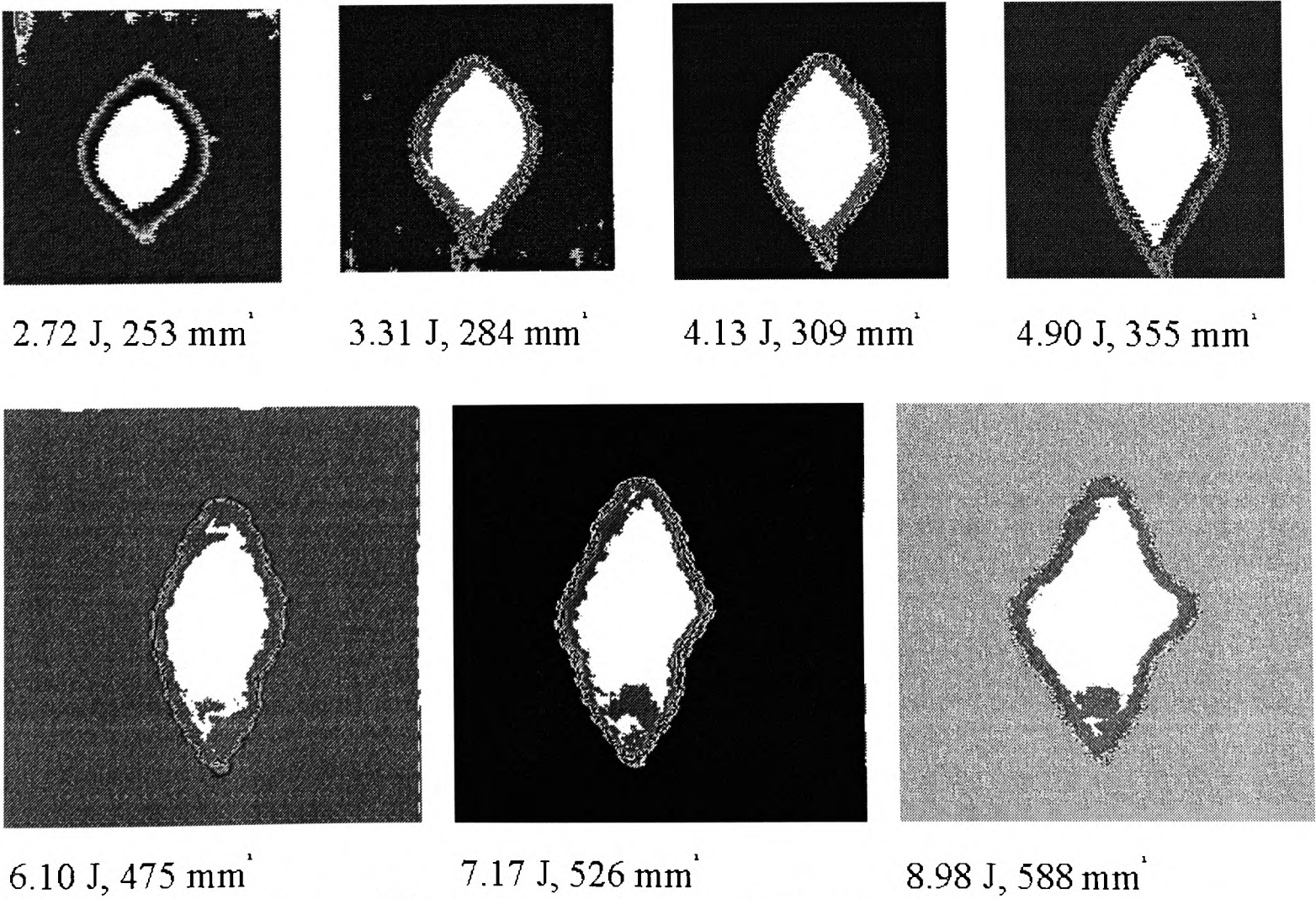
The area of delamination at each load interval is compared with the delamination area resulting from low speed impact tests in Figure 2.34. For comparison we assume that during low speed impact the incident kinetic energy of the impactor is entirely dissipated as work done on the coupon. Some of this work done on the coupon may be recovered as stored elastic energy and this allows the impactor to rebound.

If this assumption holds true then it can be seen that the area of delamination caused by impacts of up to 11 m/s is similar to that caused by quasi-static “impacts” of the same energy.

We also compare the permanent indentation resulting from the quasi-static tests with both low speed and high speed impact tests in Figure 2.35. It can be seen that both the quasi-static tests and high speed tests resulted in greater permanent indentation than the low speed impacts.

Load/ Unload Interval	Indentation (mm)	Work Done (J)	Permanent Indentation (mm)	Delam. Area (mm ²)	Comments
1	0.706	0.597	0	0	Elastic (no measurable damage), visible surface mark
2	0.945	1.116	0.02	0	Elastic but measurable matrix indentation, slight deformation visible on back face
3	1.626	2.721	0.12	253	Compression fibre breakage cracks (front face), delamination, transverse split (back face)
4	1.844	3.305	0.12	284	Increased delamination
5	2.125	4.131	0.14	309	
6	2.379	4.892	0.17	355	
7	2.753	6.1	0.32	475	
8	3.231	7.165	0.97	526	Major back face damage
9	4.203	8.977	1.44	588	

Table 2.6



C-Scans of Quasi-Static Indentation Intervals, 21 ply Coupon

Figure 2.33

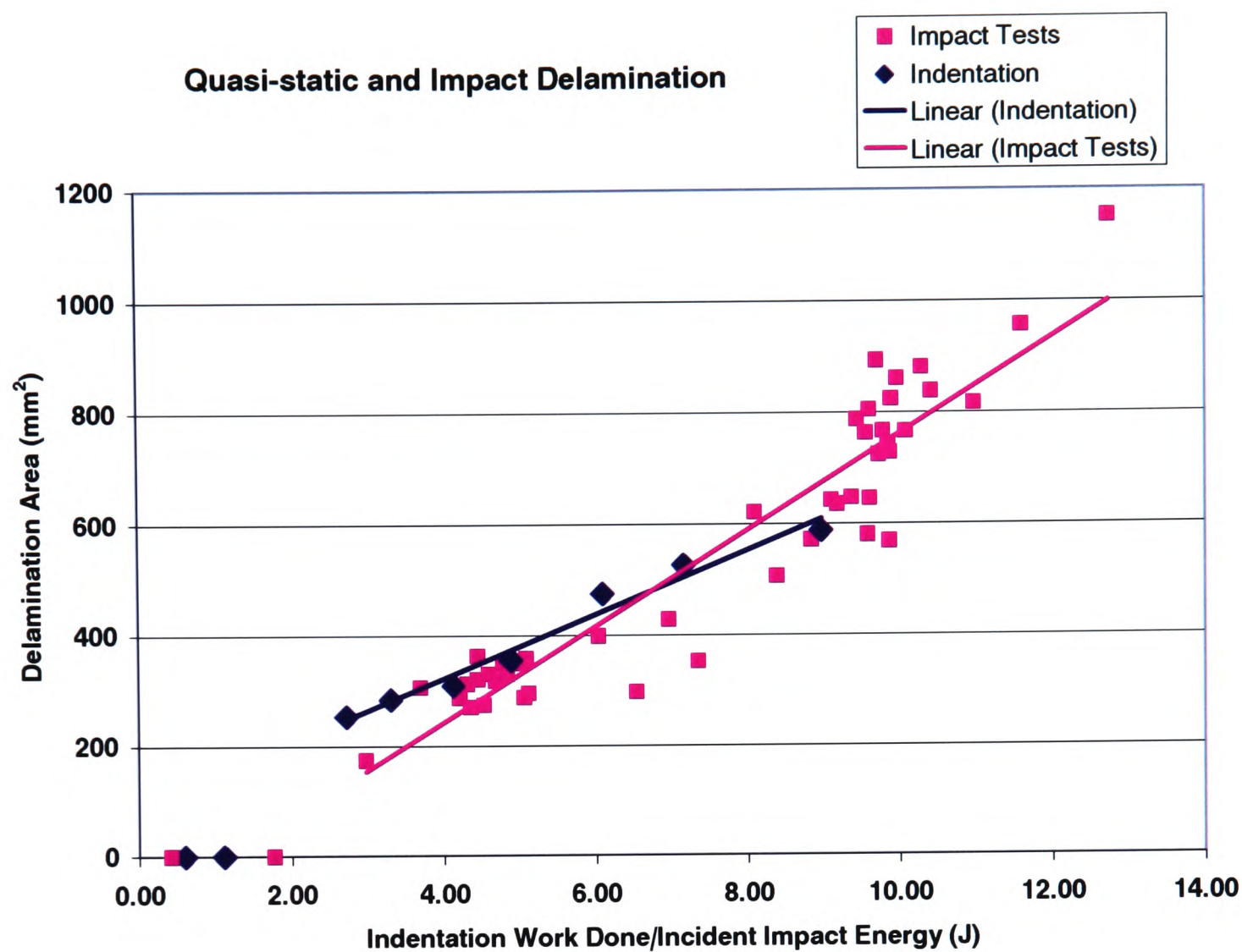


Figure 2.34

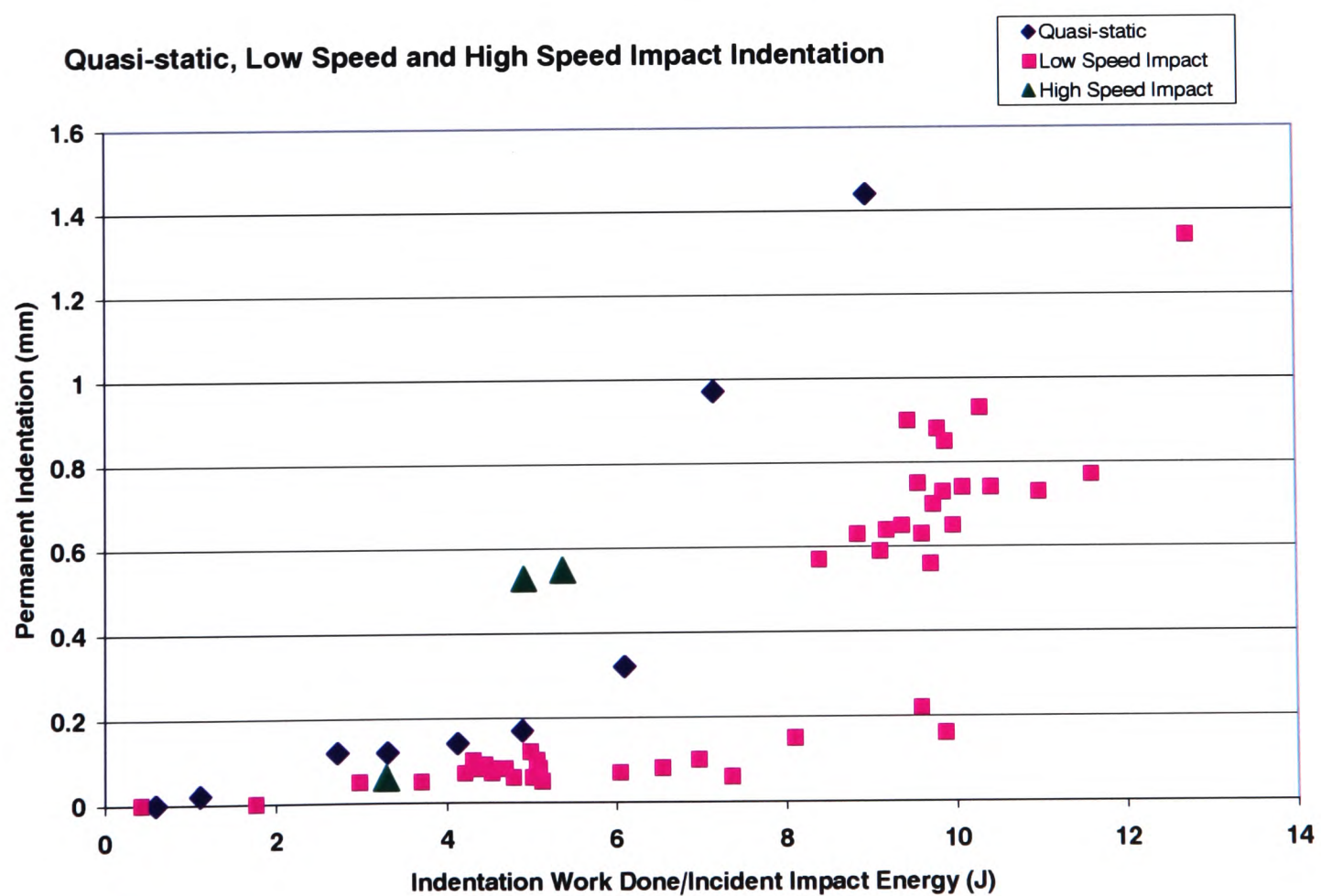


Figure 2.35

2.10 Conclusions

In this chapter it has been shown that, after visual inspection, indentation measurement is the quickest method of measuring the severity of hard body impact damage on a component in-situ. For characterising the exact distribution of delamination and fibre cracks X-ray computer microtomography is the most accurate method. However a combination of ultrasonic C-scan and optical microscopy is a more flexible approach for rapid assessment.

Low speed impact damage produces a distribution of delamination cracks throughout the thickness of a $0^\circ/90^\circ$ lay-up laminate. There will also be fibre fracture which initiates on the back face of the coupon. Increasing incident impact energy results in increasing permanent indentation and increasing delamination area. There is significant scatter with both measurements.

High speed impact damage is more localised, and tends, at higher energies, towards shear out failure of the laminate. Although the delamination areas for high and low speed impact are similar for the same incident impact energy, the permanent indentation is greater for high speed impacts.

Quasi-static indentation damage was very similar to low-speed impact damage. Delamination area and fibre fracture patterns were comparable. However the permanent indentation resulting from a given indentation work done was greater than that measured for the same incident low speed impact energy.

Absorbed energy was measured on repeated loading and unloading quasi-static indentation tests. The absorbed energy was attributed to three modes: fracture energy, frictional work and inelastic matrix behaviour.

Chapter 3

Review of Composite Modelling Techniques

Testing of composite components is expensive and time consuming and modelling techniques which reduce the requirement for testing can be useful tools in reducing the cost and time involved in design.

Predicting the structural behaviour of a composite laminate is relatively straightforward and can be accomplished either approximately, by assuming homogeneous material behaviour, or, more accurately, by the use of “classical” laminate mechanics. It is therefore possible to determine impact behaviour up to the point of failure by Newtonian mechanics.

Predicting the point of failure, and the progression of damage thereafter, however, poses a problem. The difficulty lies in the modelling of the brittle fracture process.

Various approaches have been taken. It is possible to avoid the complexity of accurately modelling structural behaviour, failure and damage development by using approximate methods. However, accurate prediction requires more complex techniques.

A number of stress based failure criteria have been proposed for predicting the initiation of damage. However, for modelling the behaviour throughout the impact process, before and after damage initiation, some authors have considered energy to be a more useful measure than stress. Fracture mechanics techniques, which consider the energy dissipated by propagating cracks, have been applied to composites, but have been limited by the difficulty of modelling a multitude of cracks with different failure modes. Compromise techniques, which use accurate modelling of structural behaviour by finite element analysis combined with incorporated results from coupon tests, have also been proposed.

Because of the difficulty of applying fracture mechanics to the complex failure modes in composites, the most widely adopted modelling techniques to date are purely stress based. Finite element analysis is used with various stress based failure criteria and reduced strengths are introduced as failure is detected.

3.1 Approximate Methods

Approximate methods may be used as a precursor to more accurate analysis or testing.

Sierakowski and Newaz [16] reviewed a number of analytical models for the impactor / laminate impact event. Most of the models reviewed consider the impact event as a three step process: the contact between impactor and target causes pressure which is time dependent; the pressure gives rise to stresses in the laminate; the stresses cause failure. In reality this process is actually continuous. Failures in the laminate will cause a change in mechanical properties which will affect the contact pressures; these in turn affect the stresses, and the process will continue until the contact has ended and motion has ceased. However the models considered by Sierakowski and Newaz have generally been concerned only with determining the maximum stresses during an elastic impact event; these are then applied to a failure criterion to determine whether failure has occurred. None of the models consider a continuous process of gradual failure throughout the impact event.

To determine the contact pressures between impactor and laminate Sierakowski and Newaz considered two separate deformation mechanisms: local Hertzian contact and structural deformation of the laminate. The relative magnitudes of these mechanisms in the coupon tests conducted for this thesis are examined here.

3.1.1 Hertzian Contact Stiffness

Many authors have considered local Hertzian contact stiffness to be significant in predicting the force/time history of the impact event. The total deflection may be considered to be a combination of the deflection due to flexural bending and the local Hertzian deflection.

$$\delta = \delta_{bend} + \delta_{hertz} \quad 3.1$$

If it is assumed that the local contact deformation is dominated by matrix, rather than fibre deformation, we may assume that the effective Hertzian contact modulus $E^* \approx E_2 \approx 9.4$ GPa. The elastic indentation δ_{hertz} by the (rigid) indenter is therefore:

$$\delta_{hertz} = \frac{1}{2} \left(\frac{9P^2}{2E^{*2}R} \right)^{\frac{1}{3}} \quad 3.2$$

where R is the indenter tip radius.

3.1.2 Plate Bending (Isotropic Assumption)

To approximate the structural deformation of the plate we may assume homogeneous bending behaviour. According to simple bending theory (Timoshenko & Woinowsky - Krieger [17]) the central deflection δ_{bend} of a circular, simply supported plate under a central point load P is:

$$\delta_{bend} = \frac{3(3+\nu)(1-\nu)Pa^2}{4\pi Eh^3} \quad 3.3$$

where a and h are the plate radius and thickness, and E is the isotropic bending modulus. We take E as an average of the unidirectional longitudinal modulus E_1 and the transverse modulus E_2 . Therefore $E = 74.2$ GPa..

The relative magnitudes of δ_{hertz} and δ_{bend} in Figure 3.1 are compared with the measured indentation stiffness for a 21 ply coupon. It can be seen that for the support configuration used, the Hertzian contact deflection is almost a third of the bending deflection and cannot be neglected. This simple approximation gives a reasonable first estimate of the plate/impactor stiffness.

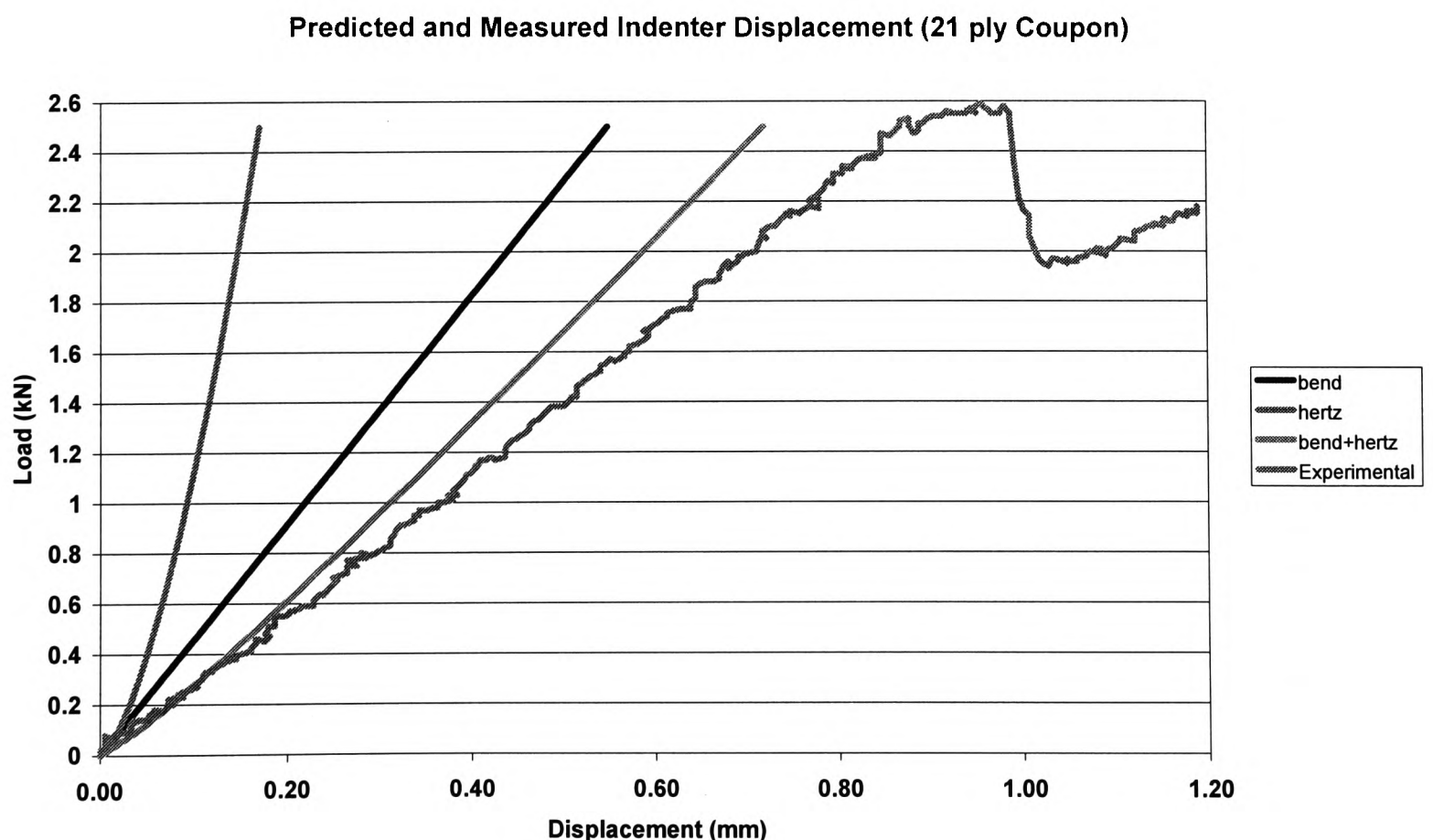


Figure 3.1

If we take the assumption of homogeneous, isotropic, behaviour further, we may approximate the first failure load for indentation by assuming it is governed by tensile failure. Woinowsky-

Kreiger [17] proposes the following approximate formula for the maximum bending tensile stress in an isotropic, simply supported plate under a very strong load concentration:

$$\sigma_{\max} = \frac{P}{h^2} \left[(1 + \nu) \left(0.485 \log \frac{a}{h} + 0.52 \right) + 0.48 \right] \quad 3.4$$

Assuming homogeneous behaviour we take the maximum tensile stress as the average of the longitudinal and transverse composite strengths T_1 and T_2 and let $\sigma_{\max} = 887$ MPa. Therefore the predicted first failure load P_{thres} for a 21 ply coupon is :

$$P_{thres} = 4.77 \text{ kN (approximate prediction)}$$

Since this prediction differs significantly from the measured indentation failure load $P_{thres} = 2.51$ kN it is clear that more accurate methods of predicting failure are necessary.

3.1.3 Delamination Prediction

Clark [18] proposed a model for delamination arising from impact damage which does not quantify the size, but does explain the orientation of the delaminations between particular plies. He considered the deflection under transverse impact of two adjacent, non-parallel plies (Figure 3.2). At points A there are peeling (tensile) forces acting between the upper and lower ply adjacent to the impact site. At points B the deflection of the fibres in the upper ply provides interlaminar compression. Consequently the delamination between two plies is elliptical in shape with the major axis of the ellipse lying along the fibre direction of the lower of the two plies.

Clark's model is in agreement with the distribution of delamination cracks observed in low speed impact testing. Figure 2.18 is a microscope image of a sectioned, impact damaged coupon. The section is perpendicular to the 90° plies. There are delamination cracks visible on every ply interface throughout the thickness. The widest cracks in the section, however, are those on the interface of a 90° ply above a 0° ply which is in accordance with Clark's model.

C-scan images of overall delamination area in Figure 2.20 are also consistent with Clark's model. The concave diamond shape of overall delamination is less surprising if it is appreciated that it consists of several elliptical delaminations at different ply interfaces, all having perpendicular major axes.

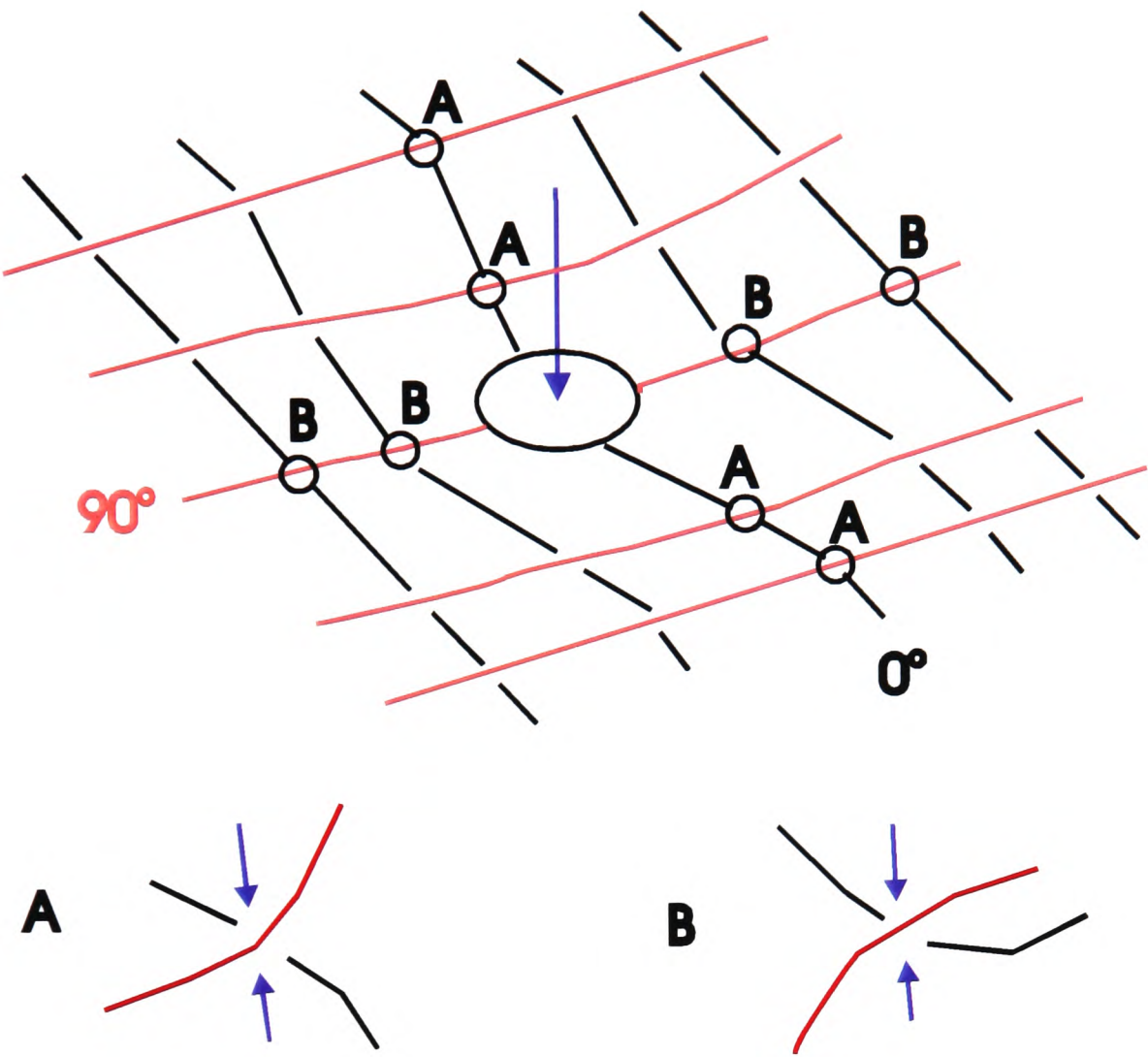


Figure 3.2

3.2 Failure Criteria

Approximate methods are not sufficient to accurately predict failure. In order to predict the initiation of damage in a composite under impact loading a failure criterion is necessary. The main criteria that have been proposed are considered here.

Initial criteria developed for composites considered single plies and plane stress. These were developed into criteria for multi-ply laminates subjected to three-dimensional stress states and with several failure modes.

Stress-based criteria have been almost universally adopted. Some have proposed simple uniaxial maximum value criteria. However most authors have accepted that biaxial, or triaxial, loading will cause failure at values below the uniaxial strengths and consequently have proposed interaction equation criteria.

3.2.1 Single Ply Failure Criteria

Maximum Value Criteria

The simplest composite failure criterion is the maximum stress criterion proposed by Murray and Schwer [19]. Failure occurs when a direct stress reaches a strength value. Alternative maximum value criteria are the maximum strain criterion, or the Grant-Sanders criterion. The Grant-Sanders criterion uses a maximum stress criterion for longitudinal fibre failure and a maximum strain criterion for both longitudinal and transverse matrix cracking. Maximum value criteria are convenient for their simplicity, though they take no account of the interaction effects of biaxial stress states.

Maximum Stress Criterion

$$\begin{aligned}
 \sigma_1 &= T_1 \text{ or } C_1 && \text{(Fibre failure)} \\
 \sigma_2 &= T_2 \text{ or } C_2 && \text{(Matrix failure)} \\
 |\tau_{12}| &= S_{12} && \text{(Shear failure)}
 \end{aligned}
 \tag{3.5}$$

Maximum Strain Criterion

$$\begin{aligned}
 \epsilon_1 &= \epsilon_{ft} \text{ or } \epsilon_{fc} && \text{(Fibre failure)} \\
 \epsilon_2 &= \epsilon_{mt} \text{ or } \epsilon_{mc} && \text{(Matrix failure)} \\
 |\gamma_{12}| &= \gamma_{12s} && \text{(Shear failure)}
 \end{aligned}
 \tag{3.6}$$

Grant-Sanders Criterion

$$\begin{aligned}
 \sigma_1 &= T_1 \text{ or } C_1 && \text{(Fibre failure)} \\
 \epsilon_{m1} &= \epsilon_{mt} && \text{(Matrix cracking in fibre direction)} \\
 \epsilon_{m2} &= \epsilon_{mt} && \text{(Matrix failure)} \\
 |\tau_{12}| &= S_{12} && \text{(Shear failure)}
 \end{aligned}
 \tag{3.7}$$

Interaction Criteria

The maximum value criteria fit experimental data well for predominantly uniaxial loading, but are less accurate in predicting failure under biaxial loading. The stress interaction criteria below have been proposed for biaxial stress states.

The Puck Criterion uses an interaction equation for matrix failure and a maximum stress criterion for fibre failure.

Puck Criterion

$$\sigma_1 = T_1 \text{ or } C_1 \quad (\text{Fibre failure}) \quad 3.8$$

$$\left(\frac{\sigma_2}{T_2 C_2} \right)^2 + \sigma_2 \left(\frac{1}{T_2} - \frac{1}{C_2} \right) + \left(\frac{\tau_{12}}{S_{12}} \right)^2 = 1 \quad (\text{Matrix failure}) \quad 3.9$$

Unlike the Puck criterion the Tsai-Hill [19] criterion does not differentiate between failure modes and uses a single interaction equation to predict composite failure.

Tsai-Hill Criterion

$$\left(\frac{\sigma_1}{T_1} \right)^2 - \frac{\sigma_1 \sigma_2}{T_1^2} + \left(\frac{\sigma_2}{T_2} \right)^2 + \left(\frac{\sigma_{12}}{S_{12}} \right)^2 = 1 \quad (\text{Composite failure}) \quad 3.10$$

3.2.2 Multiply Laminate Failure Criteria

In order to predict the failure of a three-dimensional, laminated composite, the failure criteria must consider the full, three dimensional stress state. The criteria must account for the through-thickness stresses as well as the in-plane stresses. The 3D Tsai-Wu [20] criterion is a single interaction equation for composite failure that considers the three dimensional stress state.

3D Tsai-Wu Criterion

$$\frac{\sigma_1^2}{T_1 C_1} + \frac{(\sigma_2^2 + \sigma_3^2)}{T_2 C_2} + \frac{(\tau_{12}^2 + \tau_{13}^2 + \tau_{23}^2)}{S^2} - \frac{(\sigma_1 \sigma_2 + \sigma_1 \sigma_3)}{\sqrt{1/T_1 C_1 T_2 C_2}}$$

$$- \frac{\sigma_2 \sigma_3}{T_2 C_2} + \sigma_1 \left(\frac{1}{T_1} - \frac{1}{C_1} \right) + (\sigma_2 + \sigma_3) \left(\frac{1}{T_2} - \frac{1}{C_2} \right) = 1 \quad (\text{Composite failure}) \quad 3.11$$

The Tsai-Wu criterion does not distinguish different composite failure modes. The Hashin [21] criterion, however, does. In the Hashin criterion there are four separate criteria, two for fibre failure, one for matrix cracking and one for matrix crushing. By considering different failure modes in this way it is possible to perform an analysis which can record the progression of the three different modes of failure.

Hashin Criterion

Fibre Tensile Failure

$$\left(\frac{\sigma_1}{T_1} \right) + \left(\frac{\tau_{12}^2 + \tau_{13}^2}{S_{13}^2} \right) = 1 \quad 3.12$$

Fibre Compressive Failure

$$\sigma_1 = C_1 \quad 3.13$$

Matrix Cracking

$$\left(\frac{\sigma_2 + \sigma_3}{T_2} \right)^2 + \left(\frac{\tau_{23}^2 - \sigma_2 \sigma_3}{S_{23}^2} \right) + \left(\frac{\tau_{12}^2 + \tau_{13}^2}{S_{13}^2} \right) = 1 \quad 3.14$$

Matrix Crushing

$$\left(\frac{\sigma_2 + \sigma_3}{2S_{23}} \right)^2 + \left[\left(\frac{C_2}{2S_{23}} \right)^2 - 1 \right] \left(\frac{\sigma_2 + \sigma_3}{C_2} \right) + \left(\frac{\tau_{23}^2 - \sigma_2 \sigma_3}{S_{23}^2} \right) + \left(\frac{\tau_{12}^2 + \tau_{13}^2}{S_{13}^2} \right) = 1 \quad 3.15$$

As an alternative to the Hashin criterion, Chang & Chang [21] proposed their own failure criterion to be implemented in their progressive damage model. The Chang-Chang criterion is simpler than the Hashin criterion. There is only one criterion for fibre failure and both the interaction equations for matrix cracking and crushing failure modes have fewer terms. This reduced complexity makes the Chang-Chang criterion more suitable for numerical analysis since computation time can be reduced.

Chang-Chang Criterion

Fibre breakage

$$\left(\frac{+\sigma_1}{T_1} \right)^2 + \left(\frac{\tau_{12}}{S_{12}} \right)^2 = 1 \quad 3.16$$

Matrix cracking

$$\left(\frac{+\sigma_2}{T_2}\right)^2 + \left(\frac{\tau_{12}}{S_{12}}\right)^2 = 1 \quad 3.17$$

Matrix crushing

$$\frac{1}{4}\left(\frac{-\sigma_2}{S_{12}}\right)^2 + \frac{C_2\sigma_2}{4S_{12}^2} - \frac{\sigma_2}{C_2} + \left(\frac{\tau_{12}}{S_{12}}\right)^2 = 1 \quad 3.18$$

3.3 Impact Energy

Stress or strain based criteria have been shown to be appropriate tools for predicting the initiation of failure in composites. However, because of the complex stress distributions in post failure damage progression, many authors have suggested that it is more useful to look at the energy transfers involved in the impact event.

The experimental studies conducted for this thesis have shown a strong correlation between incident impact energy and resulting delamination area (Figure 2.21 and 2.23). It has also been found that there is close agreement between the delamination area resulting from high speed, low speed and quasi-static impacts at equivalent incident impact energies or indentation work done (Figures 2.28 and 2.34). This suggests that energy may be the most useful measure for predicting and comparing impact damage in composites.

Sun [23] numerically analysed the impact of a steel ball on a laminated composite cantilever with the aim of predicting the work done by the projectile on the target. It was proposed that the incident kinetic energy of the projectile was absorbed in two forms: vibrational energy of motion and displacement of the laminate; and energy dissipated as damage in the contact zone. The energy absorbed was calculated either by comparing the initial and rebound velocities (v_0 and v_1) of the projectile or computing the work done by the contact force:

$$U_d = U_0 - U_1 = \frac{1}{2}m(v_0^2 - v_1^2) = \int_0^{\delta_1} P d\delta \quad 3.19$$

Lifshitz et al. [24] conducted low-velocity drop weight impact tests on 100 mm and 55 mm long

48 and 40 ply CFRP beams. They compared the energy dissipated in both impact and quasi-static tests using a similar approach to Sun and made the following observations:

- 1. The effective rigidity of the beams was the same under impact and static loading.
- 2. The failure load and work done at the point of failure under impact were lower than the corresponding values under static conditions ($U_{df} < U_{sf}$).
- 3. However, the incident kinetic energy, U_0 , must be at least equal to the static fracture energy, U_{sf} , in order to cause fracture of the beam. The excess kinetic energy, above the impact fracture energy, U_{df} , increases the elastic deformation and the extent of damage.
- 4. Short beams fail by delamination between inner layers, and long beams fail by fracture of back layers.

In the plate impact tests conducted here, the quasi-static fracture energy was lower than the impact damage energy threshold for both 21 and 29 ply coupons (Table 3.1). This result is not consistent with the observations of Lifshitz for impacts on beams.

Threshold Indentation Fracture Energies

Fracture Energy	Quasi-static, U_{sf}	Low Speed Impact, U_{df}
21 ply Coupon	1.15 J	2 J
29 ply Coupon	1.46 J	2.5 J

Table 3.1

Delfosse and Poursartip [14] examined the energy dissipated in plate impact tests and proposed the following energy balance for impacts on CFRP laminates:

$$U_{impact} = U_{elastic} + U_{indent} + U_r + U_{md} + U_{fd} + U_{sl}$$

3.20

where U_{impact} is the incident kinetic energy of the impactor, $U_{elastic}$ is recoverable elastic strain energy in the laminate, U_{indent} is energy absorbed in creating a permanent indentation, U_{md} and U_{fd} are the energies absorbed in matrix and fibre damage, and U_{sl} is system loss energy. System loss energy includes vibration, heat, inelastic behaviour of the supports and so forth. They suggested that the energy absorbed in matrix and fibre damage may be related to the resulting damage area A by fracture mechanics with an energy release rate term G :

$$U_{md} = G_{md} A_{md} \quad 3.21$$

$$U_{fd} = G_{fd} A_{fd} \quad 3.22$$

As a general conclusion Delfosse and Poursartip state that, “whereas a force-based approach works well for the *onset* of damage, an energy-based approach is more suitable for the determination of the *extent* of damage”.

An examination of the force/displacement curves for quasi-static indentation tests (Figure 2.30) suggests that this conclusion is appropriate. First failure points can be fixed by the maximum force reached, but after first failure, due to drops in the load supported by the damaged coupon, it is necessary to use displacement, or energy, to determine the extent of damage.

3.4 Prediction of Delamination by Fracture Mechanics

For some time delamination has been regarded as the most critical damage mode affecting the post impact compression strength of composites. This explains the attention of researchers and why so many attempts have been made to predict the delamination damage resulting from impact.

By considering energy dissipation during impact, some authors have attempted to predict the size of delamination damage in composite laminates using techniques based on fracture mechanics.

Wu and Springer [25] have proposed a method for predicting the locations and sizes of delaminations occurring in a rectangular composite plate subjected to low-velocity impact. Finite element analysis is used to predict the stress distribution in the plate under impact and the sizes of the delaminations between plies are then predicted by a proposed failure criterion. For the stress analysis the following assumptions are made: the composite material is elastic; strains vary linearly with displacements; there is no friction between impactor and plate; damping is negligible; gravitational forces are negligible; and the properties of the composite plate remain unchanged during impact (i.e. the properties of the damaged material are the same as the undamaged material). A Hertzian contact between impactor and plate is assumed with an additional allowance for permanent indentation. Eight-noded brick elements are used in the finite element model of the plate. Wu and Springer's delamination criterion was developed by

dimensional analysis. The length, l_D , and width, w_D , of the delamination at a particular ply interface are given by equations of the form:

$$\frac{l_D}{l_0} = C_1 \left(\frac{\sigma_{33\max} \sqrt{l_0}}{K_{IC}} \right)^{C_2} (Q^*)^{C_3} \left(\frac{t_f}{t_m} \right)^{C_4} \left(\frac{D^B}{D^T} \right)^{C_5} \quad 3.23$$

$$l_0 = \frac{2K_{IC}^2}{\pi T_2^2} \quad 3.24$$

Where K_{IC} is the fracture toughness and T_2 is the transverse tensile strength. σ_{max} is the maximum stress in the through thickness direction during impact, calculated by finite element analysis; t_f is the total duration of the impact; and t_m the time taken for the stress to reach the peak value. Q^* is the dimensionless difference of stiffness in the direction considered of the two plies between which delamination occurs. The constants C_1 to C_5 must be determined experimentally for a particular material architecture.

Although Wu and Springer achieved good agreement between their model and experiments their criterion has not been adopted by other authors. The fact that the model only predicts delamination, and no other failure modes, is a major drawback. Assuming that the laminate behaviour is linearly elastic throughout the impact is also very limiting. The model is really only applicable to impact situations where the predominant, or only, damage mode is delamination, and the damage has negligible effect on the overall stiffness of the laminate structure.

Wu and Springer's approach was empirical; their equation was developed to fit experimental data. Other authors have applied fracture mechanics more fundamentally to delamination cracks.

Liu and Chang [26] used fracture mechanics to investigate the growth of a single delamination in a laminated composite plate under transverse concentrated loading. The analysis was quasi-static but the results are applicable to low speed impacts where stress wave effects may be neglected. It was assumed that matrix cracking is the initial failure mode and that delaminations grow from those matrix cracks. Finite element analysis was used to determine the stress distribution in the plate and the delamination was grown incrementally by using the following

delamination criterion:

$$\frac{G_I}{G_{IC}} + \frac{G_{II}}{G_{IIC}} + \frac{G_{III}}{G_{IIIC}} = 1 \quad 3.25$$

where G and G_c are the strain energy release rate and critical strain energy release rate for crack opening modes I, II and III (opening, sliding and tearing). Analysis indicated that the size of the initial matrix crack significantly affects the subsequent growth of delamination. Liu and Chang's model indicates that a fracture mechanics criterion may successfully be used to predict the growth of delaminations. However, their analysis is not itself directly applicable to other problems. An initial matrix crack must be independently defined by some other analysis or criterion and the model considers only a single delamination.

Fracture mechanics modelling of delamination damage which is applicable to real problems requires a technique which firstly predicts the matrix cracks which initiate delamination, and then predicts the growth of the delaminations themselves. Finn and Springer have proposed such a technique.

Like Liu and Chang, Finn and Springer [27] also maintain that transverse matrix cracking is the precursor to delamination. They proposed a method for calculating the locations, shapes and sizes of delaminations resulting from impact on a laminate. The technique combines a stress based failure criterion for matrix cracking with a fracture mechanics energy balance for delamination. They assume that for a delamination to occur there must first be transverse matrix cracking in a ply adjacent to the interface. To predict this matrix cracking they propose the use of dynamic finite element analysis of the impact with one of the three-dimensional stress interaction failure criteria for transverse matrix cracking examined above (Section 3.2).

Finn and Springer also assume that the matrix crack must open up for a delamination to be produced. This leads them to propose an energy criterion for delamination crack growth that only considers the contribution of stresses in the adjacent plies that will open a longitudinal matrix crack in that ply. They consider an interface at which either the top or bottom ply (or ply group) is cracked and assume that delamination will occur if the 'useful' strain energy available from the plies exceeds the fracture work, Γ . Their criterion is:

$$U_t dV_t + U_b dV_b \geq \Gamma dA \quad 3.26$$

where:

$$U_t = \frac{1}{2}(\tau_{23}\gamma_{23} + \Omega\tau_{12}\gamma_{12}) \quad 3.27$$

$$U_b = \frac{1}{2}\Omega(\sigma_{22}^+\varepsilon_{22} + \tau_{12}\gamma_{12}) \quad 3.28$$

U_t and U_b are the strain energy densities of the top and bottom plies of the interface, respectively.

$\Omega = 1$ if the ply group's other interface is unrestrained

$\Omega = 0$ if the ply group's other interface is restrained

The other interface of a ply or ply group is unrestrained if either it is the outside ply group of the laminate or if the other interface is delaminated from a previous time increment in the dynamic analysis.

Finn and Springer's method is unfortunately limited to non-penetrating, low-velocity, impacts. The method uses dynamic finite element analysis to provide the stress and strain fields for the delamination criterion and therefore can calculate the development of delamination areas throughout the impact. However the properties of the composite are assumed to be unchanged by damage. Transverse matrix cracking and delamination are the only failure modes predicted, and although the locations of these failures are predicted, they are assumed to have no effect on the stiffness of the composite structure. This limits the technique to impacts in which the only, or dominating, damage mode is delamination.

3.5 Numerical Analysis Incorporating Coupon Tests

Numerical techniques involving stress based failure criteria and/or fracture mechanics seem promising for analysing impact damage in composites. However, the complexity of modelling brittle fracture has led some authors to attempt to simplify the analysis by incorporating measurements from small scale coupon tests into models of larger structures.

For low velocity impacts Robinson and Davies [28] observed that impact damage is independent of the impactor mass and a function of the impact energy only. They also observed that the same level of damage occurred in structures of different scales at the same peak impact force with the same level of absorbed energy. Consequently they proposed a model for the prediction of the impact energy required to cause a certain level of damage in a structure from

the results of a coupon test. Their method is as follows:

- i) Determine maximum contact force P_{max} in coupon test from force-time trace.
- ii) Determine damage energy U_d absorbed in coupon test (either from measurement of loss in kinetic energy of rebounding impactor or by calculation of maximum strain energy in coupon).
- iii) Calculate elastic strain energy in structure for the same peak force P_{max} from (i). Complicated structures may require the use of numerical finite element analysis.
- iv) Add elastic strain energy from (iii) to the damage energy U_d from (ii) to find the incident impact energy to cause the same level of damage in the structure as in the coupon.

This technique eliminates the requirement to model the impact damage process entirely. It is only necessary to conduct a linear elastic stress analysis of the structure. However, any analysis also requires a small scale coupon test on the same material and lay-up as the structure.

Davies et al. [29] further developed this work to “embed” coupon tests as a calibrator in the finite element model of the full-sized structure.

3.6 Numerical Analysis with Stress Based Failure Criteria

Coupon tests incorporated into structural models can reduce the need for complex analysis but are limited to particular impact geometries. Every different impact condition and laminate architecture will require a different coupon test. There is still a requirement for a general technique that can analyse a broad range of impact types and material architectures and structures. Stress based failure criteria implemented in finite element analysis to model damage progression has been proposed as a solution. Success of this technique in modelling quasi-static loading of composites has led to its application to dynamic, impact, problems.

Chang & Chang [22] proposed a progressive damage model for two dimensional notched composites subjected to tensile loading. The model is capable of assessing damage in laminates with arbitrary ply orientations and of predicting the ultimate tensile strength of a notched laminate. The model consists of two parts: the stress analysis; and the failure analysis. Stresses and strains are analysed using classical lamination theory. The failure analysis relies on a

proposed failure criterion combined with a proposed property degradation model for damage accumulation. The model was validated by comparison of numerical results with experimental data from tensile tests of laminates containing open circular holes.

In the stress analysis the tensile load is applied incrementally, and, as local failures are predicted and modelled, the stresses redistribute. The set of failure criteria proposed by Chang & Chang has been outlined above (3.2.2). Three different in-plane failure modes were considered: fibre-matrix shearing; fibre breakage; and matrix cracking.

As failure occurs in an element the property degradation model is applied. For matrix cracking the transverse modulus, E_2 , and Poisson's ratio, ν_{12} , are reduced to zero. For fibre breakage and shearing, E_2 and ν_{12} are also reduced to zero, and the longitudinal modulus, E_1 , and shear modulus, G_{12} , are both reduced according to a Weibull distribution dependent on the damage zone area A .

The numerical procedure followed in the analysis is as follows:

- 1) Load is increased incrementally
- 2) Stresses and strains calculated
- 3) Failure criteria checked:
 - If no damage return to 1st step
 - If damaged then change mechanical properties and redistribute stresses

Although the Chang & Chang failure criteria were initially proposed for two dimensional, quasi-static, loading problems they have subsequently been applied by others to three dimensional, dynamic problems with some success.

Majeed et al. [30] modelled transverse impact on composite coupons using an explicit, non-linear, finite element code (LS-DYNA) and Chang & Chang's failure criteria. The post failure degradation rules were slightly modified from those originally proposed, however. For all three failure modes the relevant stiffnesses were reduced to zero instantaneously, in one time step.

The post failure degradation rules are:

Fibre failure	$E_1 = E_2 = G_{12} = \nu_{12} = 0$
---------------	-------------------------------------

Matrix cracking	$E_2 = G_{12} = \nu_{12} = 0$
Matrix crushing	$E_2 = \nu_{12} = 0$

Both low (5.4 J, 1.37 m/s) and high energy (30.4 J, 3.23 m/s) impacts on a coupon were modelled. The coupon modelled was 152.4 x 101.6 mm and a 4.44 mm thick quasi-isotropic CFRP lay-up. Belytschko-Tsay shell elements were used to model the coupon. Since the shell elements have no through thickness stiffness it was also necessary to model the local coupon/impactor indentation stiffness with a numerical spring.

Numerically derived force and strain history data were compared with experimental data. Low energy simulations were largely elastic, and the impact force and strain history results showed satisfactory agreement with the experimental data. The high-energy models generated significant damage, and the impact force history results were found to be highly dependent on the mesh density used for the coupon.

Majeed et al. concluded from their investigation that low energy impact could be modelled satisfactorily in this way but that the high energy impact results were inaccurate. They noted that the Chang & Chang failure criteria were proposed for two dimensional, quasi-static problems and were therefore not optimised for these impact analyses. It was also suggested that the post-failure degradation rules were perhaps too severe. Reducing element stiffnesses to zero in one time step in a dynamic analysis will result in stress wave effects which could significantly affect the analysis results.

3.7 Conclusions

We have seen that there is a limited role for approximate methods in the analysis of impact on composite structures. First estimates of elastic structural response may be obtained as a design aid, but prediction of failure is rather more difficult. We can qualitatively describe the type of damage we may expect, in particular the orientation of delaminations in a laminate but quantitative predictions require more complex numerical methods.

There has been no generally adopted fracture mechanics approach to impact damage in composites despite the potential suitability of the approach. The problems involved are both those particular to composites, and also the general problems of modelling dynamic fracture

mechanics. The difficulties with composites include the problems of modelling multiple cracks with several different, interacting failure modes and the requirement to combine failure initiation criteria with crack growth criteria. The more general problems of modelling dynamic fracture mechanics are the difficulties of modelling the rate of growth of a crack while accounting for dynamic, inertial effects and also avoiding strong dependence of the solution on the degree of discretisation of the finite element analysis.

For this thesis one approach has been taken in an attempt to model the experimental coupon impact tests conducted and to appraise the technique. The approach taken builds on the work of Majeed et al. using a stress based failure criteria in a dynamic finite element analysis with post-failure property degradation rules.

The work of Majeed et al. using modified versions of the Chang & Chang failure criteria is promising. However some points must be addressed. If through-thickness direct and shear stresses are significant then we must use brick elements rather than shell elements in the finite element mesh. This allows the through-thickness direct and shear stresses to be modelled. It is also apparent that an additional failure criterion is necessary to model delamination. Majeed et al. observed that the instantaneous post failure property degradation rules they used were rather severe. This suggests that an approach using a progressive reduction in stresses after failure could be more successful.

Chapter 4

Numerical Analysis of Impact

4.1 DYNA 3D

Numerical modelling of the coupon impact tests was conducted for this thesis using the Vec/DYNA 3D finite element method computer program package. Laminated composite failure is modelled within DYNA by the use of a composite damage model based on the failure criteria proposed by Chang & Chang. The Vec/DYNA 3D package consists of three programs: a pre-processor INGRID; the finite element calculation program DYNA; and a post-processor program TAURUS.

DYNA was developed in the early 1980s at the Lawrence Livermore National Laboratory, and was further developed by J. O. Hallquist [31] at Livermore Software Technology Corporation. OASYS LS-DYNA [32] is a developed version of the DYNA3D code marketed by Ove Arup Systems. The package includes an improved post-processor package and this was used for some of the analysis reported here.

4.1.1 Finite Element Method

The finite element method is an approximate numerical technique for finding a solution to a physical problem governed by known differential equations. An approximate numerical technique is necessary where an exact, closed form, mathematical solution either does not exist or would be too labourious to determine. Finite element analysis is a technique which allows the problem to be reduced to a number of trivial numerical calculations and is therefore ideal for solution by computer.

The finite-difference method is a related, and predecessor, technique. In the finite-difference approximation to a differential equation, the derivatives in the equations are replaced by difference quotients which involve the values of the solution at discrete mesh points of the domain. The resulting discrete equations are solved, after imposing the boundary conditions, for the values of the solution at the mesh points. Although the finite difference method is simple in concept, it suffers from several disadvantages. The most notable are the inaccuracy of the derivatives of the approximated solution, the difficulty in accurately representing

geometrically complex domains, and the inability to employ non-uniform and non-rectangular meshes.

In the finite element method a geometrically complex domain of the problem is represented as a collection of geometrically simple subdomains, called finite elements. The elements may be non-uniform and non-rectangular. Over each finite element approximation functions are derived using the basic idea that any continuous function can be represented by a linear combination of algebraic polynomials. The undetermined parameters are the values of the solution at a finite number of preselected points, called nodes, on the boundary and in the interior of each element. Computation of the finite element mesh, with application of the boundary conditions, provides numerical values for the solution at each node.

The finite element method is most commonly applied to model the stress and strain distributions in loaded structures. For this thesis it has been used to model the complex dynamic problem of impact involving material failure on a composite coupon.

4.1.2 Explicit Time Integration

DYNA3D is a finite element analysis package designed for the efficient solution of dynamic events. It uses explicit, rather than implicit, time integration and this makes the package particularly suitable for dynamic and impact events. Implicit codes are generally more suitable for quasi-static problems.

In an explicit code the acceleration of each node is assumed to be constant throughout one time step. This makes inversion of the mass matrix to find the solution at one time step relatively trivial. For stability the time step must be less than the shortest natural period of the mesh and therefore usually depends on the size of the smallest element in the mesh. This ensures accurate modelling of stress wave transmission. The time step may also be limited by material and geometric non-linearities or contact and friction algorithms which may need a short time step for accuracy.

In an implicit code, acceleration is assumed to vary linearly over one time step, this allows greater accuracy over a longer time step. However, inversion of the mass matrix is no longer trivial and takes up the majority of the computation time. The time step used in an implicit

analysis must be less than the shortest natural period of the structure that is of interest; this tends to give a much larger time step than that used in an explicit analysis (typically 100-1000 times larger). Table 4.1 compares the major differences between explicit and implicit finite element codes.

Comparison of Explicit and Implicit Finite Element Codes

	EXPLICIT	IMPLICIT
Matrix Inversion	No	Yes
Timestep Size	Small	Can be large
Timestep Limited By	Stability	Accuracy
Cost Per Timestep	Small	Large
Cost Dominated By	Element Processing	Matrix Inversion (Time Integration)
Cost Increase for Doubling Mesh Density	x 16	x 128
Element Formulation	Simple	Complex

Table 4.1

Analysis cost is defined as the length of computation time necessary to obtain a solution. With complex, non-linear problems this time can be very considerable even on powerful, modern computers.

In the implicit code, typically 75 to 90% of the computation time is involved in matrix inversion. It is therefore most efficient to use complex, accurate elements, and have as long a time step as possible. We may put the computation cost as:

Implicit Code Cost = $C_1N(1+C_2W^2)$ (cost $\propto$ NW^2)

where N = the number of degrees of freedom and W = the width of the stiffness matrix wavefront. If increase the mesh density by a factor S , the cost of a 3D mesh increases as S^7 .

In the explicit code practically all the computational time is in element processing. It is therefore advantageous to use simpler elements but shorter time steps to maintain accuracy.

Explicit Code Cost = $C_1N_1 + C_2N_2 + \dots$ (cost $\propto$ NS)

where N_i = number of elements of type i and C_i = cost per element of type i . In 3D the cost increases as S^4 .

Therefore an explicit analysis is cheaper computationally when the model is large and there are factors limiting the time step. An impact event is an example of such a case and we have used an explicit code (DYNA 3D) here for modelling impact damage in composites.

4.1.3 Element Formulation

In DYNA 3D the elements are formulated to maximise the accuracy relative to the time taken to process the elements, as the cost of introducing extra degrees of freedom is small.

For the CFRP impact analysis 8 node brick elements were used with 3 degrees of freedom at each node. The elements have a single integration point (rather than the more usual 8 point $2 \times 2 \times 2$). This speeds processing, though the accuracy of the individual element, is reduced and there can be problems with hourglassing (section 4.1.4).

4.1.4 Hourglassing

The solid brick element has 24 degrees of freedom (8 nodes, each with 3 degrees of freedom). The single integration point calculates 6 stresses (3 direct and 3 shear) so therefore only 6 modes of deformation are stiff. There are 6 rigid body modes of deformation (3 translational and 3 rotational) which leaves a further 12 modes of deformation which have no stiffness and have to be suppressed. These are the so called hourglassing modes. The four hourglass modes on one axis of an eight node element with a single integration point are shown in Figure 4.1.

Mesh distortion due to hourglassing degrades the accuracy of the elements and if excessive could cause the analysis to “crash”. Hourglassing can be suppressed by either stiffness or viscous control. Viscous control is used in DYNA 3D. Forces are applied at the nodes to oppose velocities in the hourglass mode shapes. Hourglassing problems can be avoided by not subjecting single solid elements to bending, not applying point loads, and avoiding single node or line contacts in contact surfaces.

In DYNA 3D the default value of the hourglass viscosity is 0.1. Despite this hourglassing effects were observed in the analysis and consequently the hourglass viscosity was therefore

increased to the maximum recommended value of 0.15. This was found to reduce the hourglassing problem and improve the analysis results.

Hourglass Modes

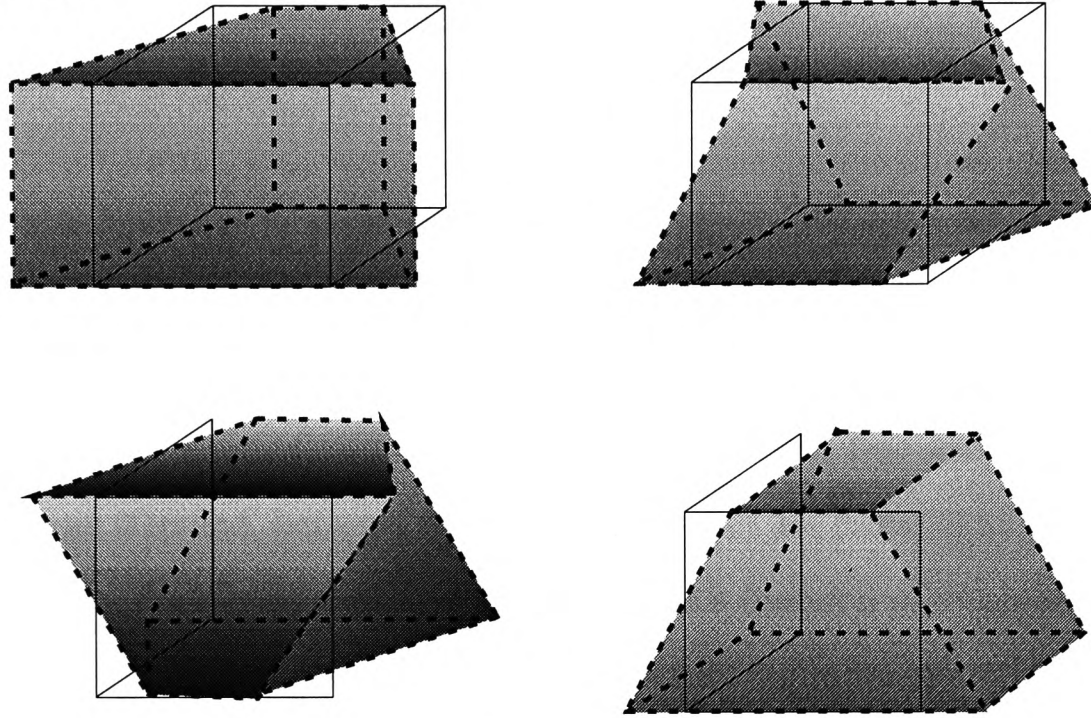


Figure 4.1

4.1.5 Contact Surfaces

In order to determine the contact behaviour between two bodies in DYNA3D it is necessary to define contact surfaces. For any possible contact one master and at least one slave surface must be defined. The contact surface algorithm in DYNA3D checks to see if nodes of one surface have penetrated the other and if so applies a force at the nodes proportional to the penetration. The interface stiffness used is calculated from the bulk modulus, volume and surface area of the master element. For modelling of the coupon impact two contact surfaces were defined. One was the coupon and support contact, the other the coupon and impactor contact.

4.1.6 Composite Failure Criteria and Material Type

The composite failure criteria implemented in DYNA 3D are a modified version of the Chang & Chang stress based failure criteria. They are denoted as material type 22. Material 22 is approximately orthotropic elastic with the addition of a nonlinear shear stress term α . However, CFRP has generally been observed to be linear elastic up to failure and so α has been set to zero here.

A stress based failure model is included to incorporate the effects of fibre breakage, matrix

cracking, matrix crushing, and delamination failure modes. The failure modes are shown in Figure 4.2 with respect to the local (ply) coordinate system. Upon failure the stress components corresponding to the failure mode are reduced to zero linearly over the next 100 time steps. This stress reduction function has been found useful to minimise spurious oscillations, or ringing, that would be caused by the abrupt release of energy at failure.

The following assumptions are made in adopting this type of failure mechanism:

- Fibre dominated behaviour of the composite.
- Each layer is represented by one brick element.
- Mesh density will affect damage propagation. A fine mesh will give a smaller time step and, because at failure the stresses are reduced to zero over 100 time steps, the rate of stress reduction will be a function of mesh density.
- No separation of layers can occur. Delaminations are modelled as a reduction of the normal stress within one layer.

Fibre Fracture Criterion

$$\left(\frac{+\sigma_1}{T_1} \right)^2 + \left(\frac{\tau_{12}}{S_{12}} \right)^2 = 1 \quad 4.1$$

The material strengths for T300/914 are:

$T_1 = 1700 \text{ MPa}$ (Longitudinal Tensile Strength)

$S_{12} = 86 \text{ MPa}$ (Ply Shear Strength)

Once failure has occurred the following stresses are reduced progressively to zero:

σ_1	Longitudinal Stress	σ_2	Transverse Stress
σ_3	Normal Stress	τ_{12}	Shear Stress
τ_{23}	Shear Stress	τ_{31}	Shear Stress

Matrix Cracking Criterion

$$\left(\frac{+\sigma_2}{T_2} \right)^2 + \left(\frac{\tau_{12}}{S_{12}} \right)^2 = 1 \quad 4.2$$

The material strengths for T300/914 are:

$T_2 = 74 \text{ MPa}$ (Transverse Tensile Strength)

$S_{12} = 86 \text{ MPa}$ (Ply Shear Strength)

Once failure has occurred the following stresses are reduced progressively to zero:

σ_2 Transverse Stress τ_{23} Shear Stress

Matrix Crushing Criterion

$$\frac{1}{4} \left(\frac{-\sigma_2}{S_{12}} \right)^2 + \frac{C_2 \sigma_2}{4S_{12}^2} - \frac{\sigma_2}{C_2} + \left(\frac{\tau_{12}}{S_{12}} \right)^2 = 1 \tag{4.3}$$

The material strengths for T300/914 are:

$C_2 = 237 \text{ MPa}$ (Transverse Compressive Strength)

$S_{12} = 86 \text{ MPa}$ (Ply Shear Strength)

Once failure has occurred the following stresses are reduced progressively to zero:

σ_2 Transverse Stress τ_{23} Shear Stress

Delamination Criterion

$$\left(\frac{+\sigma_{33}}{T_3} \right)^2 + \left(\frac{\tau_{31}}{S_{31}} \right)^2 + \left(\frac{\tau_{23}}{S_{23}} \right)^2 = 1 \tag{4.4}$$

The material strengths for T300/914 are:

$T_3 = 74 \text{ MPa}$ (Normal Tensile Strength)

$S_{13} = 86 \text{ MPa}$ (Transverse Shear Strength)

$S_{23} = 86 \text{ MPa}$ (Transverse Shear Strength)

Once failure has occurred the following stresses are reduced to zero:

σ_1 Normal Stress τ_{23} Shear Stress

τ_{31} Shear Stress

Material 22 Failure Modes

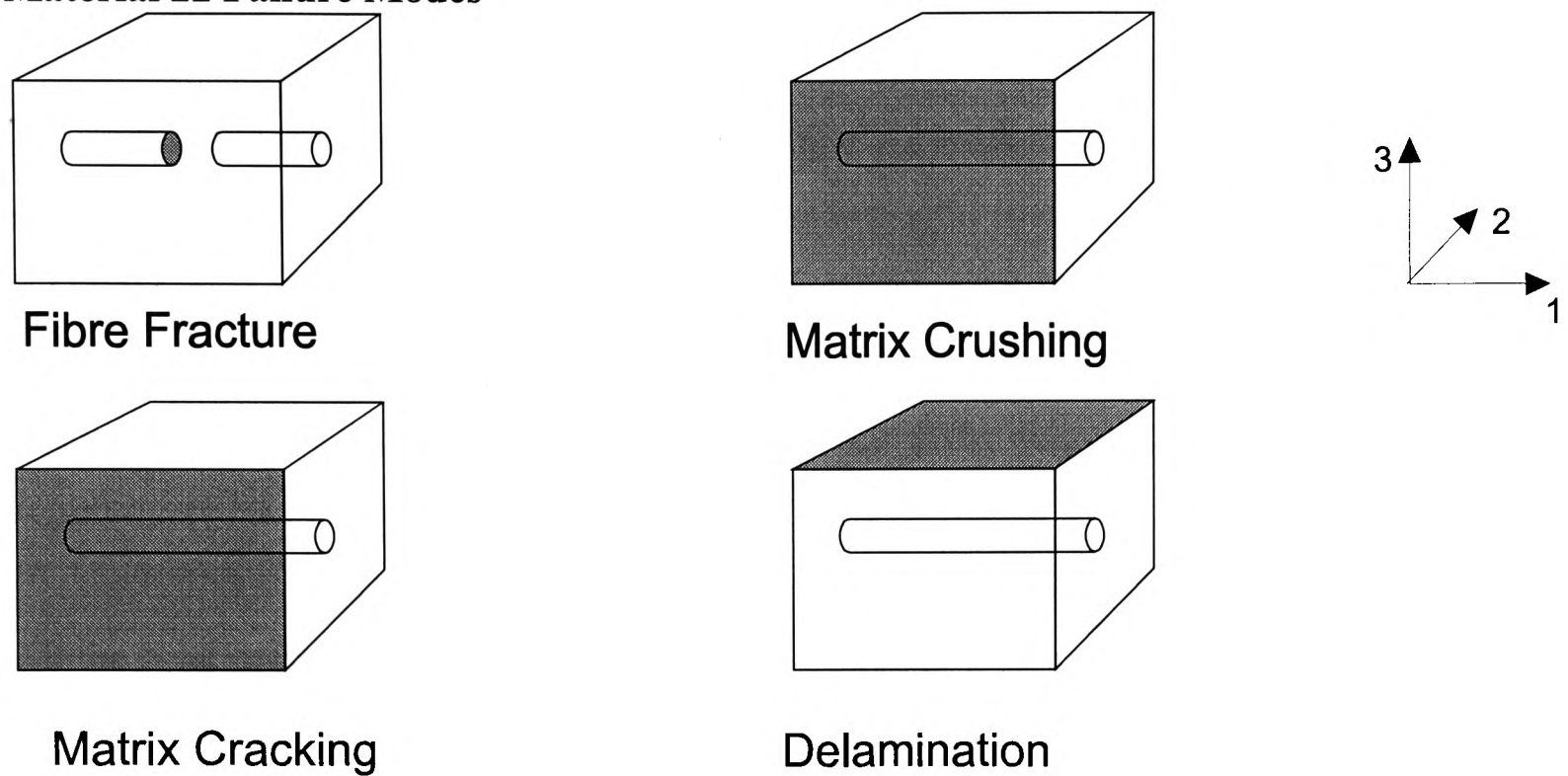


Figure 4.2

4.1.7 Description of Model

Analysis of the coupon impact event was performed for both quasi-static indentation and high speed impacts. Both models consisted of three parts: the coupon; the support; and the indenter/projectile. Eight noded brick elements were used for all three parts. For efficiency of computation, analysis was conducted with a one quarter model with two planes of symmetry. All nodes lying on the planes of symmetry were restrained to provide the correct boundary conditions in the global x and y axis directions.

Global coordinate axes were aligned so that the x and y axes were in the plane of the coupon and parallel to the 0° and 90° fibre directions. and the z axis was normal to the coupon surface. Impact was in the z direction only.

The 21 plies of the coupon laminate, alternating $0^\circ/90^\circ$, were modelled with one element depth per ply, each of 0.124mm thickness. The coupon was modelled as one quarter (42.5 x 42.5 mm) of an 85 x 85 mm square to represent the 85 mm width of the experimental coupons. 20 x 20 elements in the xy plane were used with increased mesh density at the impact zone

Each ply in the coupon was modelled as a separate material to aid post-processor imaging of the damage progression. Each was defined as material type 22 - the orthotropic composite damage

model. The material strength and stiffness values used are detailed below (4.1.8, Table 4.2).

For simplicity the support was modelled as a ring of 12 brick elements with an internal radius of 22.5 mm (to model the 45 mm diameter window used in the experimental tests), and an external radius of 25 mm and a depth of 2.5 mm. Initial modelling was conducted with a much more complex modelled support with an internal radius edge of 0.5mm. However the stress concentration at the edge was found to be insignificant in the simulation and consequently the simpler support model was used for computational efficiency.

All the base nodes of the support were restrained in the z direction to provide the boundary condition for the impact.

Both the support and the impactors were modelled as fully rigid (material type 20) to further reduce computation time. Preliminary models had used a linear elastic material (material type 1) with the appropriate properties but the deformations were determined to be insignificant compared to the structural and indentation deformation of the coupon.

Sliding contact surfaces without friction were defined for the plate and support contact and the plate and impactor contact.

4.1.8 Material Data

The material data used for modelling the strength and stiffness of T300/914 CFRP were supplied by Rolls-Royce or evaluated from small specimen tests. Small specimen tests were conducted by Hallett [33] in order to obtain appropriate material data for use in DYNA 3D.

The material characterisation tests conducted were in-plane tension, through-thickness tension and shear. The data obtained from these tests were used to validate DYNA 3D simulations using material type 22 of the same tests. These were successful and therefore the same material data was applied to simulation of structural impact tests on beam specimens [34].

The development of damage in the simulated beam impact was very similar to that observed experimentally by high speed photography. Stiffness of the modelled beam was also consistent with the experimental force/time history. However, the impact failure force was rather lower

than that measured experimentally, being only 70% of the expected load. The high experimental failure threshold was attributed to the statistical, Weibull, effect seen in ceramics where the measured bending tensile strength is higher, due to the distribution of flaws, than the uniaxial tensile strength. DYNA 3D uses constant strength values, irrespective of the mode of loading and so cannot model this behaviour.

The same material data used by Hallett is used here for simulation of quasi-static and high speed impacts on plates. The values used are listed in Table 4.2.

Material Data Values for Unidirectional T300/914 CFRP

E ₁	Elastic modulus longitudinal direction	(N/m ²)	139E+09
E ₂	Elastic modulus transverse direction	(N/m ²)	9.4E+09
E ₃	Elastic modulus normal direction	(N/m ²)	9.4E+09
K _f	Bulk modulus of failed material	(N/m ²)	2E+9
S ₁₂	In plane (longitudinal/transverse) shear strength	(N/m ²)	86E+6
S ₂₃	Transverse/normal shear strength	(N/m ²)	86E+6
S ₃₁	Normal/longitudinal shear strength	(N/m ²)	86E+6
v ₁₂	Poisson's ratio in-plane		0.0209
v ₂₃	Poisson's ratio transverse/normal		0.33
v ₃₁	Poisson's ratio normal/longitudinal		0.0209
G ₁₂	Shear modulus	(N/m ²)	4.5E+9
G ₂₃	Shear modulus	(N/m ²)	2.85E+9
G ₃₁	Shear modulus	(N/m ²)	4.5E+9
T ₁	Longitudinal tensile strength (axial fibre failure)	(N/m ²)	1700E+6
T ₂	Transverse tensile strength (matrix failure)	(N/m ²)	74E+6
T ₃	Normal tensile strength (matrix failure)	(N/m ²)	74E+6
C ₂	Transverse compressive strength (matrix crushing)	(N/m ²)	237E+6
ρ	Density	(kg/m ³)	1.58E+3

Table 4.2

4.1.9 Strain Rate Dependency

There is no facility for strain rate dependency in Material 22 in DYNA 3D. Behaviour up to the point of failure is linear elastic for all four failure modes.

The small specimen tests on T300/914 conducted by Hallett [33] covered a range of strain rates from 10^{-4} to 600 s^{-1} . The tests were intended to reveal the magnitude of strain rate dependency of the composite material. However the variation in failure stress and strain as a function of the measured strain rate was generally less than the scatter in the experimental results.

In-plane tension tests showed a slight rate dependency with the strain to failure increasing by approximately 10% at high strain rates. The scatter for these tests was reasonable, approximately 6%, and the lower quasi-static failure strains can probably be attributed to relaxation (creep) of the epoxy matrix under load causing further stress redistribution into the load bearing fibres. Through-thickness tension tests however showed such large scatter (up to 50%) that it was not possible to accurately determine any strain rate dependency. Through-thickness shear tests also showed significant scatter (up to 30%) and no accurate trend in rate dependency could be determined.

In general it may be concluded that the strain rate dependency of T300/914 CFRP is small compared to the scatter observed, and for the purposes of numerical simulation of dynamic behaviour, rate dependency may be neglected.

4.2 Low Speed Impact

4.2.1 Impactor

The complete DYNA 3D finite element mesh used for the simulation of a low speed impact test on a 21 ply coupon is shown in Figure 4.3. The support ring is at the bottom of the diagram with the coupon and impactor on top.

To simulate low speed impact the impactor was given an initial velocity of 8.7 m/s. The density and dimensions were set to give a quarter model mass of 64g, equivalent to the experimental impactor 3 mass of 256g.

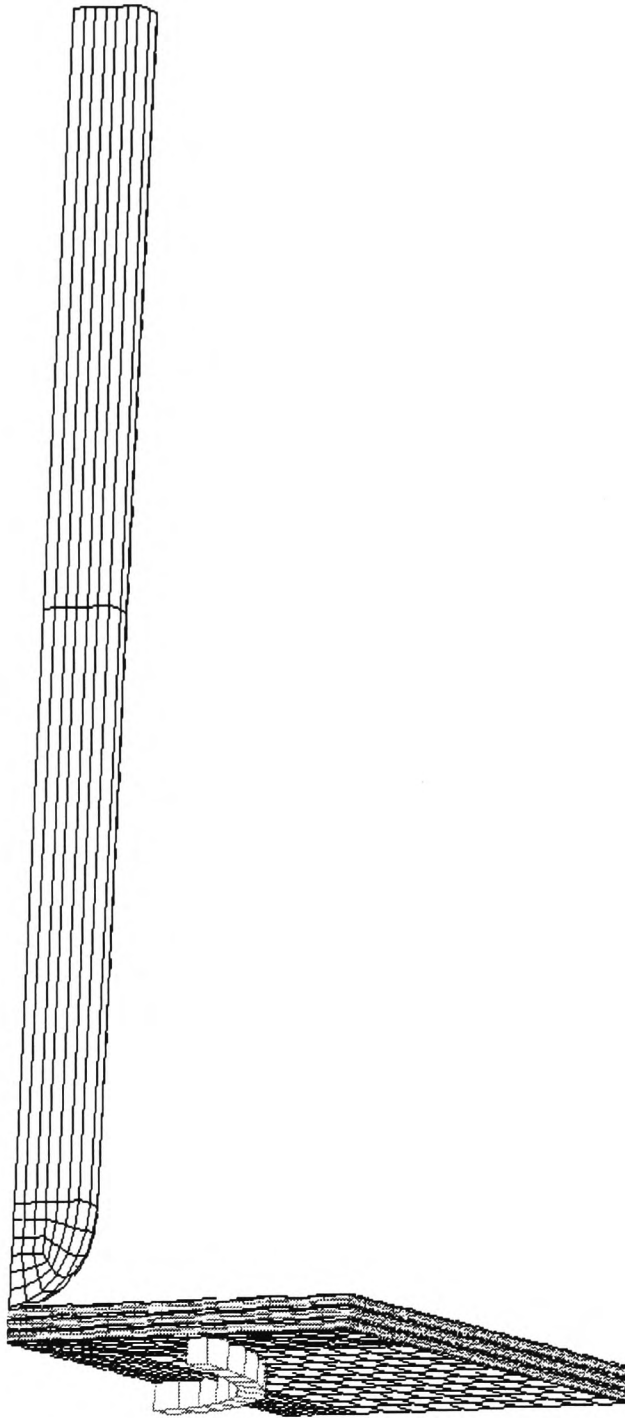


Figure 4.3

4.2.2 Indentation Stiffness

The dynamic indentation resistance provided by the coupon in the DYNA 3D simulation is compared with the experimental quasi-static stiffness of the 21 ply coupon in Figure 4.4. It is expected that at the low speed of impact (8.7 m/s) inertial (dynamic) effects of the CFRP coupon will be negligible and therefore the force/displacement behaviour of the simulated impactor should, in theory, match the quasi-static stiffness behaviour measured by experiment.

Although DYNA 3D correctly predicts drops in load due to progressive failure of the coupon the simulation is quantitatively wrong. First failure occurs at too low a load and the maximum load predicted is also too low. Although the average stiffness is initially correct the failure process does not match the experimental observation.

The area under the force/displacement graph is the work done on the coupon and therefore the absorbed energy. The rebound curve of the DYNA 3D simulation shows separation of impactor at a displacement of approximately 4 mm and the area within the curve represents the total work done. Although the simulation is not accurately reproducing the experimental stiffness behaviour it does give a reasonable approximation to the total energy absorbed in the impact.

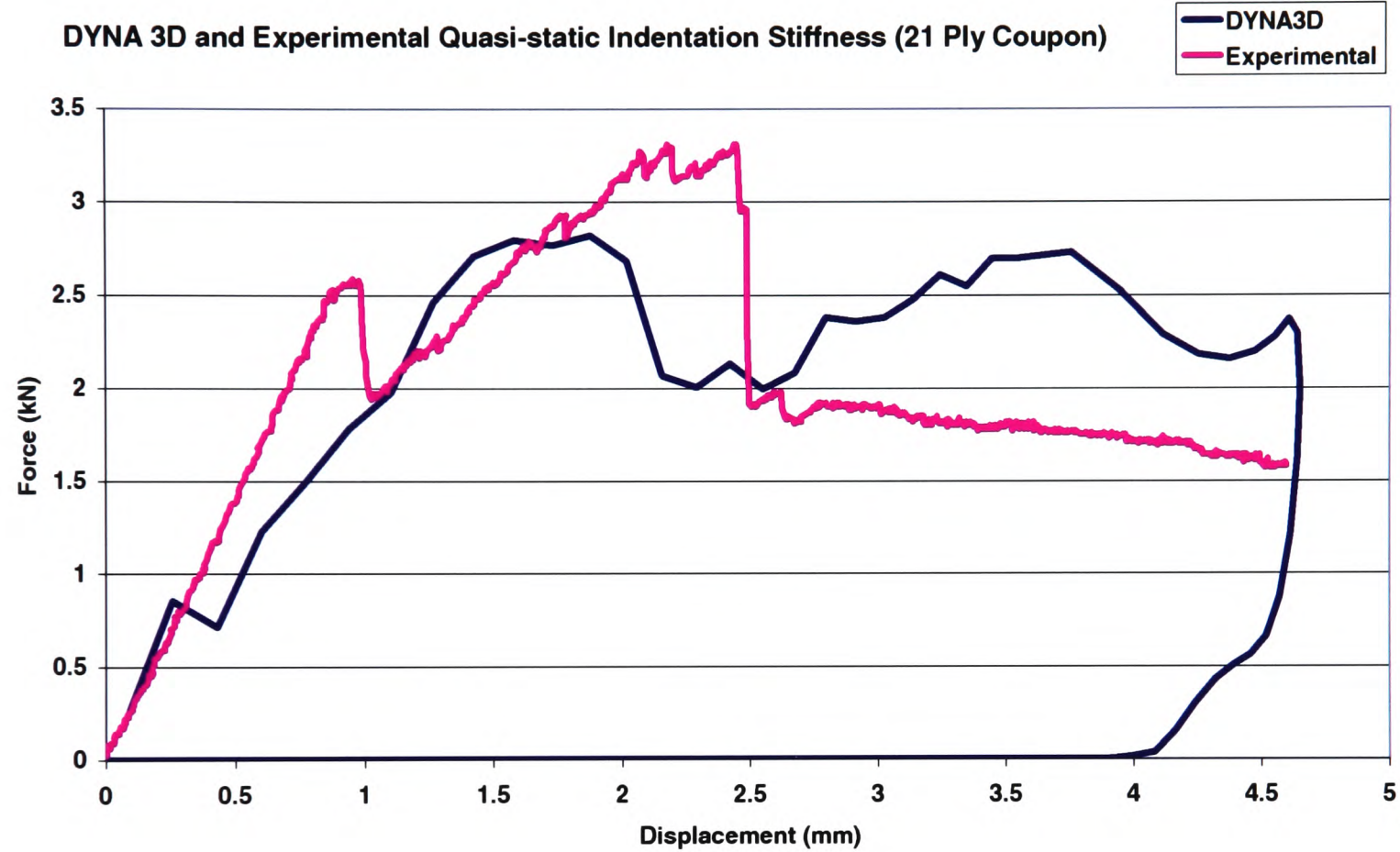
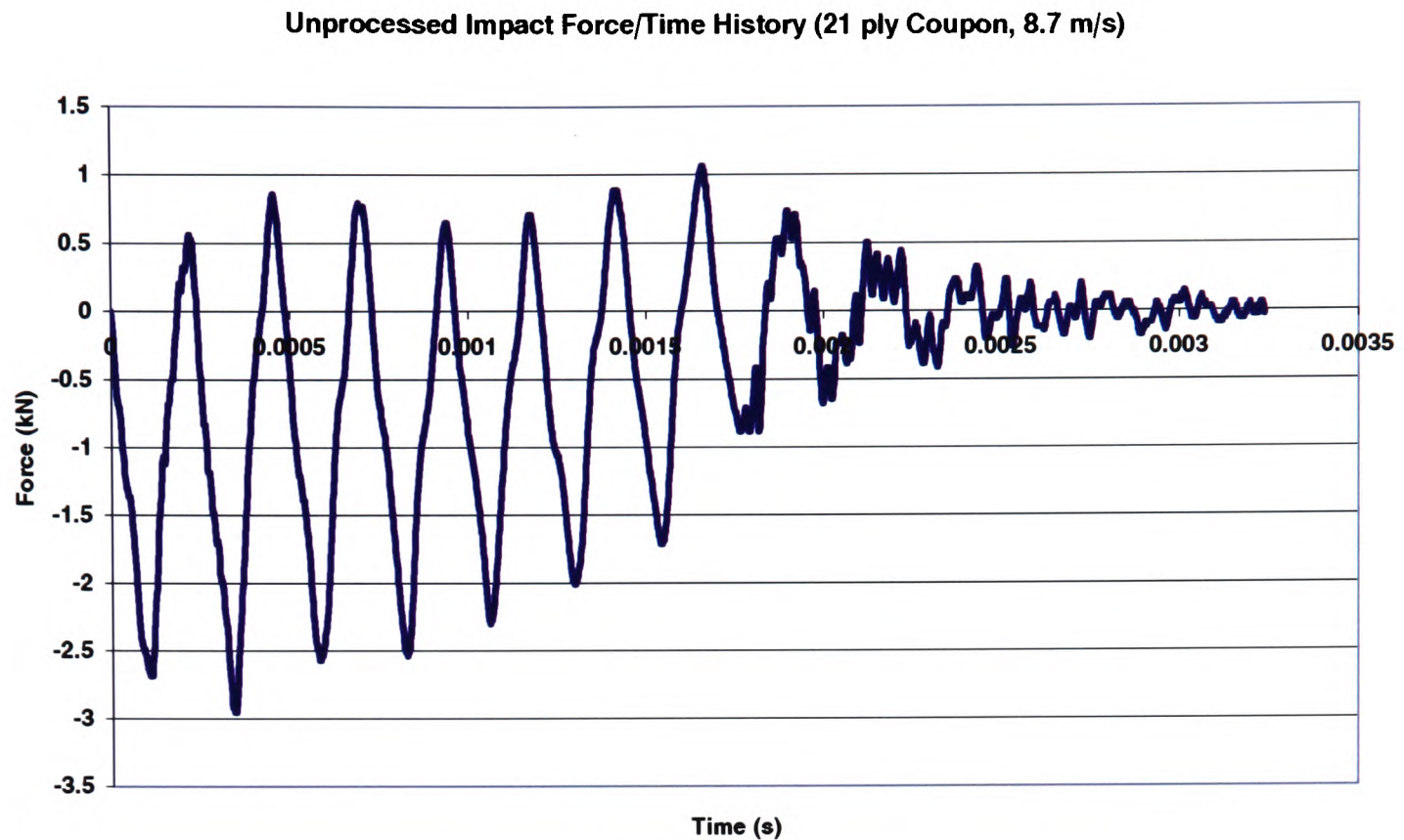


Figure 4.4

4.2.3 Force /Time History

The force/time history obtained from the DYNA 3D simulation may be compared with the experimental force/time history obtained from strain gauge measurements on the impactor. Fig. 4.5 shows the force/time history for an 8.7 m/s (9.7 J) impact as measured on a 21 ply coupon.

The force/time history measured by the impactor strain gauges contains large oscillations. These are largely due to the reflection of stress waves within the impactor itself. Because the strain gauges are mounted at the mid-point of the impactor they measure the instantaneous strain there. To determine the force time/history at the impactor tip, that is the actual force applied to the coupon, it is necessary to process the signal as outlined below.



For this calculation it is assumed that there are perfect stress wave reflections from the rear of the impactor, which has a machined flat end. The stress at the mid-point of the bar is therefore a superposition of the actual force/time input at the impactor tip and its reflection from the impactor end. Compressive stress waves are reflected as tensile stress waves and therefore the superposition is of the tip force/time input and the same signal inverted and delayed by a time equal to the impact bar length divided by the stress wave velocity.

To extract the tip force/time history we therefore use a superposition of the unprocessed strain gauge signal with itself delayed by the reflection time. For the titanium bar the reflection time is given by:

$$t = \frac{l}{c} = \frac{0.5}{4880} = 102.5 \mu s \tag{4.5}$$

Fig. 4.6 shows the result. This is the processed force/time history which represents the force at the impactor tip applied to the coupon. There are still large oscillations in the signal but these are probably due to interactions between stress wave reflections within the impact tip, vibrations of the coupon itself, and the damage process within the coupon.

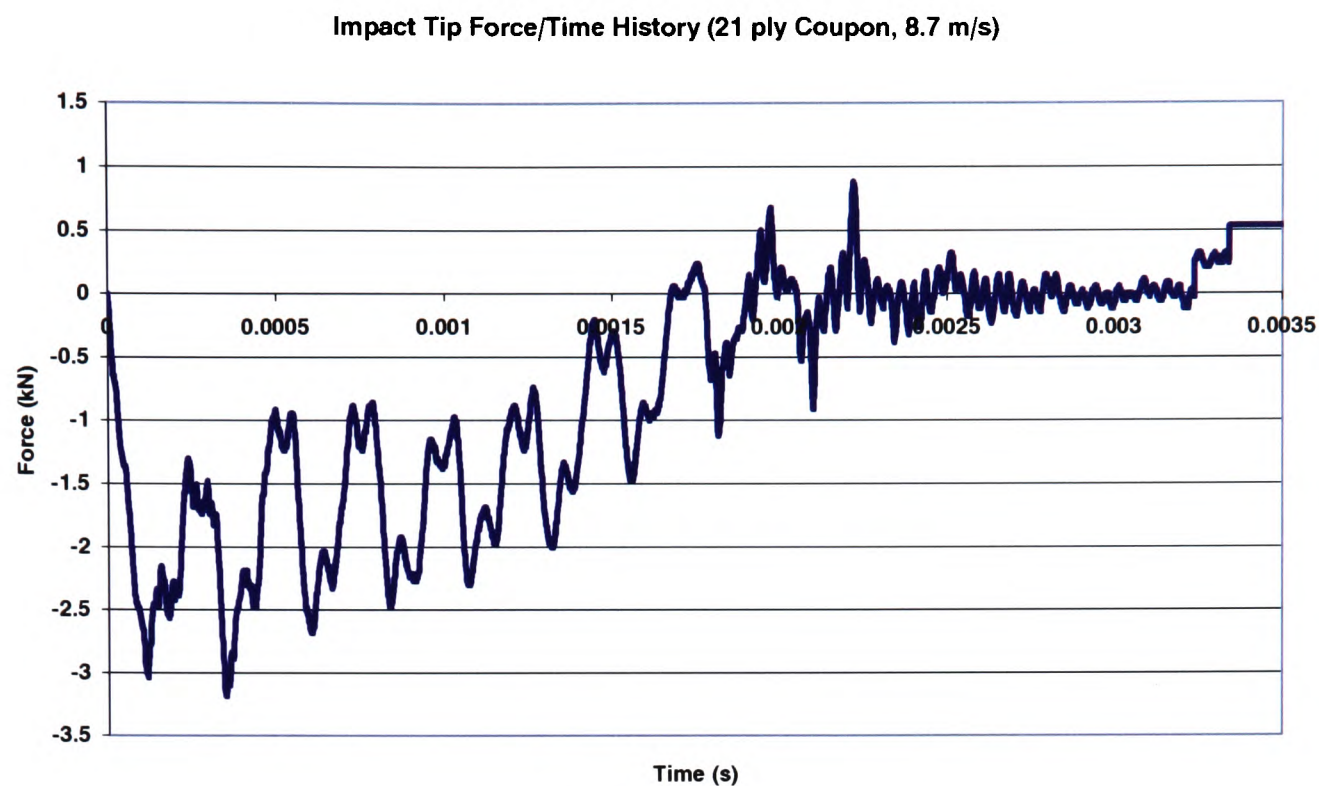


Figure 4.6

The processed force/time history obtained in Figure 4.6 is compared to the DYNA 3D simulation in Figure 4.7. It can be seen that the simulation only very approximately matches the experimental result. Because the impactor is modelled as fully rigid we cannot expect the simulation to produce any oscillation resulting from the natural response of the impactor tip. However, we would expect the simulation to correctly produce the approximate magnitude of the force/time history. The DYNA 3D simulation does do this, although the decay in contact force on rebound occurs slightly earlier than the experimental observation.

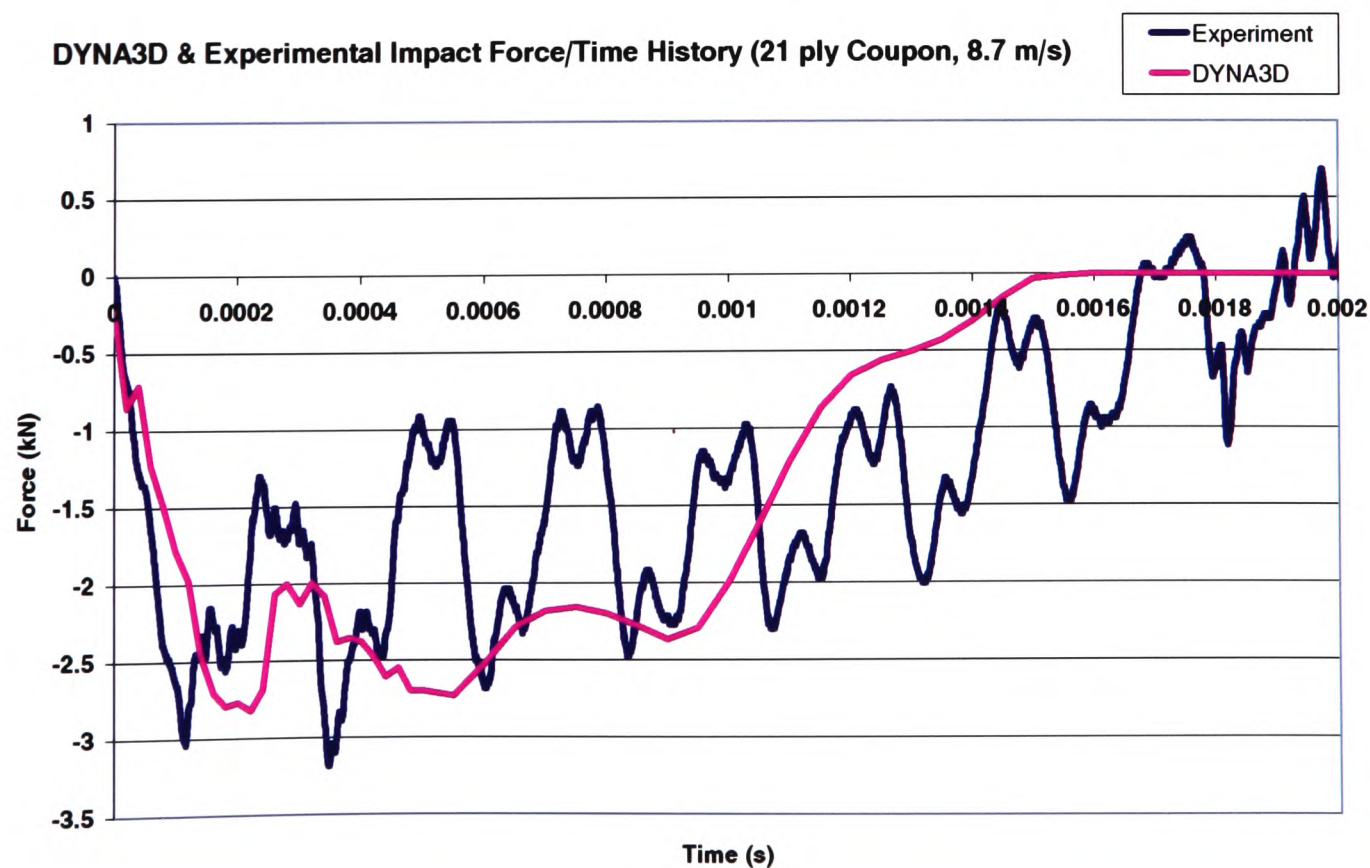


Figure 4.7

4.2.4 Damage

Figures 4.8 to 4.11 show the initial development of the four damage modes in the simulation of low speed impact. The images show the corner of the one quarter model which represents the central impact site in a real coupon. In each image the impactor tip, coupon, and support can be seen from top to bottom. In each case elements in the coupon which have failed in fibre fracture, matrix cracking, matrix crushing, or delamination modes are shown as having changed colour. In all of the images the predicted deformed shape is shown to scale.

Figures 4.9 and 4.11 show that the predicted dominant failure mode is delamination, with matrix cracking playing an important role. Figures 4.8 and 4.10 show that there is very little matrix crushing predicted and that fibre fracture occurs late in the damage development, at an impactor displacement of approximately 0.75 mm.

Deformation is controlled most significantly by the delamination failure mode. Elements which have failed in shear are given zero shear stiffness, and because of the domination of this failure mode in the prediction, the local deformation is dominated by the shearing of failed elements.

The distribution of damage may be compared with that seen in Coupon C (Figure 2.13). It is clear that delamination is the dominant damage mode in impacts of this energy. However the delamination distribution predicted by DYNA 3D is too localised. There is also too little fibre fracture predicted by the DYNA 3D model. Figures 2.14 and 2.15 show that even at low impact energies there will be some fibre fracture on the back face. In Figure 2.13 it can be seen that at higher impact energies this fibre fracture will propagate through the thickness of the coupon at the impact site.

Because of the major influence of the delamination mode on the coupon stiffness in the DYNA 3D simulation the model fails to predict fibre fracture correctly.

DYNA 3D simulation of low speed impact on 21 ply coupon
Fibre Fracture

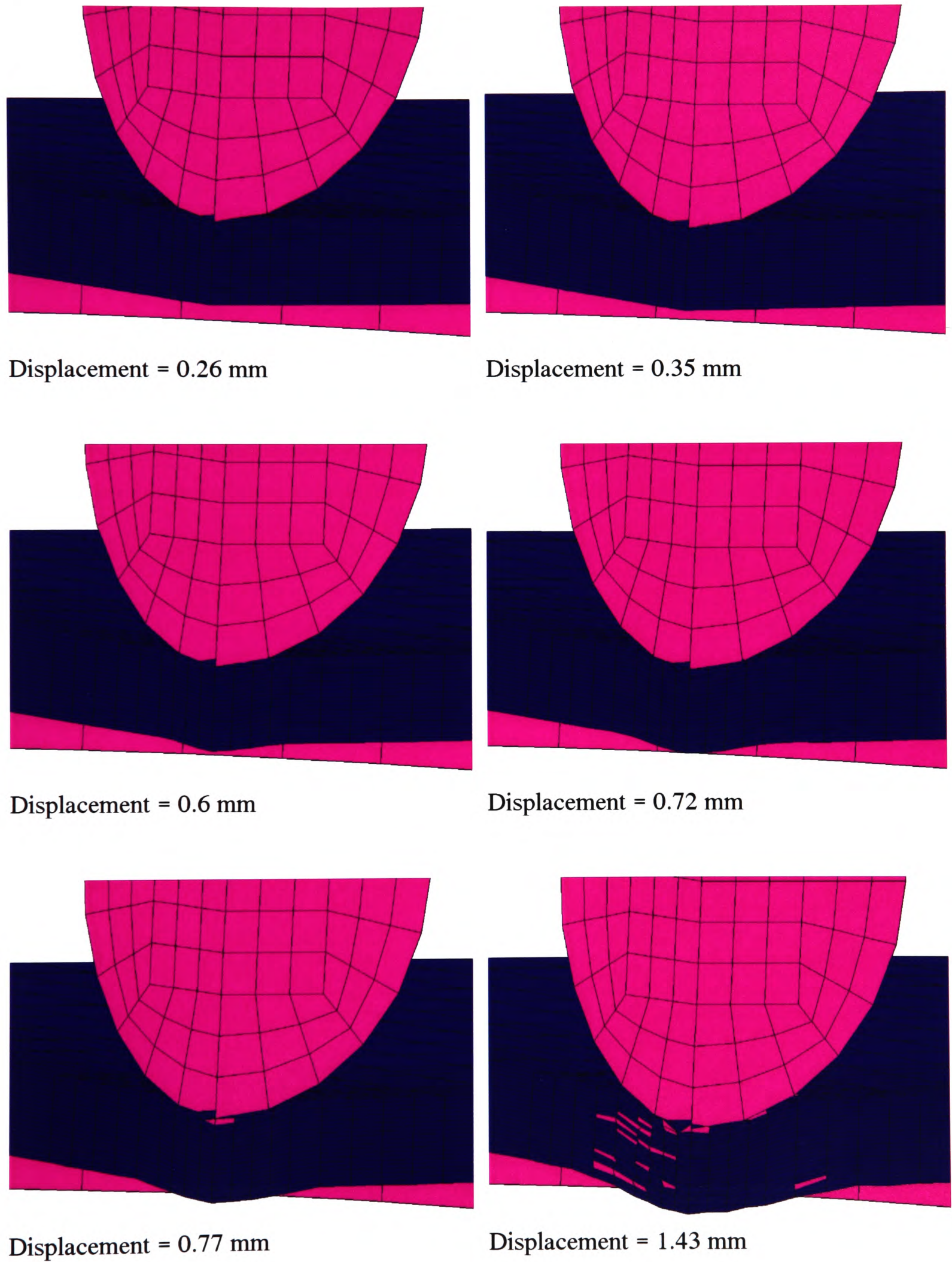


Figure 4.8

DYNA 3D simulation of low speed impact on 21 ply coupon
Matrix Cracking

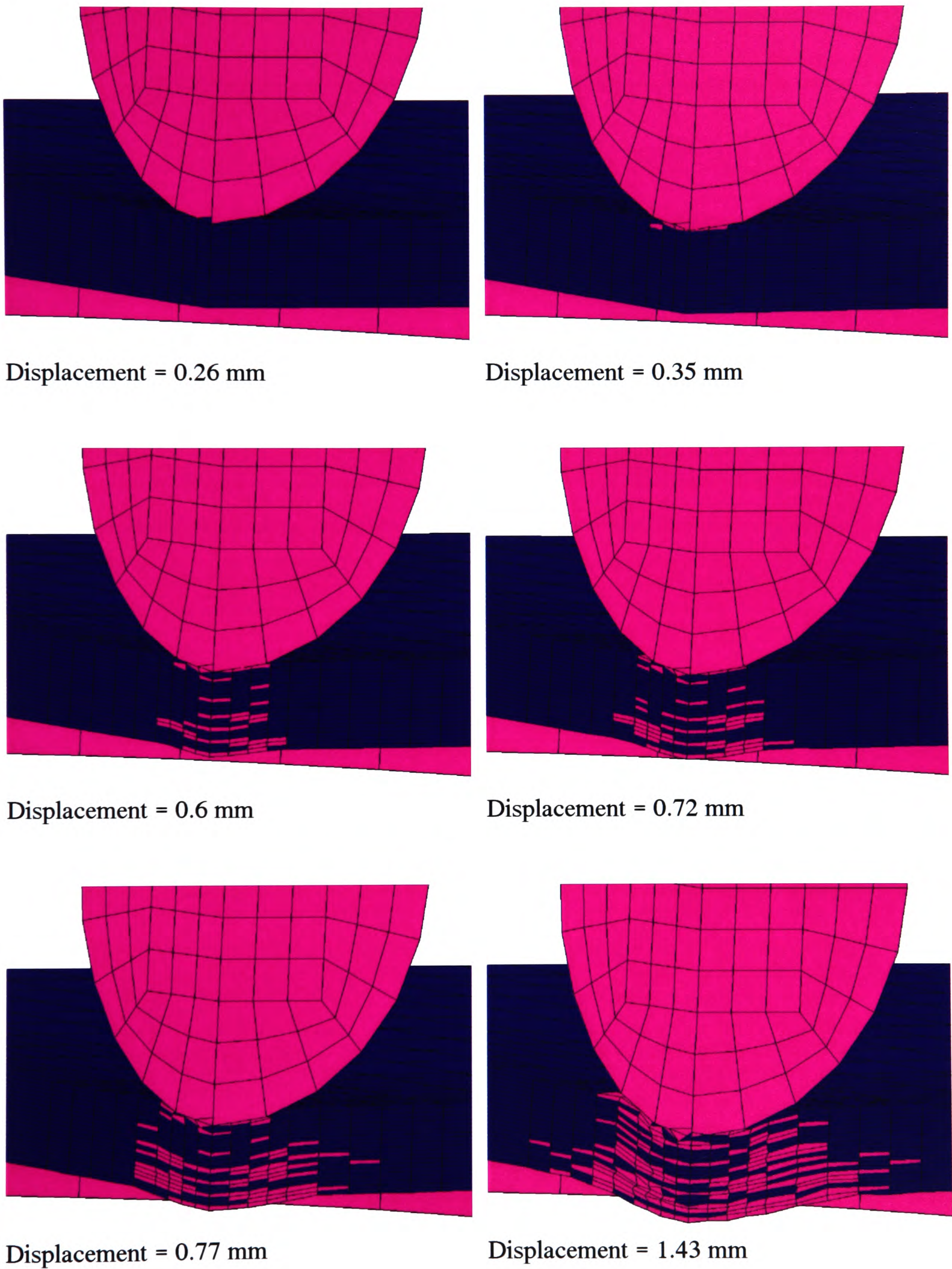


Figure 4.9

DYNA 3D simulation of low speed impact on 21 ply coupon
Matrix Crushing

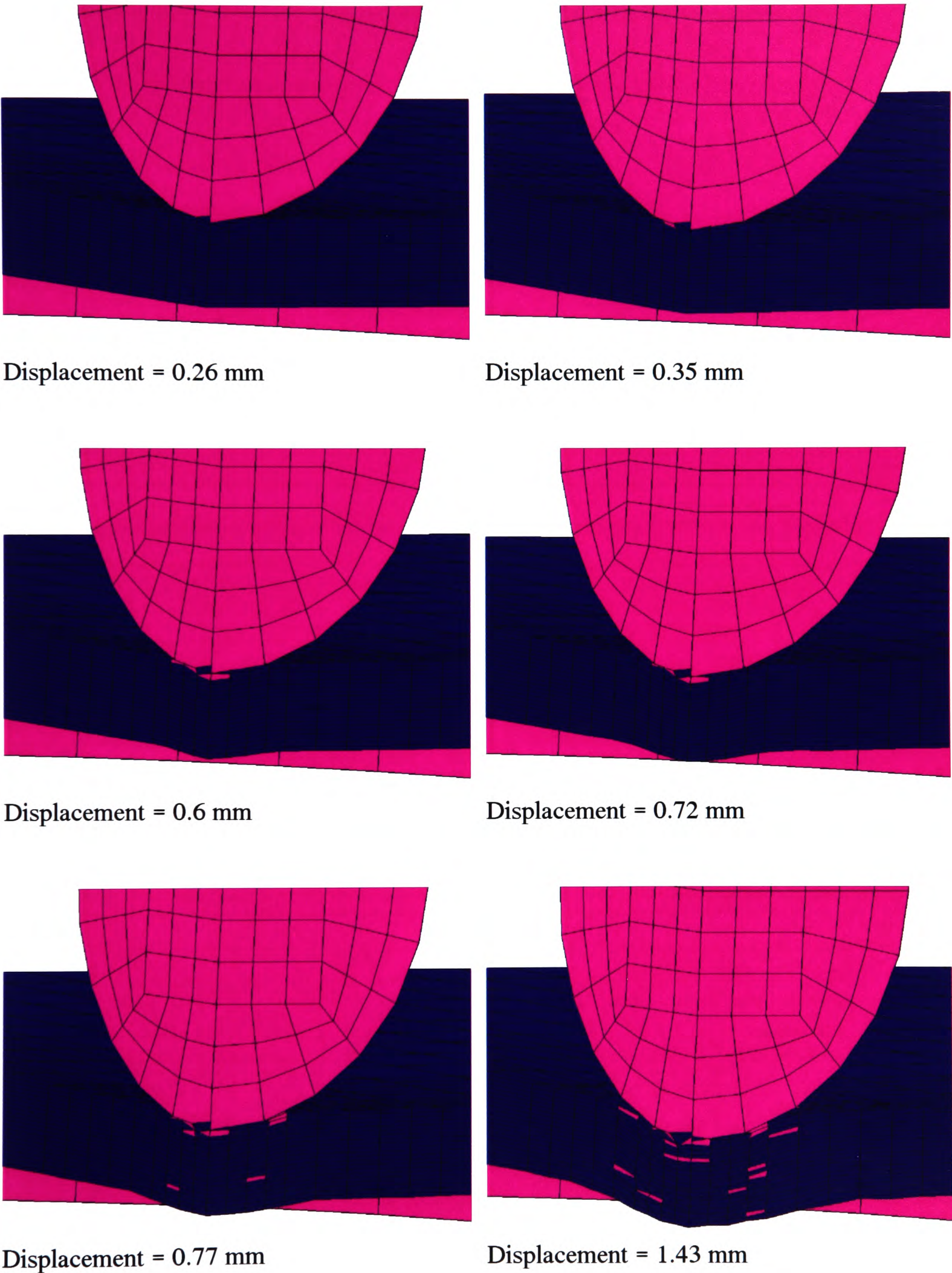
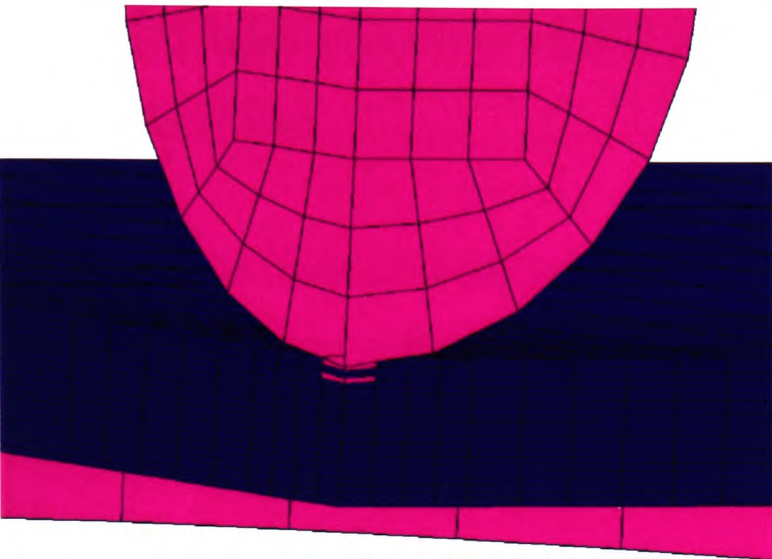
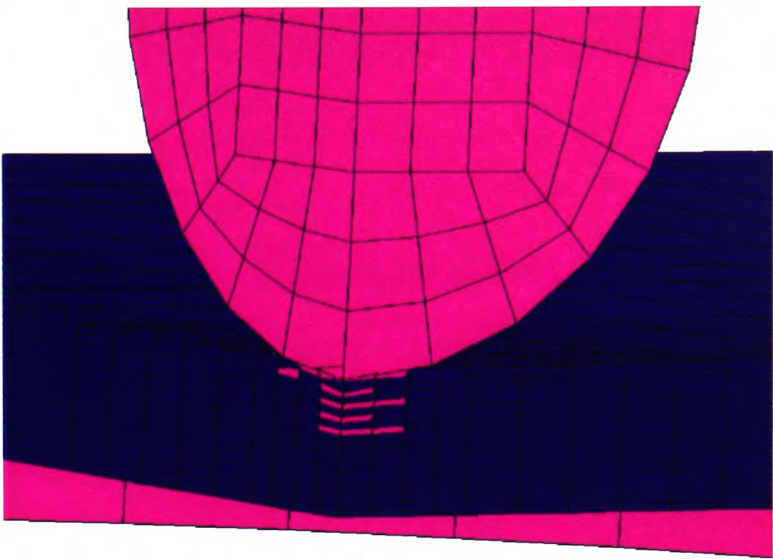


Figure 4.10

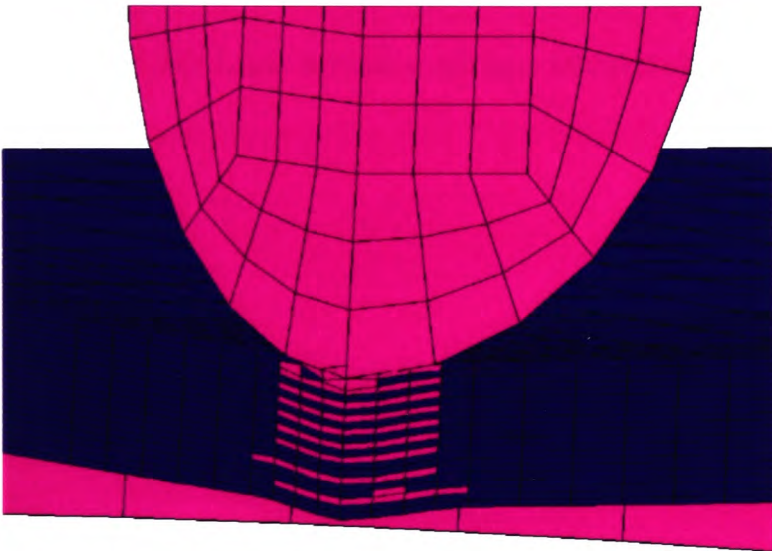
DYNA 3D simulation of low speed impact on 21 ply coupon
Delamination



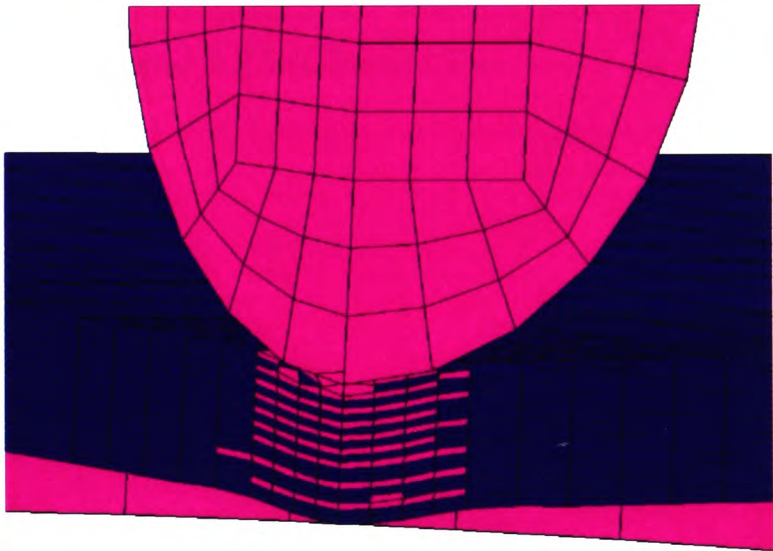
Displacement = 0.26 mm



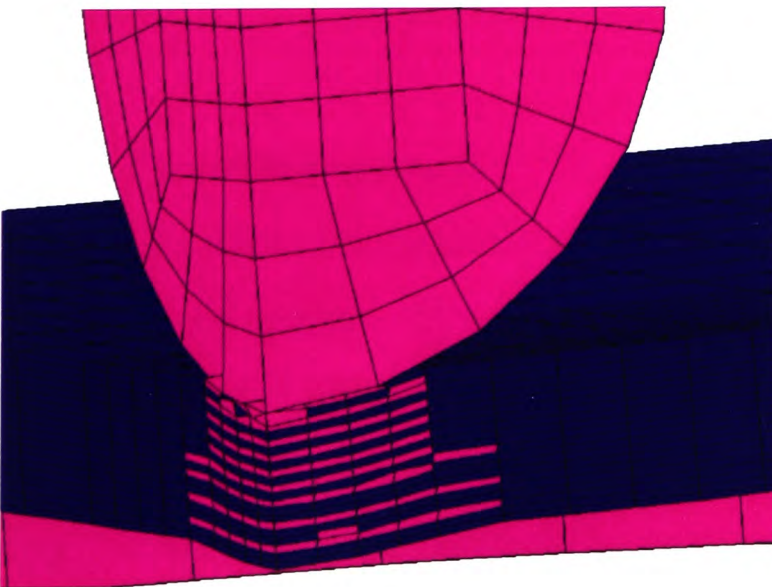
Displacement = 0.35 mm



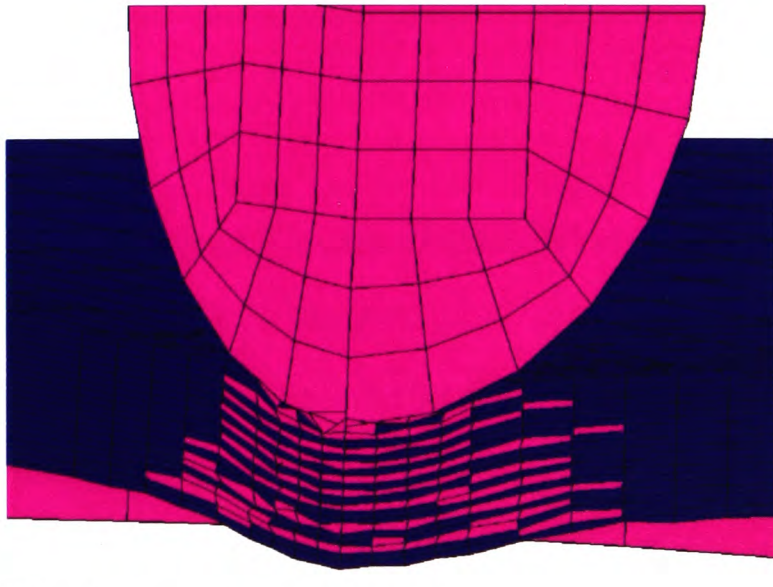
Displacement = 0.6 mm



Displacement = 0.72 mm



Displacement = 0.77 mm



Displacement = 1.43 mm

Figure 4.11

4.3 High Speed Impact

4.3.1 Projectile

Figure 4.12 shows the complete model used for the DYNA 3D simulation of high speed impact on a 21 ply coupon.

The coupon support is modelled identically to that used for the quasi-static indentation simulation. The coupon itself differs only in that the mesh density under the point of impact has been increased slightly to account for the smaller contact area of the ball bearing projectile.

The projectile itself is a quarter sphere, 6 mm in diameter, modelled as a rigid material with the density of steel (7950 kg/m^3). This gives it a mass equivalent to 0.9 g. The initial velocity of the projectile was 86 m/s. This is a kinetic energy of 3.3 J and the model was intended to simulate the high speed damage shown in Figures 2.24 and 2.25 (Chapter 2).

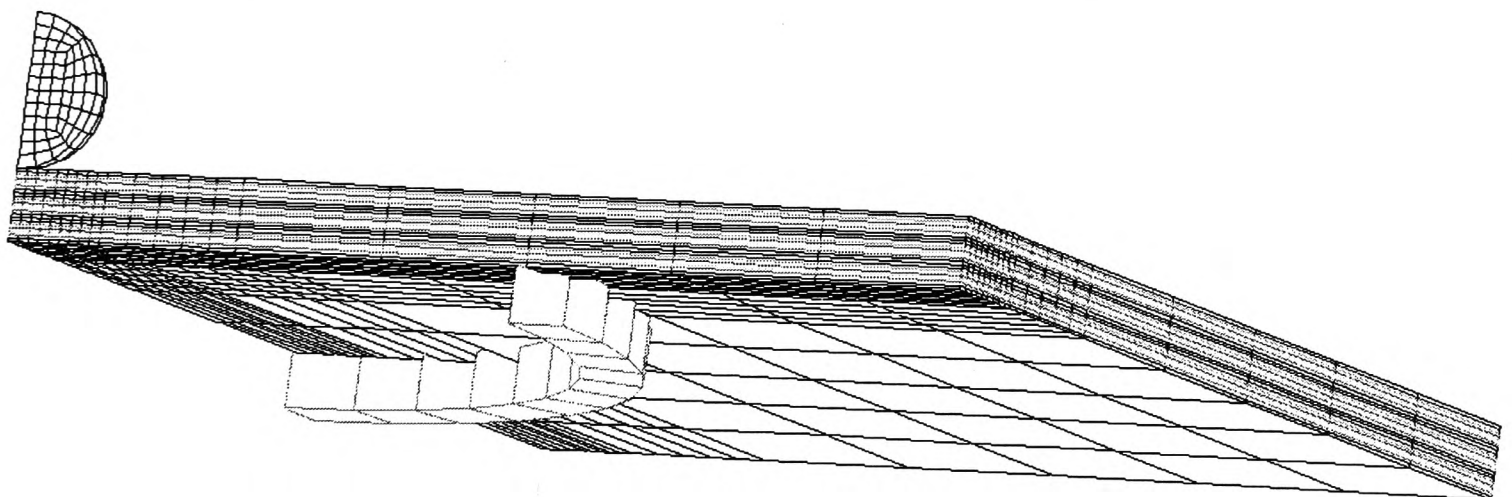


Figure 4.12

4.3.2 Force/Displacement

Figure 4.13 shows the displacement against time of the projectile in the DYNA 3D simulation of high speed impact. The coupon decelerates and rebounds the projectile although with significant absorption of energy. Energy is dissipated in the failed elements and observation of the coupon deformation throughout the impact event reveals that the recovered strain energy is stored in the structural deformation of the plate. The displacement due to the structural deformation is small compared to the local indentation associated with the failed elements. The local indentation is not recovered after the projectile rebound.

Figure 4.14 shows the force/time history for the high speed impact. Peak forces predicted are comparable with those obtained in experimental quasi-static indentation. Although inertial (dynamic) effects are significant at this impact velocity the area of fracture is very local and the peak force is governed by the failure process.

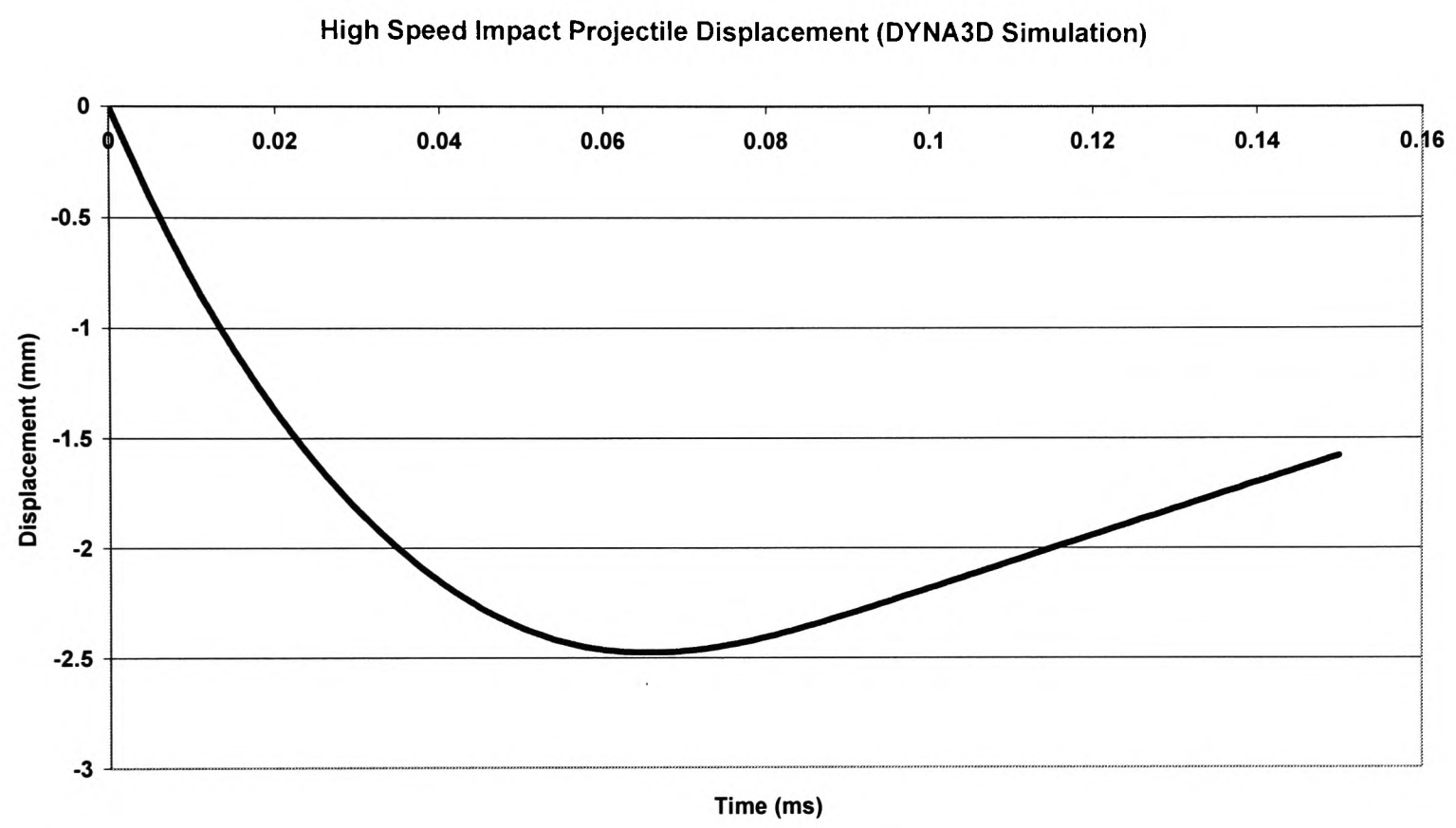


Figure 4.13

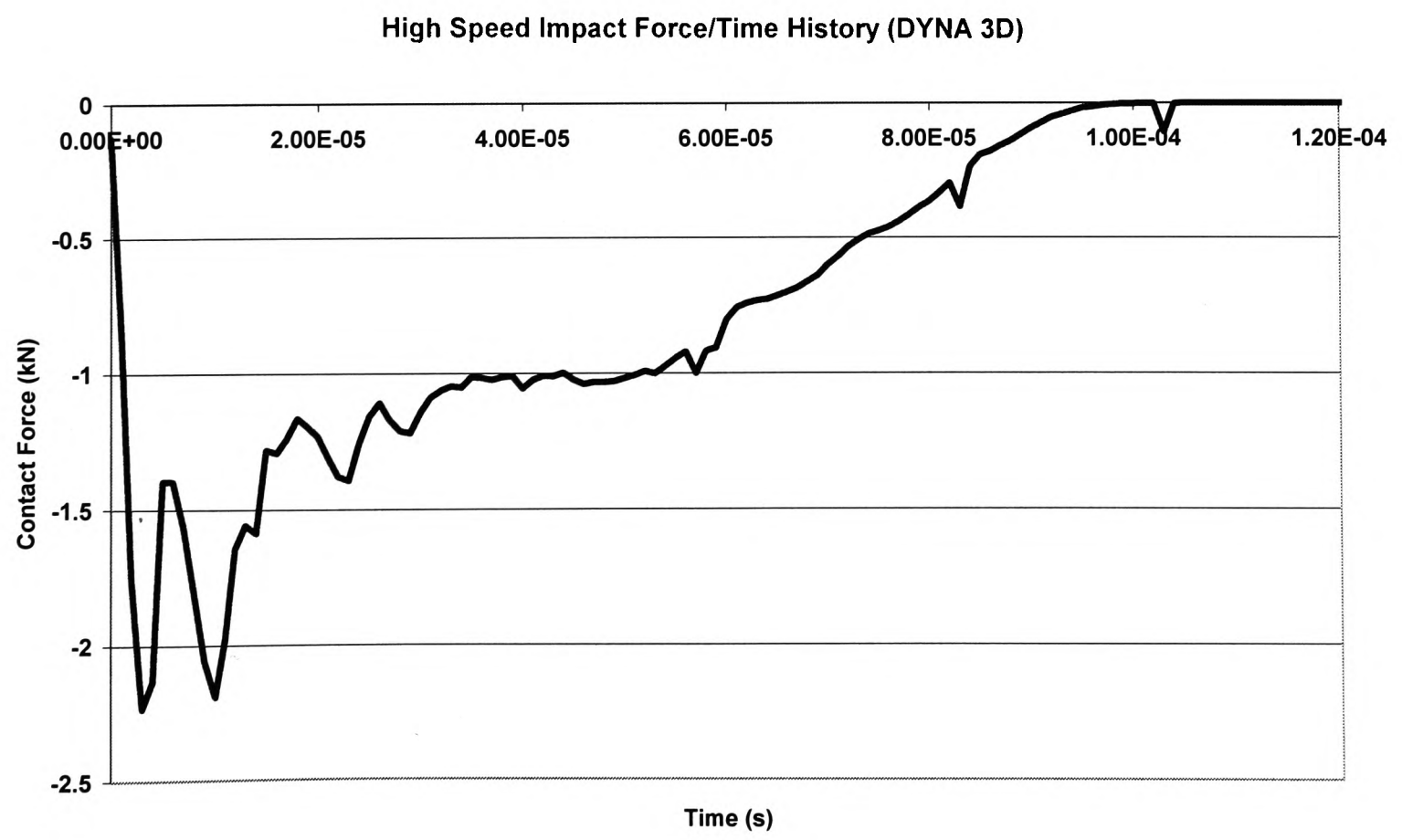


Figure 4.14

4.3.3 Damage

Figures 4.15 to 4.18 show the development of damage in the simulation of high speed impact. As in the low speed simulation the dominant failure mode is delamination with matrix cracking playing a significant role as well. Little fibre fracture or matrix crushing is predicted.

The delamination distribution is rather different to that predicted for low speed impact. In the low speed simulation, delamination is predicted for alternate plies while in the high speed model failure is predicted for every ply. This concentrated distribution of delamination failure throughout the thickness leads to very large, and unrealistic, local deformation.

Examination of the experimental C-scan (Figure 2.25) for this energy of high speed impact shows that we would expect a significant level of delamination. However the microscope section (Figure 2.24) shows that otherwise the damage at this energy level is slight; there is certainly very little permanent indentation.

The weakness of the simulation again appears in the prediction of delamination which leads to very large local deformation and reduces the influence of structural bending which would lead to tensile fibre fracture.

4.4 Conclusions

In general we can conclude that although the material model 22 in DYNA 3D has potential for predicting the location and type of damage expected in impact on CFRP structures it is currently very poor at predicting the post-failure deformation and damage progression.

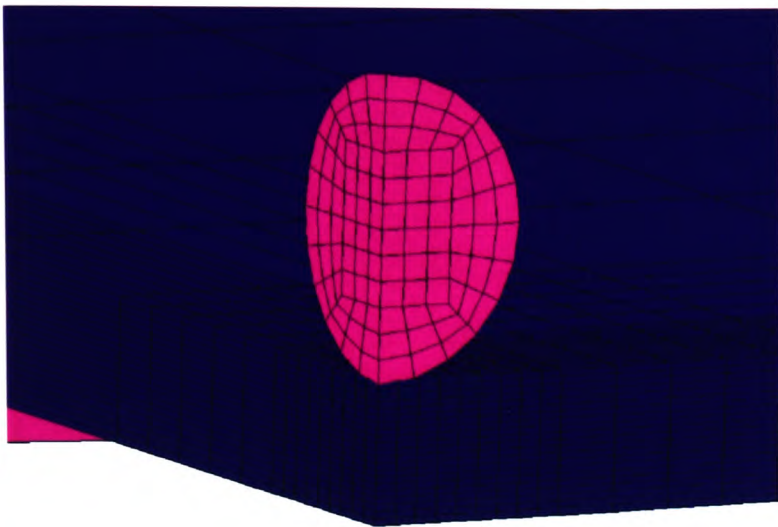
A significant problem appears to be that the delamination failure mode gives zero shear stiffness to the whole element, and therefore the whole thickness of a ply that an element represents. This is unrealistic and leads to an unrealistically low shear stiffness for the damaged laminate. A desirable alternative would be for the failure mode to create a delamination crack between plies by separating the elements, creating new nodes, and remeshing. However this process would be complex to implement and would require much longer computation times.

The low speed impact simulation predicts first failure at a lower load than the experimental measurement. The first failure mode was delamination, which is controlled by the shear

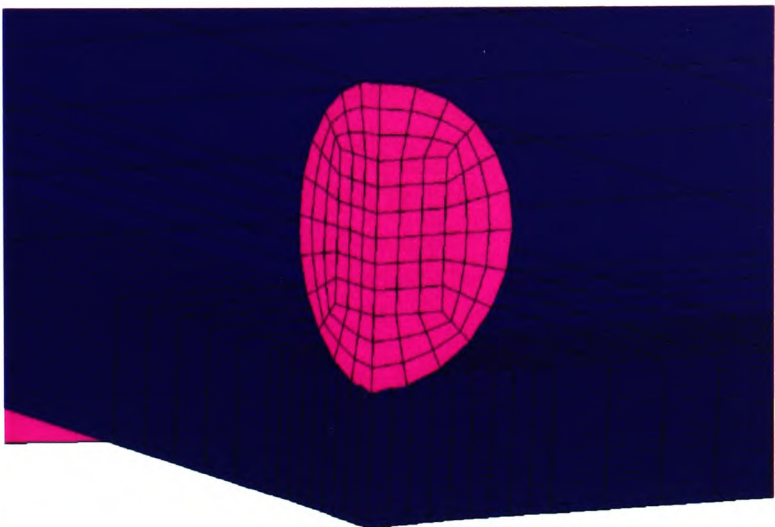
strength. It would appear therefore that the shear strength value used here for modelling a laminated plate model is too low. This demonstrates the difficulty of applying small specimen test results to structural models.

At present the use of DYNA 3D with material model 22 to predict impact damage behaviour in CFRP cannot be recommended.

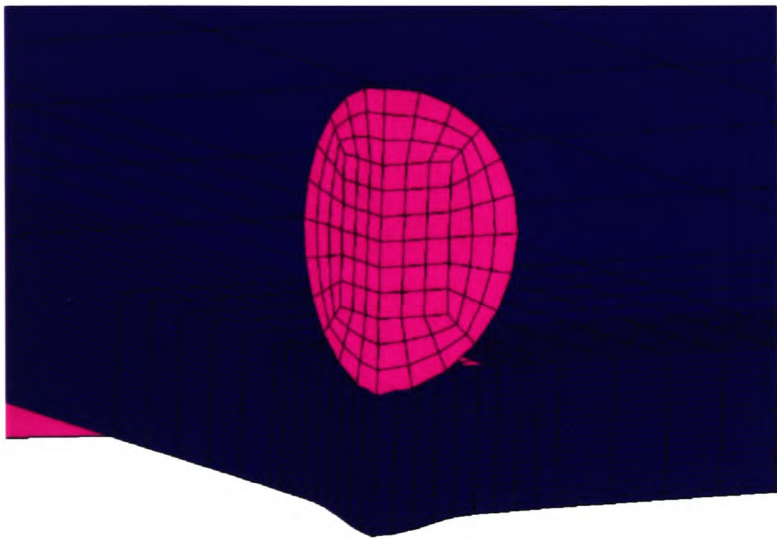
DYNA 3D simulation of high speed impact on 21 ply coupon (86 m/s, 0.9 g, 3.3 J)
Fibre Fracture



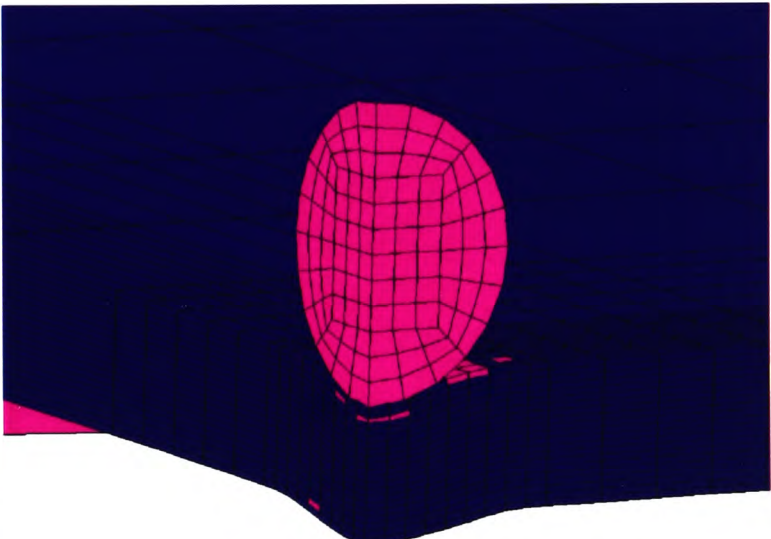
Time = 1 μ s Indentation = 0.086 mm



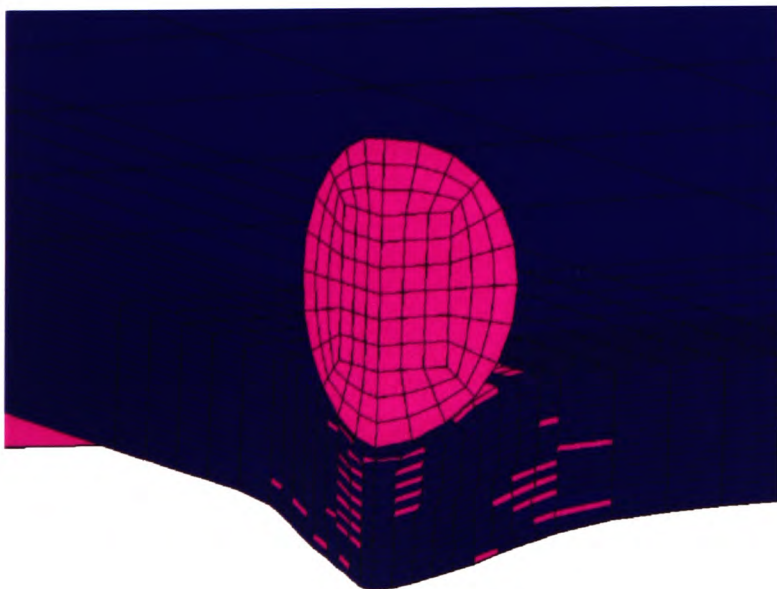
Time = 3 μ s Indentation = 0.255 mm



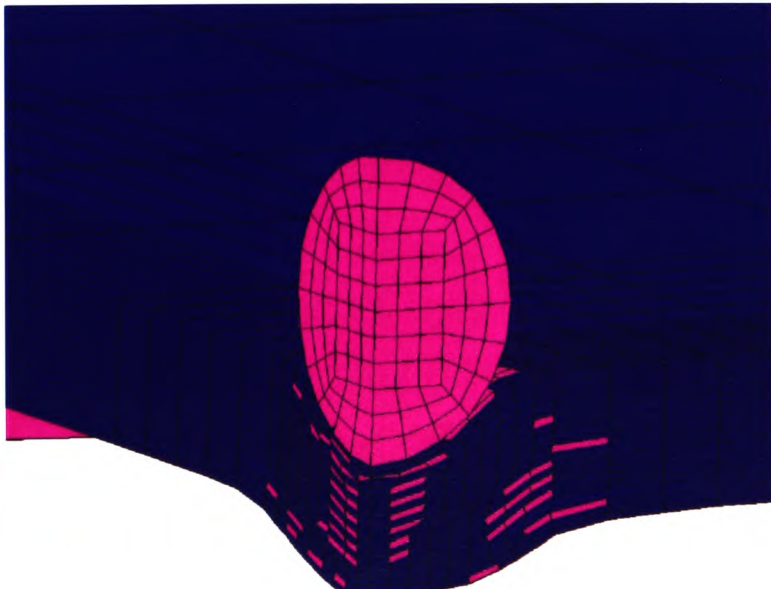
Time = 5 μ s Indentation = 0.416 mm



Time = 9 μ s Indentation = 0.715 mm



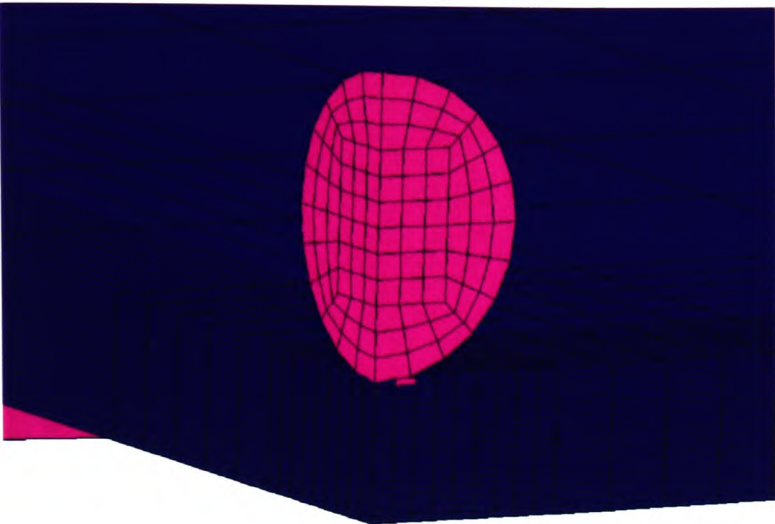
Time = 19 μ s Indentation = 1.329 mm



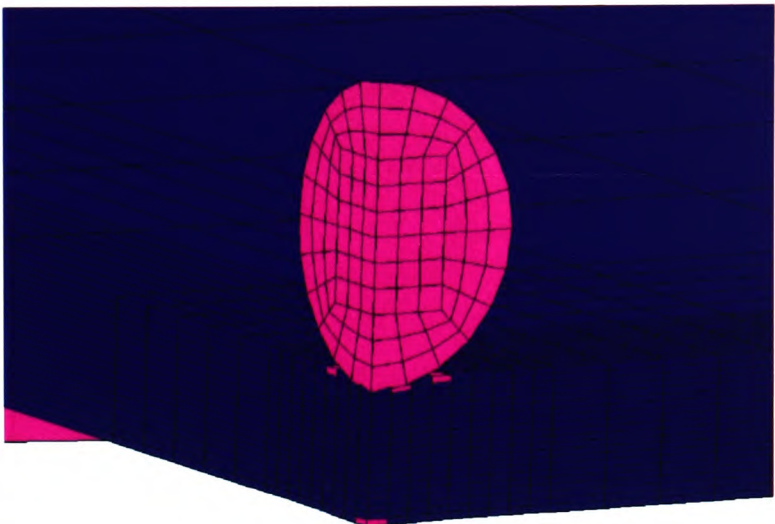
Time = 29 μ s Indentation = 1.793 mm

Figure 4.15

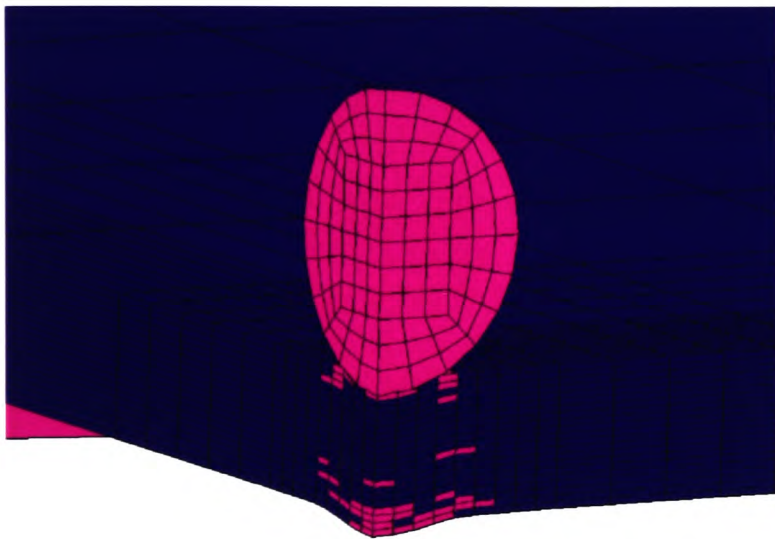
DYNA 3D simulation of high speed impact on 21 ply coupon (86 m/s, 0.9 g, 3.3 J)
Matrix Cracking



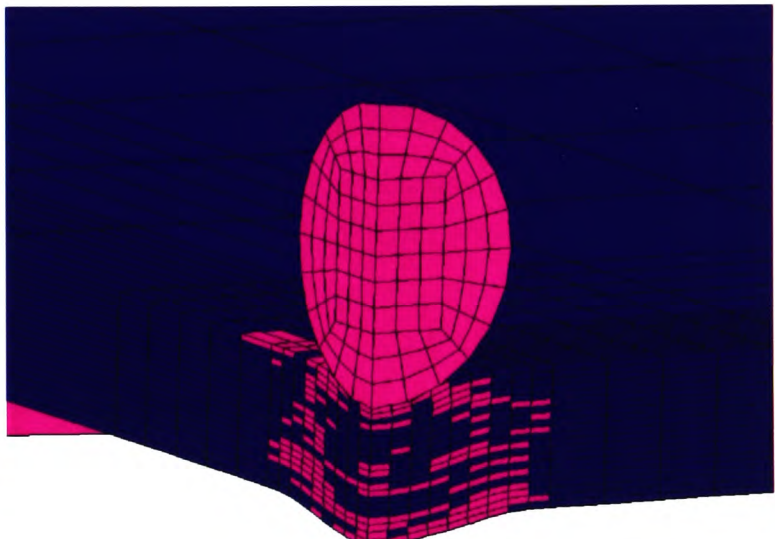
Time = 1 μ s Indentation = 0.086 mm



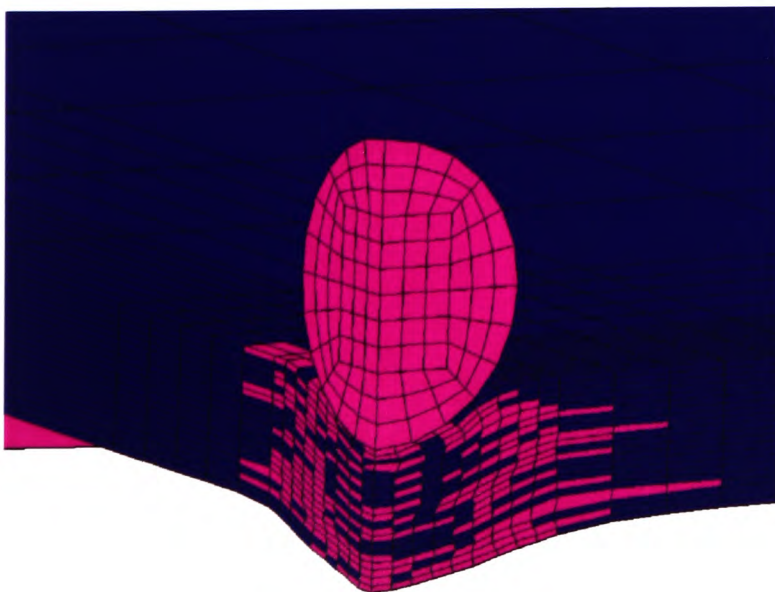
Time = 3 μ s Indentation = 0.255 mm



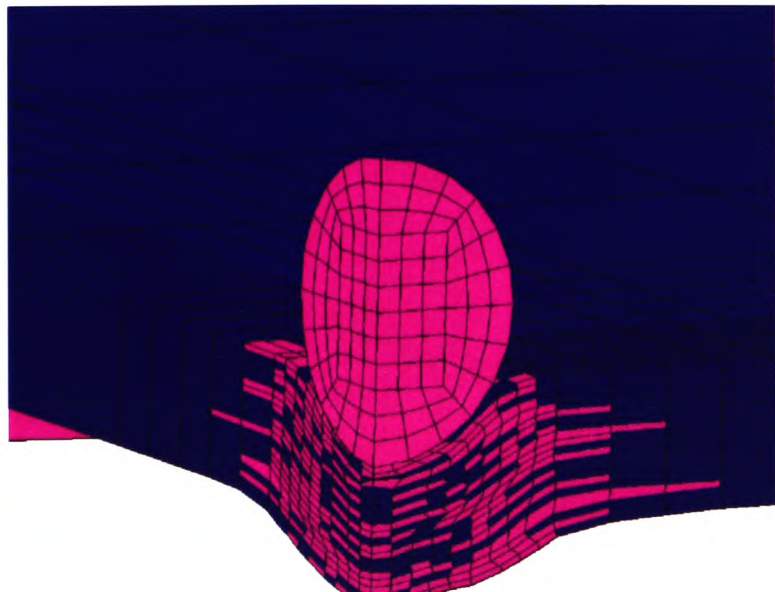
Time = 5 μ s Indentation = 0.416 mm



Time = 9 μ s Indentation = 0.715 mm



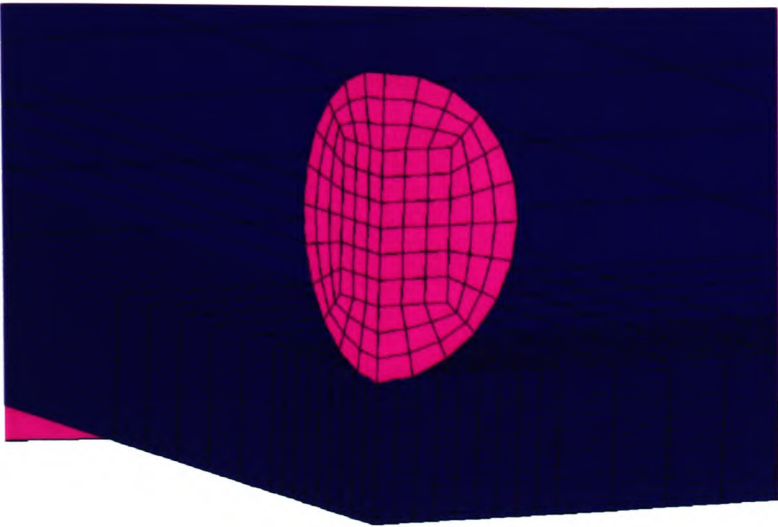
Time = 19 μ s Indentation = 1.329 mm



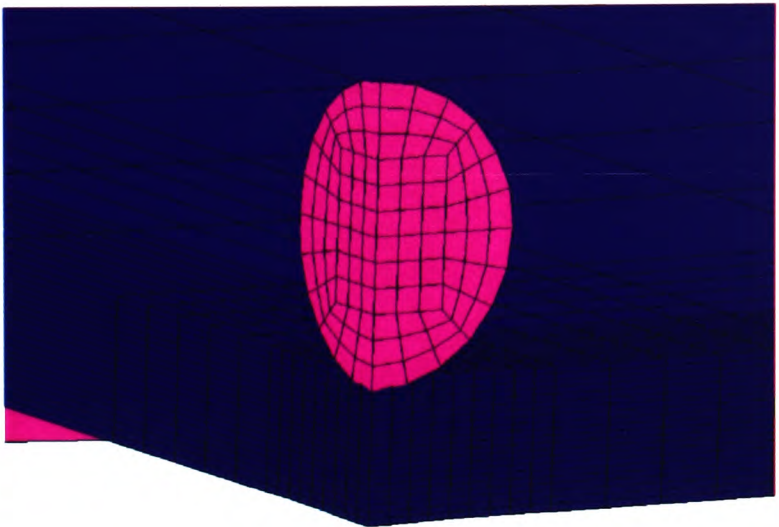
Time = 29 μ s Indentation = 1.793 mm

Figure 4.16

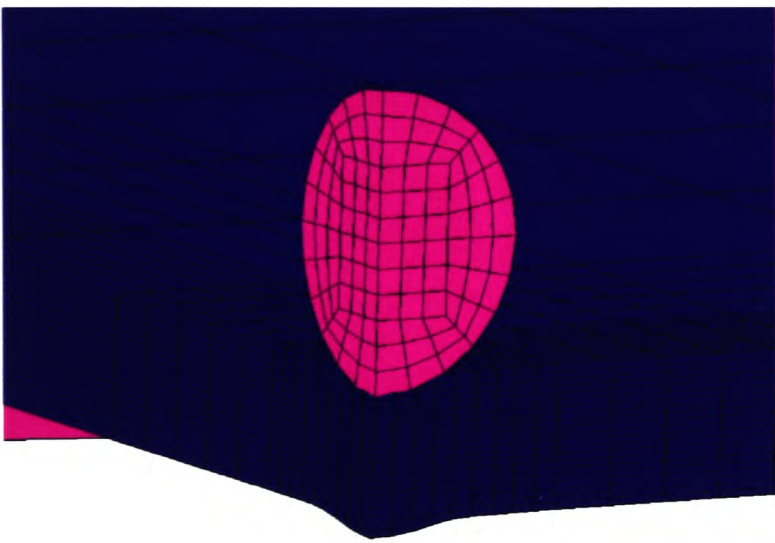
DYNA 3D simulation of high speed impact on 21 ply coupon (86 m/s, 0.9 g, 3.3 J)
Matrix Crushing



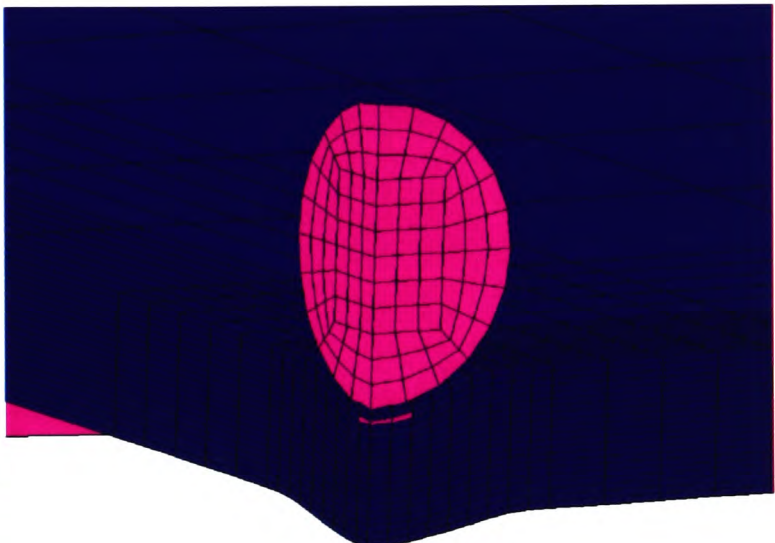
Time = 1 μ s Indentation = 0.086 mm



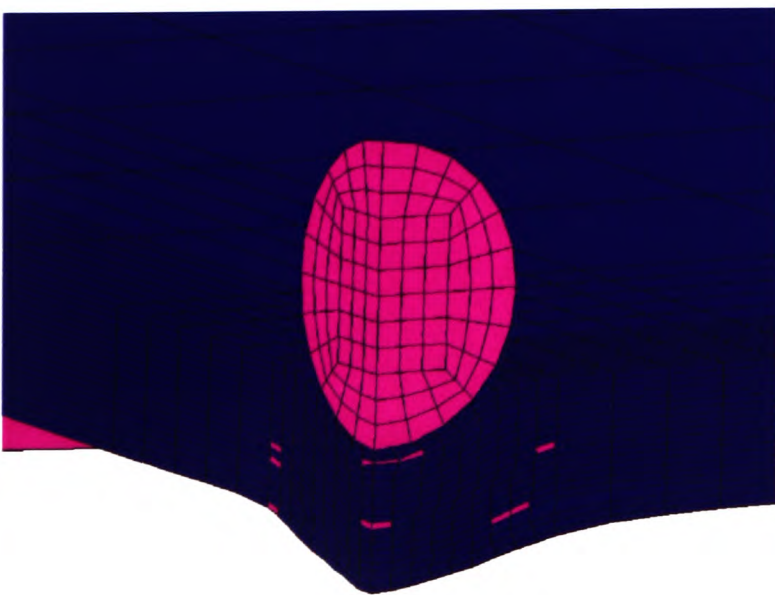
Time = 3 μ s Indentation = 0.255 mm



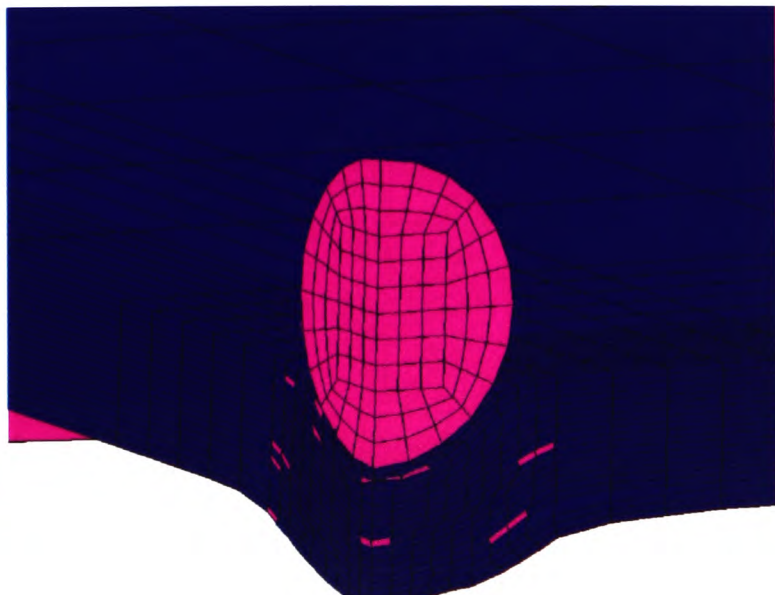
Time = 5 μ s Indentation = 0.416 mm



Time = 9 μ s Indentation = 0.715 mm



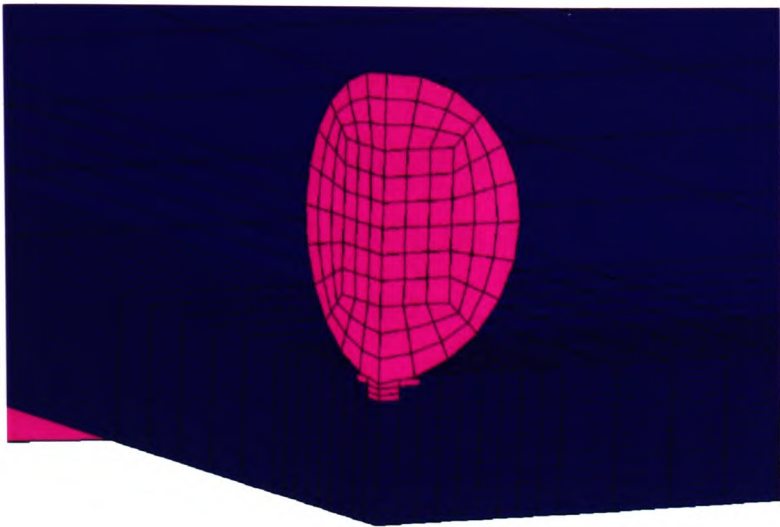
Time = 19 μ s Indentation = 1.329 mm



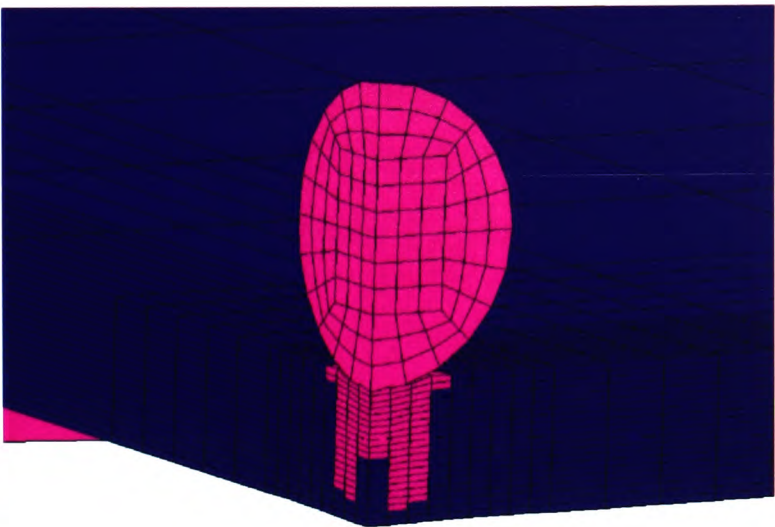
Time = 29 μ s Indentation = 1.793 mm

Figure 4.17

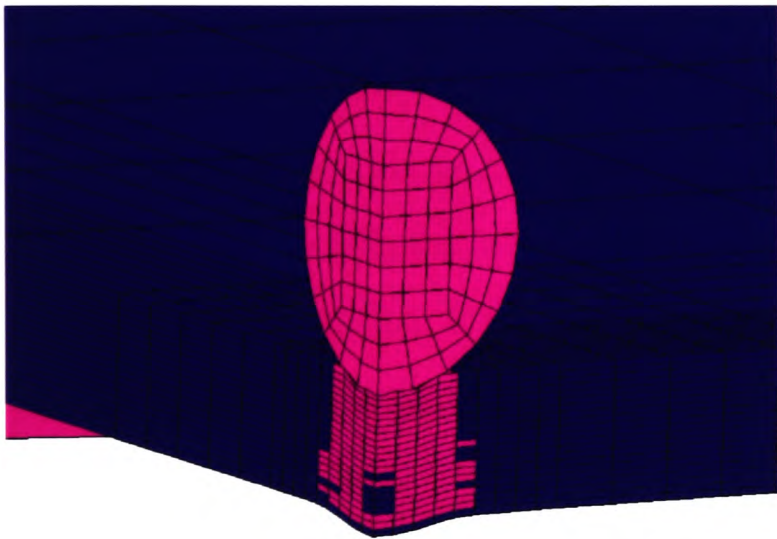
DYNA 3D simulation of high speed impact on 21 ply coupon (86 m/s, 0.9 g, 3.3 J)
Delamination



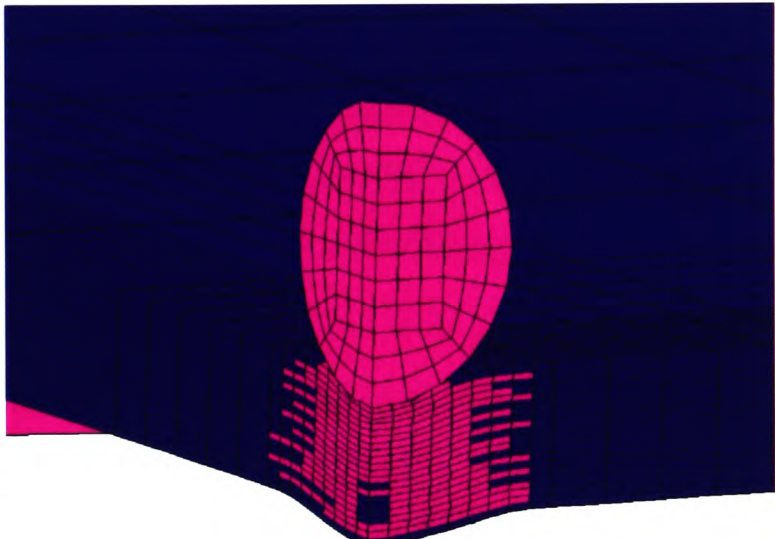
Time = 1 μ s Indentation = 0.086 mm



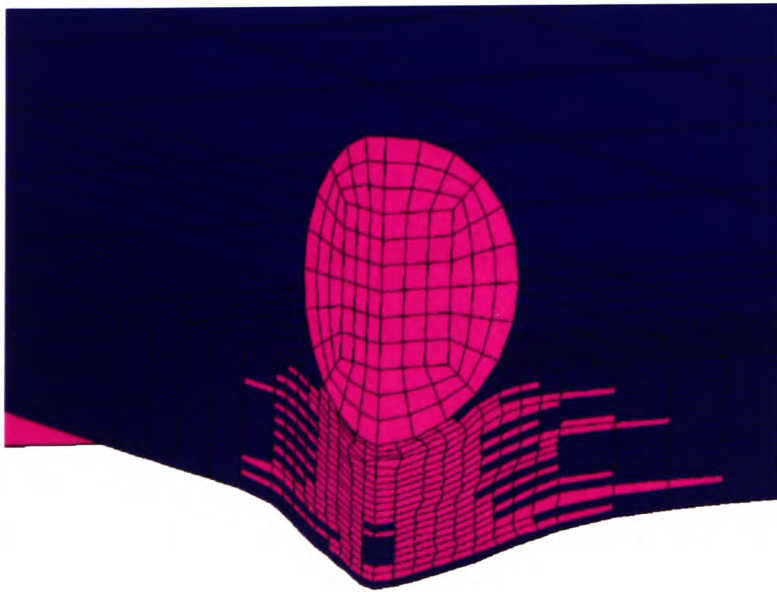
Time = 3 μ s Indentation = 0.255 mm



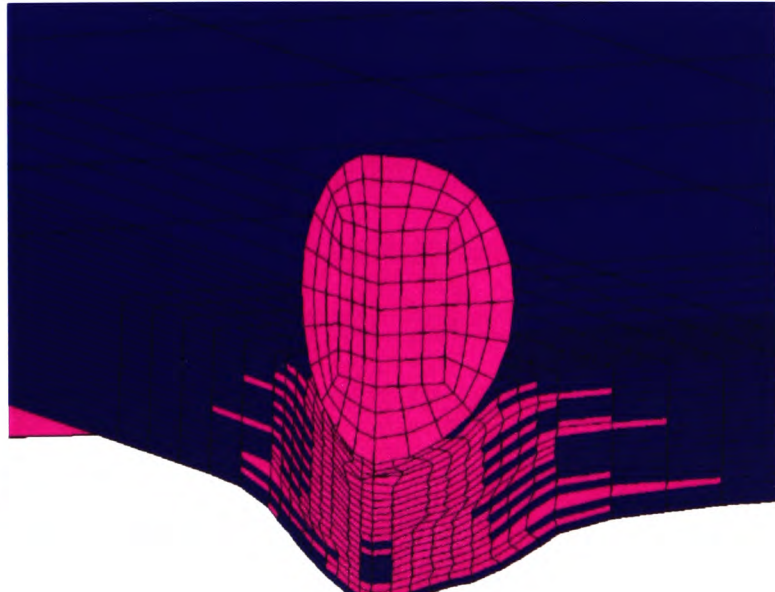
Time = 5 μ s Indentation = 0.416 mm



Time = 9 μ s Indentation = 0.715 mm



Time = 19 μ s Indentation = 1.329 mm



Time = 29 μ s Indentation = 1.793 mm

Figure 4.18

Chapter 5

Effect of Impact Damage on Residual Strength

In order to assess the effect of impact damage on residual strength, tests were conducted in tension and compression on both undamaged and impact damaged CFRP. The tests conducted and the results obtained are described in this chapter.

5.1 Design of Test Rigs

5.1.1 Industry Standard Tests

Industry standard tests are used for materials selection and quality control. For composites three tests are usually used: tension tests; compression tests; and compression after impact (CAI) tests. Unfortunately there has been little standardisation in testing methods to date.

In general compression testing methods seek to evaluate only a material property and not a structural one. Usually the required failure mechanism is either microbuckling of the composite fibres or a shear failure. Structural macrobuckling is to be avoided. This means that the test specimen must be stocky (wide and with a short unsupported length) so that the bending effects of macrobuckling are minimised and the compressive stress distribution in the section is as uniform as possible. Failure at the specimen ends will occur due to stress concentrations unless the design of the specimen prevents this. Tapered or tabbed specimens ensure that the stresses are highest in the gauge section and therefore failure will occur there.

Compression after impact (CAI) testing poses a problem. The specimen gauge section must be thin enough and have a large enough area to enclose the impact damaged zone. This invariably gives a relatively slender section which will have a tendency to macrobuckle. Industry practice in such tests is to prevent overall two-dimensional buckling of the test plate by providing restraints at the edges of the plate. Typical industry standard test methods are reviewed below.

Airbus Industrie Test Method 1.0010 [8]

The Airbus test is a CAI test for fibre reinforced plastics. The test coupon is a quasi-isotropic lay-up of approximate thickness 4 mm and size 150 x 100 mm. Compression testing is carried out through direct loading to the coupon ends with anti-buckling guides restraining the coupon

edges from out-of plane movement.

Boeing Advanced Composite Compression Tests [9]

The Boeing test specification is for compression tests of composite laminates with and without holes and impact damage.

The undamaged coupon is a tabbed specimen 12.7 mm wide, 80 mm long and with an unsupported gauge length of 4.8 mm.

The open-hole test coupon is untabbed and 38.1 x 305 mm with a central hole of 6.35 mm diameter. The test fixture for open hole compression has knurled grips which leave a gauge length of 102 mm over which lateral support against buckling is provided except for a window of 25.4 x 16.5 mm around the hole.

The CAI coupon is a plate 152 x 102 mm. For the CAI test the coupon is loaded directly on its ends with restraint to the edges provided by anti-buckling guides.

Royal Aerospace Establishment - CRAG Methods [10]

The CRAG methods cover: the measurement of tension and compression strength of an undamaged unidirectional composite coupon; a multi-directional laminate, undamaged or with central hole; and a compression after impact test.

The unidirectional specimen is a strip 110 x 10 mm with a thickness of 2 mm and tabs of light alloy or GRP (glass reinforced plastic) leaving a gauge length of 10 mm. The tabbed ends are gripped in a testing machine which can provide accurate alignment and rotational restraint through the end fixtures.

The multi-directional laminate coupon is 2-4 mm thick, 20-40 mm wide and with a gauge length of at least 100 mm so that no individual fibre in the angled plies runs under both of the end tabs. For testing of notch sensitivity a 5 mm diameter hole is drilled in the centre. Aluminium alloy or GRP end tabs of at least 40 mm length are used to grip the coupon. In the compression testing anti-buckling guides with PTFE to minimise frictional restraint are used.

Compression after impact tests use a 3 mm thick quasi-isotropic lay-up laminate coupon 180 x 50 mm. For testing, the coupon is clamped, leaving a gauge length of 100 mm. Anti-buckling guides along the edges leave a width of 40 mm unrestrained against out of plane displacement.

Defence Research Agency - Improved Test Method [35]

Curtis, Gates and Molyneux proposed an improved engineering test method for the measurement of the compressive strength of uni-directional composite materials. They found that their new waisted coupon gave 50% higher strengths than the CRAG coupon described above. This improvement is ascribed to failures occurring in the gauge length rather than at the end tabs. The new coupon differs from the CRAG specification by having a carefully machined, waisted gauge length of 8 mm with 16 uni-directional plies. The outer plies (machined away in the gauge length) are $\pm 45^\circ$ plies and act as tabs for gripping the coupon. The failure mechanism observed in the gauge length was fibre microbuckling in a kink-band (see section 5.6.2).

5.1.2 Undamaged Compression Strength Test

Pure compression tests to failure were conducted for this thesis on small, undamaged, 21 ply specimens. The tests were designed primarily to measure the Young's modulus of the material in compression. Failure strength and strain were also obtained although these values are a lower bound on the compression strength. Higher values may be obtainable with specifically designed, waisted, compression test coupons.

The specimens were 95 mm long, 27.5 mm wide, and with a thickness of 2.8 mm (21 plies). Loading was by direct end contact and the coupons were gripped firmly between aluminium plates held by clamping jaws. The unsupported gauge length was 20 mm. Strain was measured by two strain gauges mounted on opposite sides of the specimen.

The failure mode observed in these tests was either shear crushing at the specimen end or shear failure initiated at the edge of the aluminium clamping plate.

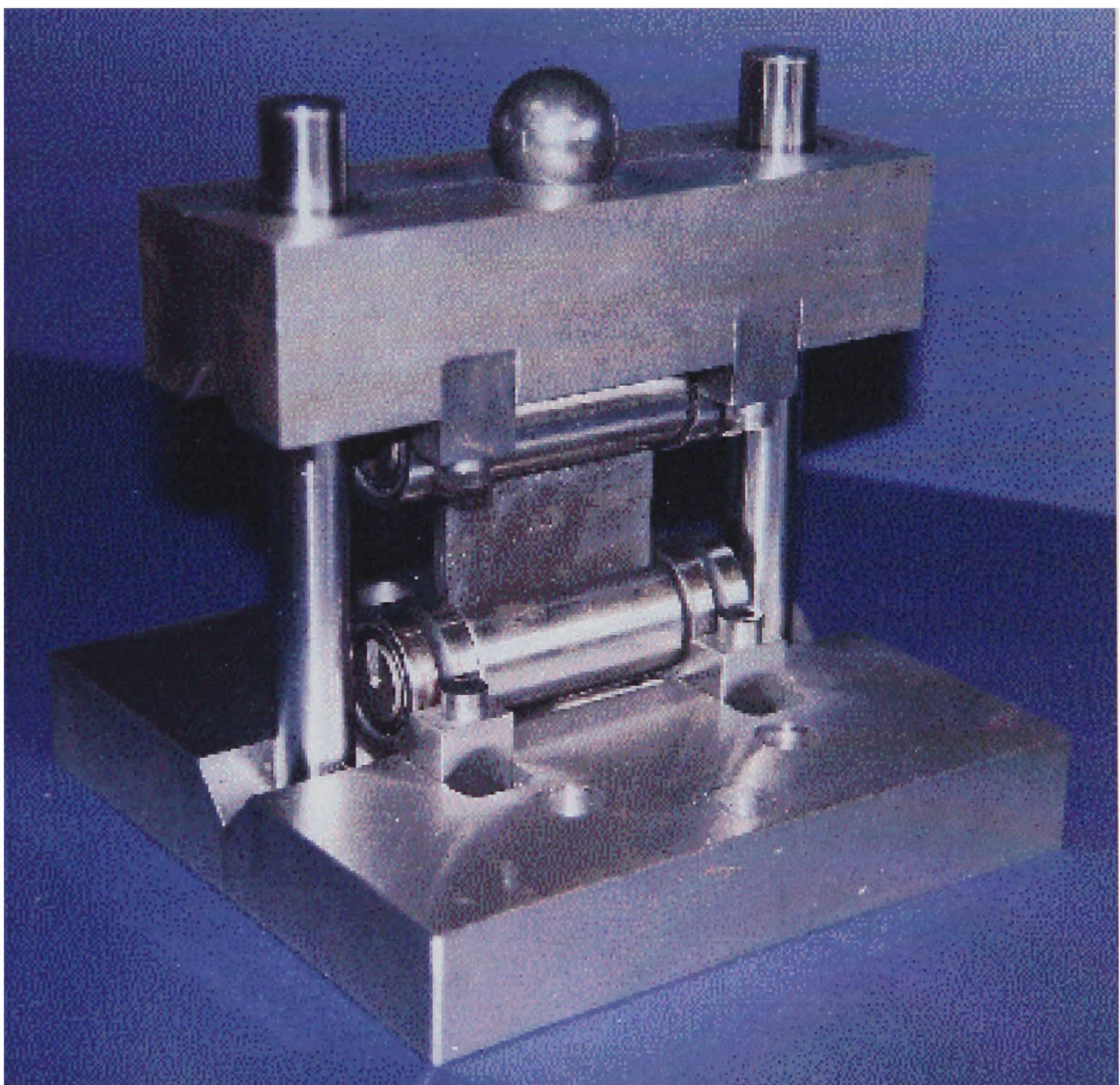
5.1.3 Pin Ended Compression Test Rig

For compression after impact tests, on the 29 ply coupons, a newly designed test fixture was used (Figure 5.1). The fixture was designed to allow pin-ended loading of the test coupons so

that failure would be initiated at the impact damaged area, and not at the loaded ends.

The 50 x 50 mm, 29 ply coupons were mounted in slotted bars on roller bearing supports. To ensure a good fit in the slots the coupon ends were packed with a fibrous, double-sided adhesive tape. The slotted bars were positioned so that they were fixed in position by the base and the loading bridge. The loading bridge was restrained to move only in a vertical plane by the use of two guide rails. Linear bearings were used to locate the loading bridge on the guide rails.

Load was applied to the loading bridge through a large ball bearing positioned in a socket on the loading bridge.



Pin Ended Compression Test Rig

Figure 5.1

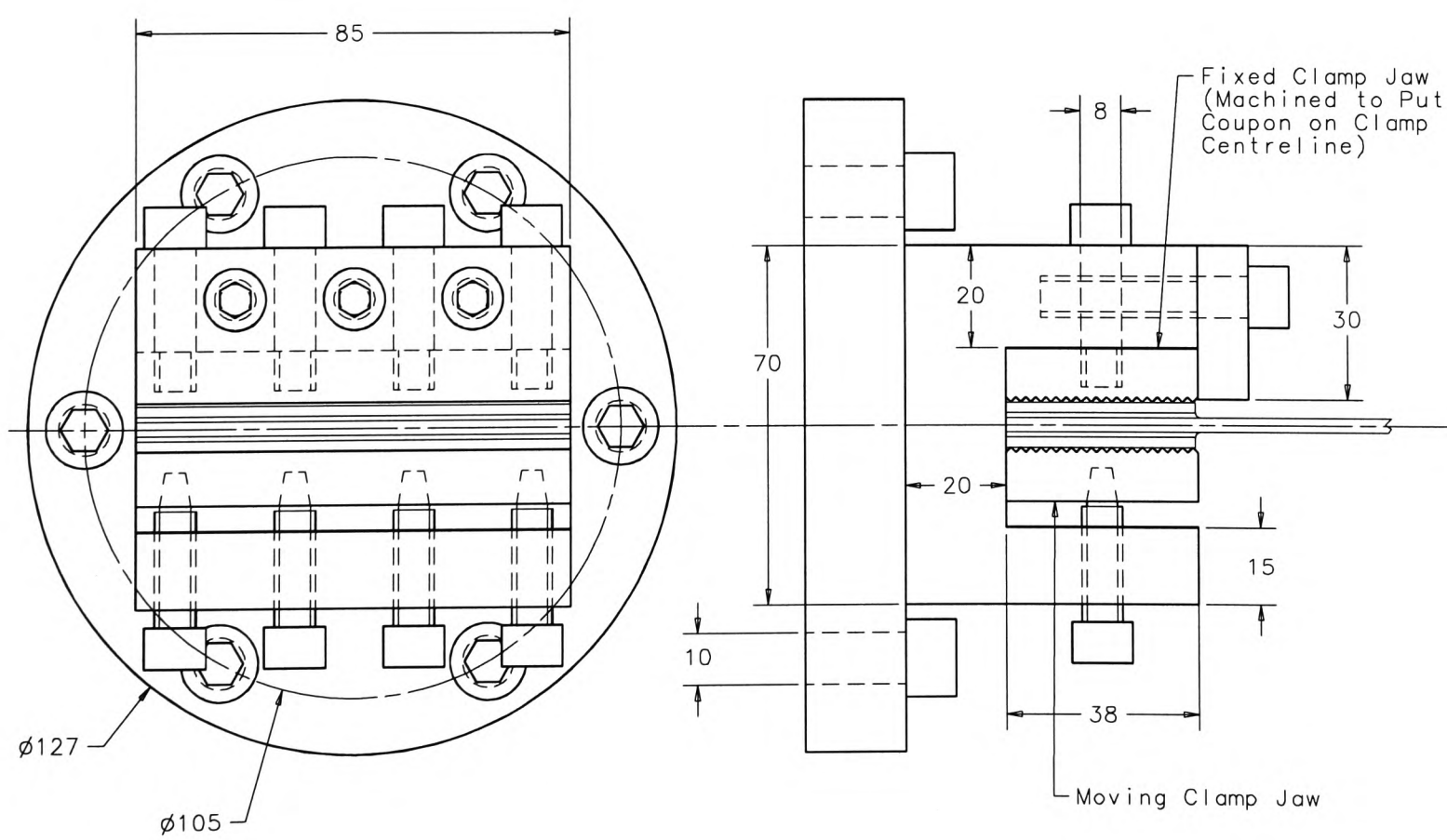
5.1.4 Tension/Compression/Fatigue Jaws

To allow tension, compression and fully reversed fatigue testing of the 140 x 85 mm, 21 ply coupons, a pair of jaws was designed and manufactured.

The jaws were designed to provide sufficient clamping force to the coupon aluminium tabs to prevent failure of the tab adhesive and to allow full load reversal without backlash.

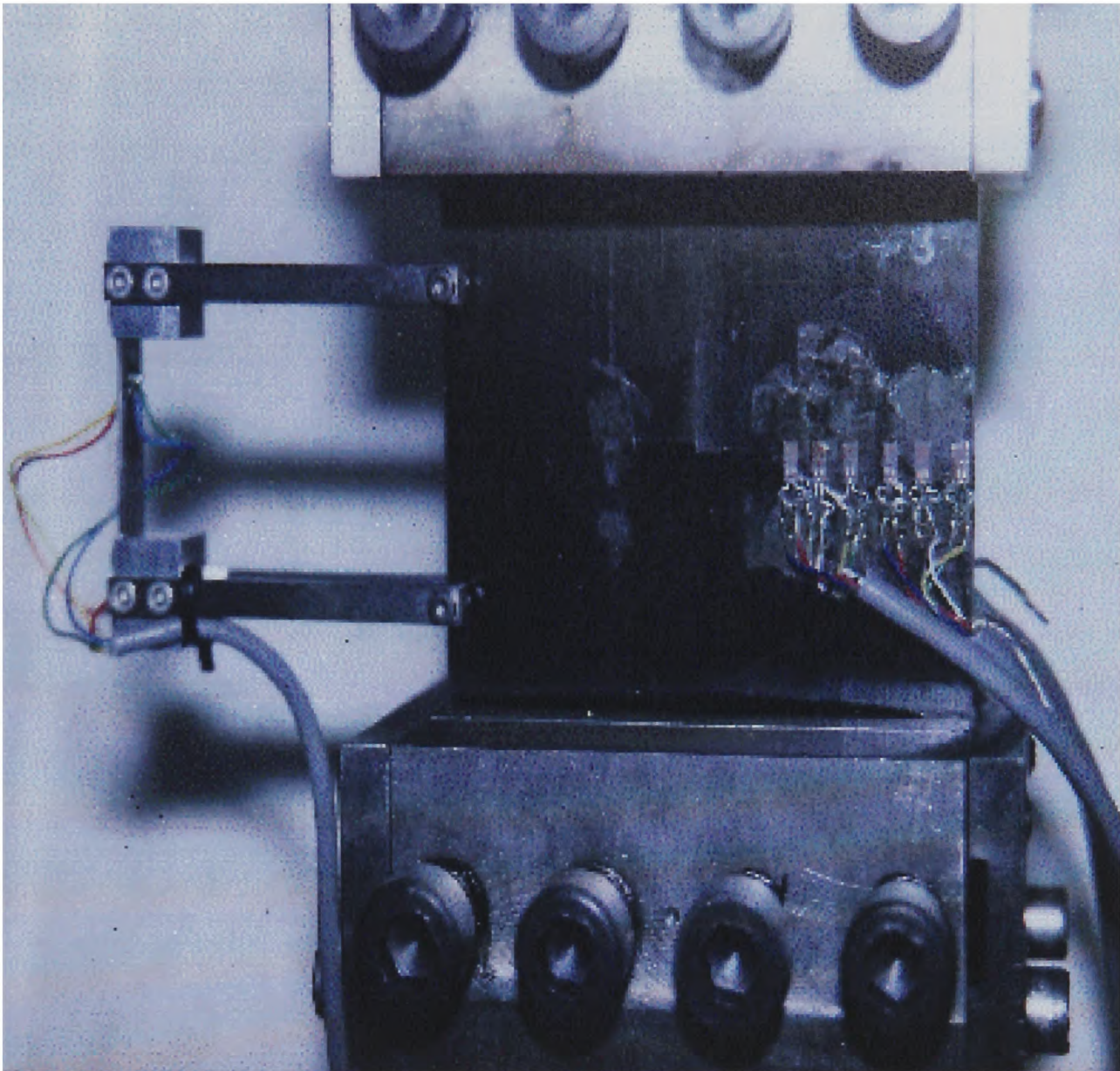
Because it was necessary to reverse the load direction it was not possible to use wedged jaws for gripping the coupon. Instead one side of the jaws was fixed and the other was movable so that a clamping force could be provided.

Figure 5.2 shows the design of the jaws. Figure 5.3 shows a coupon mounted in the jaws for testing. The displacement transducer for measuring overall average strain, and strain gauges, for measuring local strain, can be seen mounted on the coupon.



CFRP Test Coupon Clamp to fit Dartec Fatigue Machine

Figure 5.2



21 Ply Coupon in Fatigue Jaws with Displacement Transducer and Strain Gauges

Figure 5.3

5.2 Testing Machines

5.2.1 Denison

For compression after impact testing using the pin-ended compression test rig, and for static tension tests of the 140 x 85 mm tabbed coupons, a Denison testing machine was used. The Denison is a manually controlled hydraulic testing machine with an analogue readout of current and maximum load.

5.2.2 Dartec

For all cyclic loading tests of the 140 x 85 mm coupon a Dartec hydraulic testing machine was used with a 100 kN capacity load cell. This is a computer driven, servo-valve controlled, hydraulic testing machine. The machine is capable of providing cyclic loading at frequencies of up to 10 Hz as well as quasi-static ramp tests at pre-determined rates. Control is via either a stroke or load feedback loop.

For stable control of the Dartec testing machine it is necessary to set feedback control gains to values appropriate to the stiffness of the specimen under test. Both proportional and differential feedback gains were set to give stable machine response throughout the loading rate regime used in testing.

5.3 Data Acquisition

5.3.1 Strain Gauges

Surface strain measurements from the CFRP coupons were made using Kyowa strain gauges. These were of type KFG-1-120-C1-11, with a gauge length of 1 mm and a typical gauge factor of $2.10 \pm 1.0\%$. These strain gauges are temperature compensated for bonding to steel. For use on CFRP, however, it was necessary to use one dummy gauge for each active gauge in order to obtain correct temperature compensation. The dummy gauge was bonded to an unstressed coupon of CFRP under the same ambient conditions. Strain gauges were calibrated using the manufacturer's gauge factor.

5.3.2 Displacement Transducer

In order to measure the overall strain applied to the 21 ply coupons during testing a clip gauge displacement transducer was used. This may be seen attached to the side of a coupon in Figure 5.3. The clip gauge uses strain gauges bonded to a spring steel armature mounted between rigid arms. The arms were clamped on to the coupon with threaded pins. Displacement of the clip gauge arms causes bending of the spring steel armature and this is detected by an amplifier connected to the strain gauges. The clip gauge displacement transducer was statically calibrated against a vernier micrometer.

5.3.3 Data Logging

Strain gauge and displacement transducer signals were recorded on a multi-channel digital data logging system developed by Computer Instrumentation Ltd.

The logging system was primarily designed for recording maximum and minimum values of signals at intervals during fatigue tests. With appropriate adjustment of sampling rates, however, it was also suitable for recording data from quasi-static tests to failure.

5.4 Undamaged Tension

With the aim of establishing both a lower bound for the tensile strength of T300/914 CFRP and the tensile limit for fatigue tests, six undamaged tensile strength tests were conducted on 21 ply coupons. Three of these tests were of an initial waisted design of the 21 ply coupon with radiused ends, 85 mm wide tabs and a 75mm width gauge section. Three were coupons without waisting and 85 mm wide. In all of the tests the tensile failure was initiated at the stress raiser giving the highest local stress concentration. This was either the point at which the radiused end blended into the gauge length or the end tabs of the coupon. The results of the tests are detailed in Table 5.1.

Undamaged Tension Tests

Stress Raiser	Coupon Width (mm)	Failure Stress (MPa)
10 mm Radius	75	365
10 mm Radius	75	384
10 mm Radius	75	441
Coupon Tab	85	647
Coupon Tab	85	717
Coupon Tab	85	699

Table 5.1

The failure strains of a unidirectional ply of T300/914 CFRP in the longitudinal (fibre) and transverse directions are calculated in Table 5.2. The longitudinal failure strain is higher than the transverse failure strain. Therefore at the point of failure in the ultimate tensile strength test of the laminate only the 0° plies are supporting load while the 90° plies have already cracked.

Unidirectional T300/914 CFRP Failure Strains

Failure	Direction	Elastic Modulus, <i>E</i>	Tensile Strength, <i>T</i>	Failure Strain, ϵ_f
Fibre	Longitudinal, 1	139 GPa	1700 MPa	1.22 %
Matrix	Transverse, 2	9.4 GPa	74 MPa	0.79 %

Table 5.2

The strengths evaluated from the tensile tests allow the stress concentration factors associated

with the failure initiating stress raisers to be calculated. This is done by comparison to the small specimen unidirectional tensile strength of 1700 MPa. These are shown in Table 5.3. The equivalent fibre stress is 21/11 times the failure stress since only the 11 0° plies of the 21 ply laminate are resisting the load.

Stress Concentration in Undamaged Tension Tests

Stress Raiser	Failure Stress (MPa)	Equivalent 0° Ply Stress (MPa)	Stress Concentration Factor
Coupon Tab	688	1313	1.29
10 mm Radius	397	758	2.24

Table 5.3

It was concluded from these results that the stress concentration factor due to the 10 mm radius was significantly greater than that arising from the glued aluminium tabs and therefore the plain gauge width (non waisted) coupon design was chosen for the testing programme.

The small specimen tensile strength of 1700 MPa is equivalent to a fibre stress of 2833 MPa (fibre volume fraction $V_f = 60\%$) and approaches the manufacturer’s fibre strength of 3.53 GPa. (2.1.1).

5.5 Tension After Impact

5.5.1 Experimental Results

Four tensile strength tests after low speed impact damage were conducted with the 21 ply coupons. The results of the tests are detailed in Table 5.4.

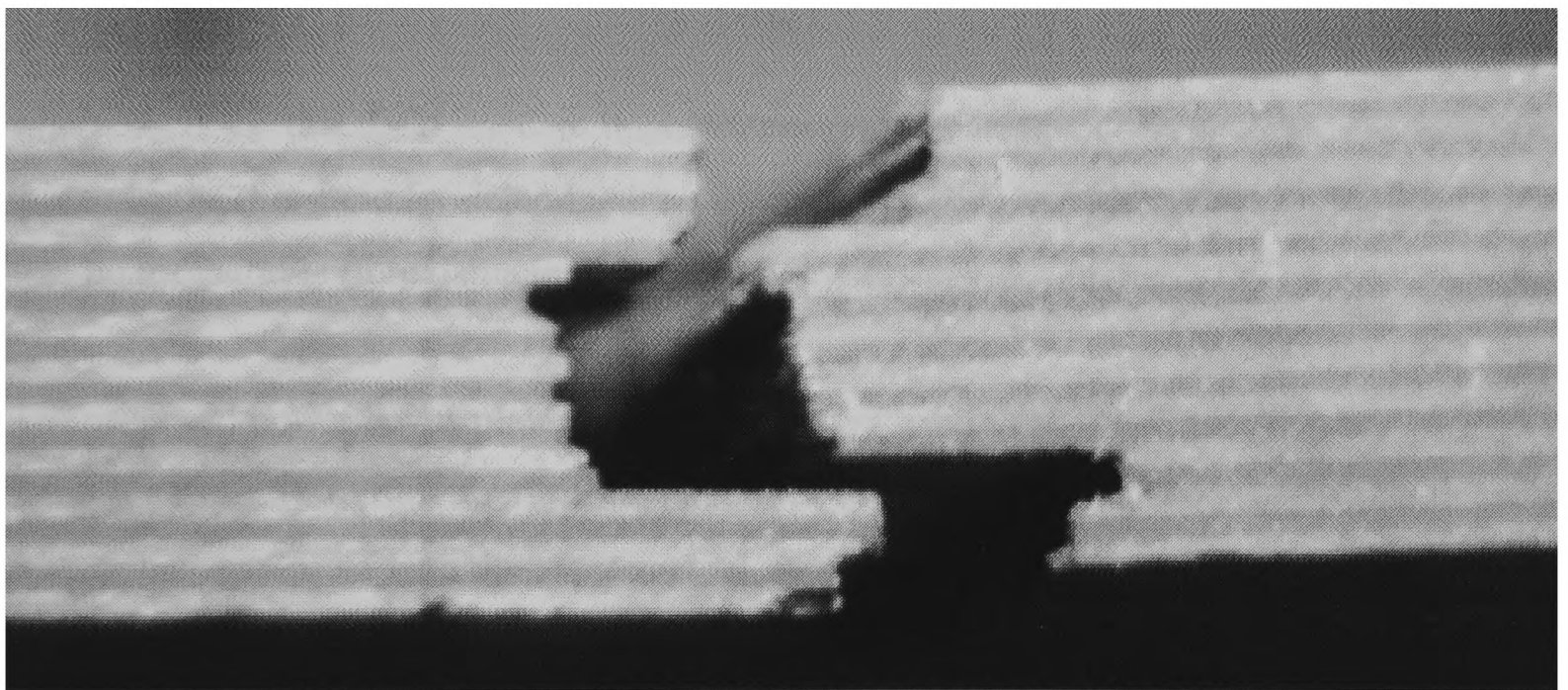
Tension After Impact Tests

Impact Energy (J)	Indentation (mm)	Delamination Area (mm ²)	Nominal Failure Stress (MPa)
12.4	5.63	1235	132
9.6	0.75	763	191
8.9	1.23	528	163
4.4	0.09	320	368

Table 5.4

The failure mode in the tension after impact tests was fast fracture of global tensile cracks initiated from the central impact damage zone. Cracks propagated across the full width of the coupon from the impact damage site, perpendicular to the load and 0° fibre direction.

Figure 5.4 shows a typical tensile failure of a 21 ply coupon. The image shows the edge of the coupon. The failure crack has propagated from the central impact damage zone to the edge shown.



21 ply Coupon, Tension after Impact, Failure at 191 MPa,
(Impact Energy 9.56 J, Indentation 0.75 mm, Delamination Area 763 mm²)

Figure 5.4

5.5.2 Predictive Model for Residual Tensile Strength

Jenq, Wang and Sheu [36] conducted tests on the residual tensile strength after impact of GFRP coupons and found good agreement with a model modified from that of Caprino. The residual strength T_r is predicted from the undamaged strength T_0 and the energy absorbed in impact U_d :

$$\frac{T_r}{T_0} = \left(\frac{U^*}{U_d} \right)^\alpha \quad 5.1$$

The parameters U^* and α must be determined from experimental data. The absorbed kinetic energy of the projectile is determined from measurement of the projectile mass, m , its initial velocity, v_0 , and its rebound velocity, v_1 (or exit velocity in the case of target perforation):

$$U_d = \frac{1}{2}m(v_0^2 - v_1^2) \quad 5.2$$

This model requires that there is a threshold absorbed energy for which there is no reduction in tensile strength. That is, if $U_d = U^*$ then $T_r = T_0$. Because the rebound velocity was not measured in the coupon impact tests it was not possible to determine the absorbed energy and therefore a direct comparison to the experimental data cannot be made. However it was felt that a similar model based on incident, rather than absorbed, kinetic energy might give a reasonable fit to the results. An alternative model for tensile residual strength is therefore suggested:

$$\frac{T_r}{T_0} = \left(\frac{U_{df}}{U_0} \right)^\alpha \tag{5.3}$$

where U_{df} is the threshold incident impactor energy for damage in the coupon. We may take this to be the threshold for delamination damage since coupon impact tests showed that there was no measurable fibre damage before delamination.

The proposed power law is plotted against the experimental tests in Figure 5.5. U_{df} is set at 2 J and for a best fit to the experimental data the exponent $\alpha = 0.91$. The proposed model is in reasonable agreement with the experimental results.

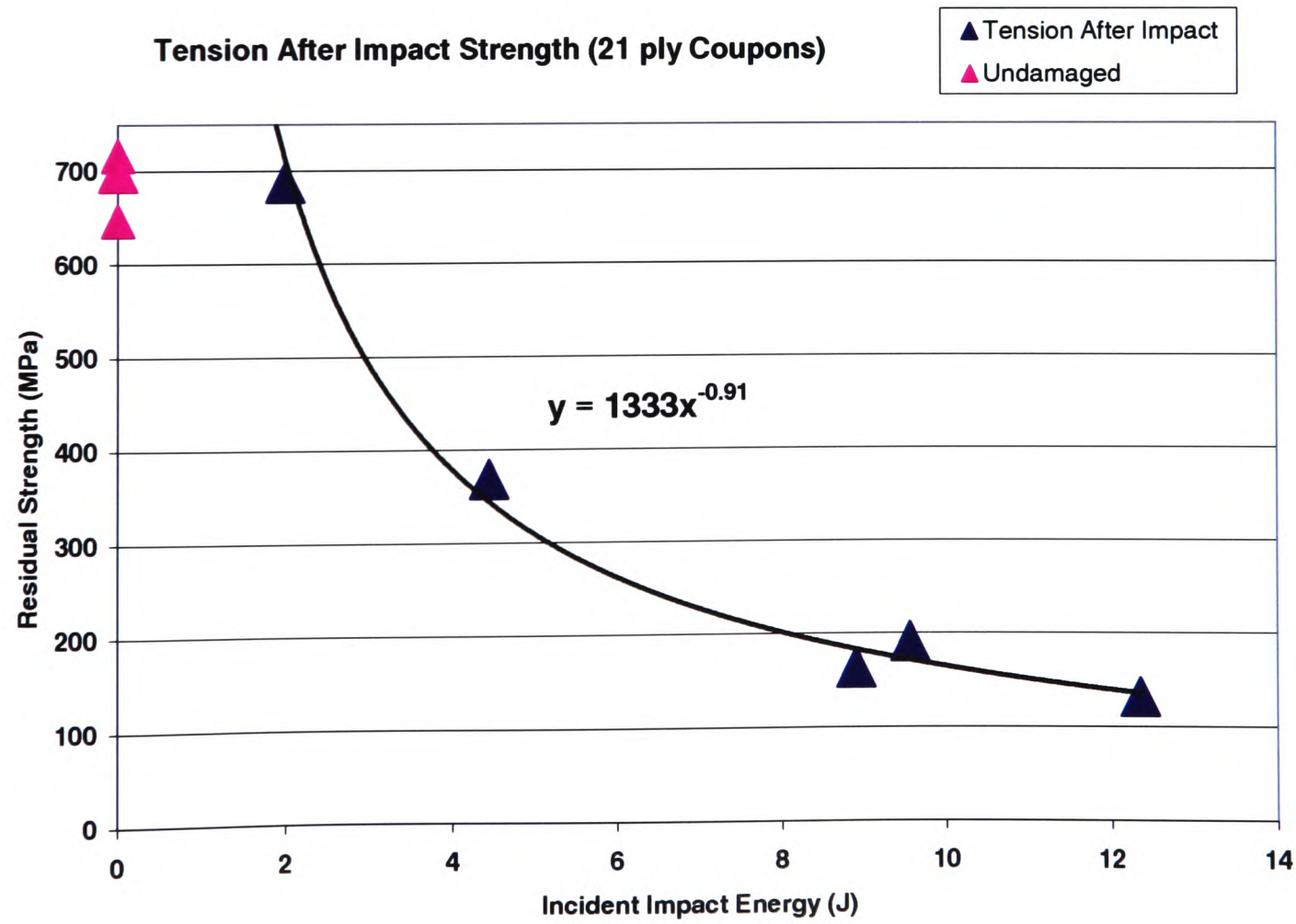


Figure 5.5

5.5.3 Equivalent Hole

One technique for quantifying the severity of impact damage is to compare the residual strength after impact damage with the reduction in strength arising from the stress concentration due to a circular hole.

At the edge of a hole in an infinite, isotropic, plate under uniaxial tension the stress concentration factor is 3. However the anisotropy of composite materials may increase the stress concentration at the edge of a hole to a greater level than this. For an orthotropic laminate with a hole under uniaxial loading along a principal axis x the maximum stress is the circumferential stress on the hole boundary at $\varphi = 90^\circ$ (Figure 5.6 [37]). Daniel and Ishai [37] give the stress concentration at this location, k_σ as:

$$k_\sigma = \frac{\sigma_{\max}}{\sigma_0} = 1 + \sqrt{2 \left(\sqrt{\frac{E_x}{E_y} - \nu_{xy}} \right) + \frac{E_x}{G_{xy}}} \quad 5.4$$

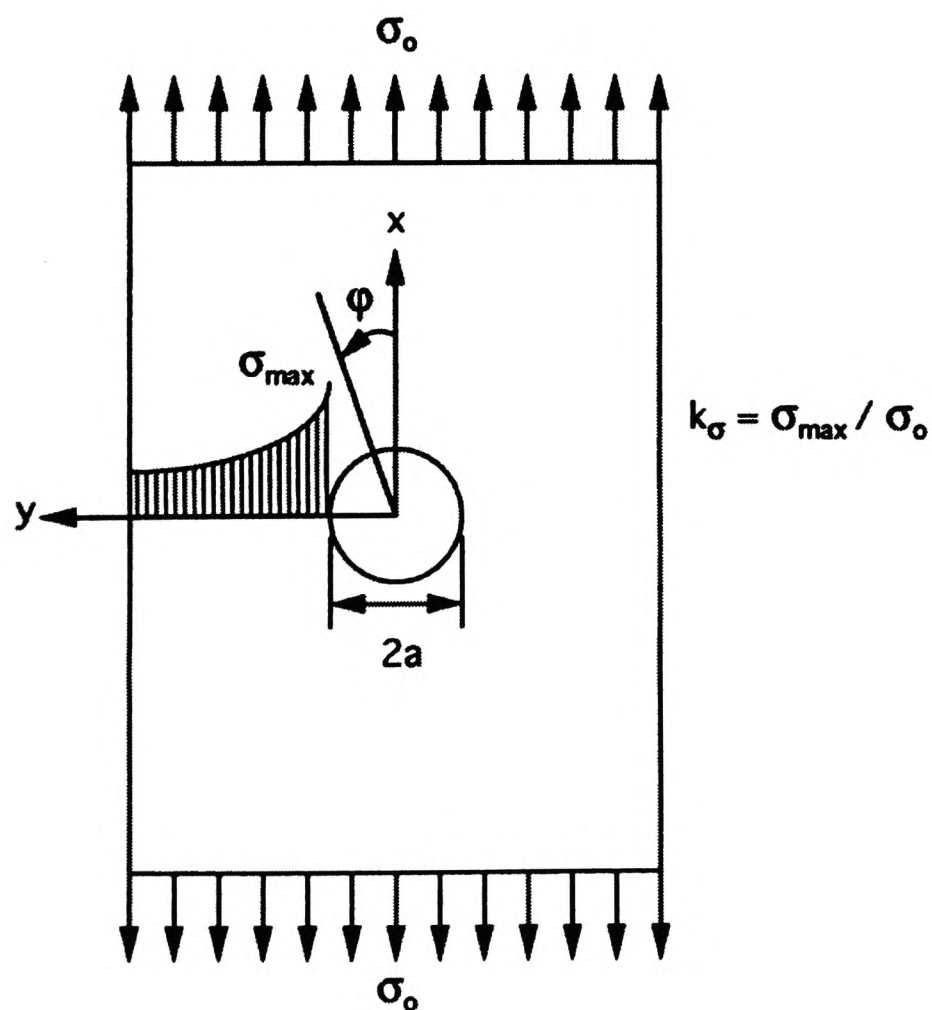


Figure 5.6

For the 21 ply $0^\circ/90^\circ$ laminate tested $k_\sigma = 5.38$. This stress concentration is independent of the size of the hole although residual strength tests have shown a dependency on hole size. This dependency may be attributed to the area of influence of the hole. The stress gradient at the stress concentration at the edge of a small hole will be much steeper than for a larger hole and the area of stress concentration will be much smaller. Although the stress distribution along the transverse axis from the hole may be calculated exactly the following approximate expression [37] can be more useful:

$$\frac{\sigma_x(0,y)}{\sigma_0} \approx 1 + \frac{1}{2}\rho^{-2} + \frac{3}{2}\rho^{-4} - \frac{k_\sigma - 3}{2}(5\rho^{-6} - 7\rho^{-8}) \tag{5.5}$$

where $\rho = y/a$ and a is the hole radius. The average stress criterion (Figure 5.7 [37]) proposes that failure is initiated if there is an average level of stress over a characteristic length a_0 that exceeds the unnotched strength. Integration of the stress distribution therefore gives the strength reduction ratio for a plate with a hole as a function of the hole radius:

$$k_F = \frac{T_r}{T_x} = \frac{2}{(1+\xi)[2+\xi^2+(k_\sigma-3)\xi^6]} \tag{5.6}$$

where

$$\xi = \frac{a}{a+a_0} \tag{5.7}$$

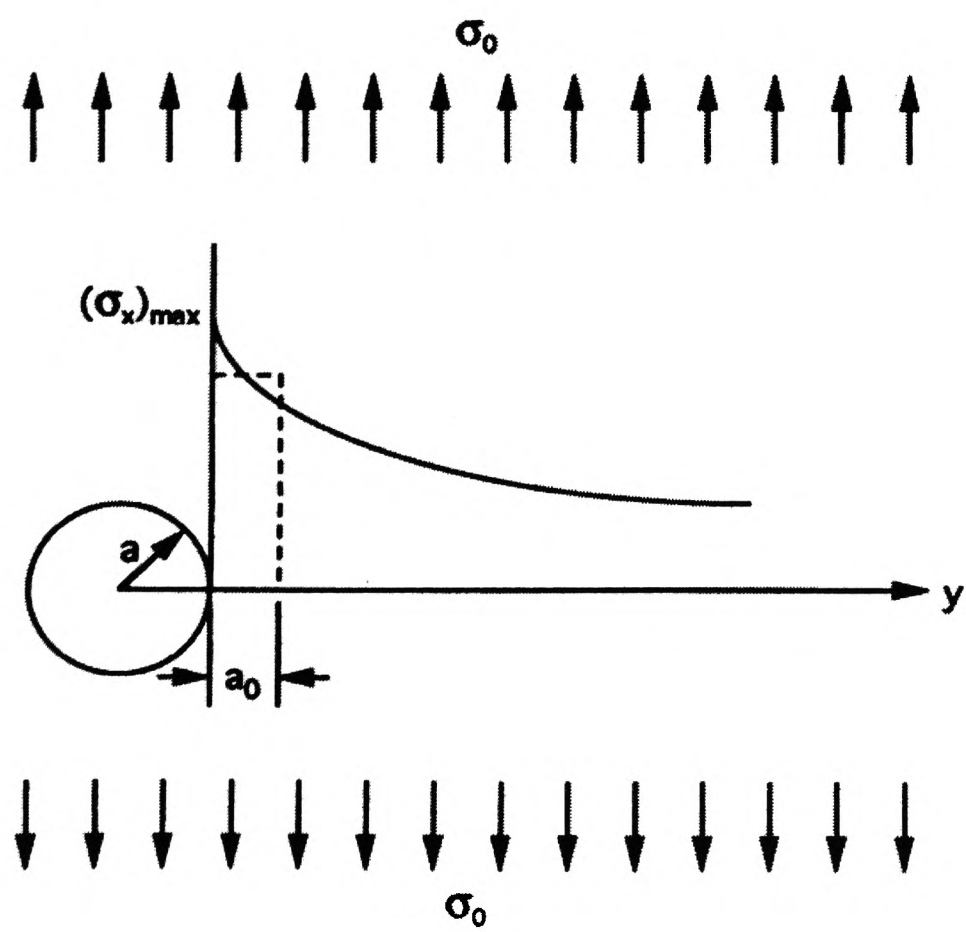


Figure 5.7

The appropriate characteristic length a_0 must be determined experimentally for a particular material lay-up. Daniel et al. [38] conducted tensile strength tests of boron/epoxy coupons containing a central hole. The coupons were 10" wide x 26" long and the hole was 1" in diameter. An 8 ply coupon with a $[0/90/0/90]_s$ lay-up (see Gibson [39] for laminate notation) was found to have a strength reduction ratio of 0.314. Reference to the expected stress concentration and the average stress criterion for failure requires a characteristic length $a_0 = 2.6$ mm to give the correct strength prediction.

The equivalent hole radius to give the same tensile strength reduction ratio measured for the three levels of impact damage tested are given in Table 5.5. The equivalent hole radius is evaluated from the average stress criterion, including an increase in the stress concentration by the ratio of the reduction in net section due to large holes in a plate of finite width. The undamaged failure stress is assumed to be 890 MPa (assuming a unidirectional fibre failure stress of 1700 MPa).

Equivalent Hole Calculation for Tension After Impact

Impact Damage	Medium (BVID)	Severe	Very Severe
Incident Energy (J)	4.4	9.3	12.4
Indentation (mm)	0.09	0.99	5.63
Delamination Area (mm²)	320	646	1235
Nominal Failure Stress T_r (MPa)	368	177	132
Strength Reduction Ratio k_F	0.41	0.20	0.15
Equivalent Hole Diameter (mm)	9.8	28.6	38.4

Table 5.5

5.6 Undamaged Compression

Compression strength of a CFRP component may be considered to be controlled by either a “material strength” or by a structural buckling strength. Which failure mode dominates depends on the laminate lay-up and the slenderness of the component.

5.6.1 Material Compression Strength Coupon

A measurement of the Young’s Modulus and a lower bound on the material compression

strength were obtained by two undamaged compression strength tests. The results are detailed in Table 5.6.

The first test was conducted with a fibrous tape used to pack the specimen between aluminium shims gripped between jaws. In this test the failure mode was crushing at the specimen end. The second test achieved a lower failure stress value with the tape packing absent. In this second test the failure mode was compressive shear initiated at the edge of the aluminium shims.

Undamaged Compression Tests

Failure Mode	Mean Young's Modulus, E (GPa)	Difference in Strain Gauge Readings	Failure Stress (MPa)
End crushing	62.3	1.6 %	692
Shear fracture at support edge	64.5	2.6 %	583

Table 5.6

Two strain gauges were used on the specimens, one on each side of the laminate. The percentage differences in the readings from the strain gauges indicate a low bending component due to buckling over the 20 mm gauge length.

For a 21 ply laminate a rule of mixtures calculation based on E_1 and E_2 and the number of 0° and 90° plies gives an expected Young's Modulus of 77.3 GPa. This is 22% higher than the measured values.

5.6.2 Models for Compression Strength

The compressive strength of carbon fibre reinforced plastics is governed by the failure mode. Various failure modes have been proposed for CFRP as well as empirical estimates of compression strength. A selection of these are reviewed below.

An upper bound for the compressive strength of any material is shear failure along the plane of maximum shear (45° for a homogeneous material). This mechanism is shown in Figure 5.8 [39]. Gibson [39] reports that for compressive strengths good agreement with experimental data is obtained for graphite/epoxy composites if the maximum shear stress is given by a rule of

mixtures based on the fibre volume fraction. The compressive strength is therefore given by:

$$C_1 = 2[S_{f12}V_f + S_{m12}(1 - V_f)] \quad 5.8$$

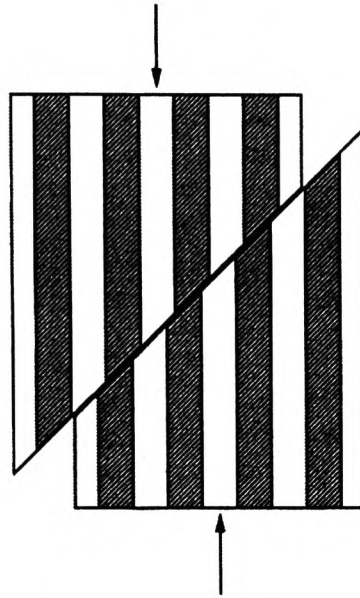


Figure 5.8

If we assume that the fibre failure mode in tension is also a shear failure then we may take $S_{f12} = \frac{1}{2} \times 1700 / 0.6 = 1416$ MPa. The matrix shear strength can be assumed to be approximately equal to the composite in-plane shear strength for T300/914 $S_{m12} = S_{12} = 86$ MPa.

We therefore calculate an upper bound for the material compressive strength of unidirectional T300/914 of $C_1 = 1768$ MPa. This is, of course, slightly higher than the unidirectional tensile strength. For a $0^\circ/90^\circ$ lay-up, however, a further rule of mixtures gives a predicted strength of $C_x = 927$ MPa. The strength measured in the compression test described above is approaching this value (692 MPa).

In a composite, shear failure requires combined shear failure of the fibres and the matrix. However a commonly observed compressive failure mechanism in CFRP is that of micro-buckling or kink-banding (Figure 5.9 [39]). This is a buckling failure of a single fibre which then propagates across the composite through neighbouring fibres as a kink-band. Stabilising resistance to micro-buckling is provided by the matrix. Rosen's model (reviewed by Haeberle and Matthews [40]) is based on classical buckling theory for straight bars on an elastic foundation. The compressive strength is assumed to depend only on the initial matrix shear modulus G_m and the fibre volume fraction V_f :

$$C_1 = \frac{G_m}{1 - V_f} \tag{5.9}$$

If we take the shear modulus $G_m = 2.85$ GPa and the fibre volume fraction $V_f = 0.6$ the predicted compressive strength is 7125 MPa. This is clearly too high.

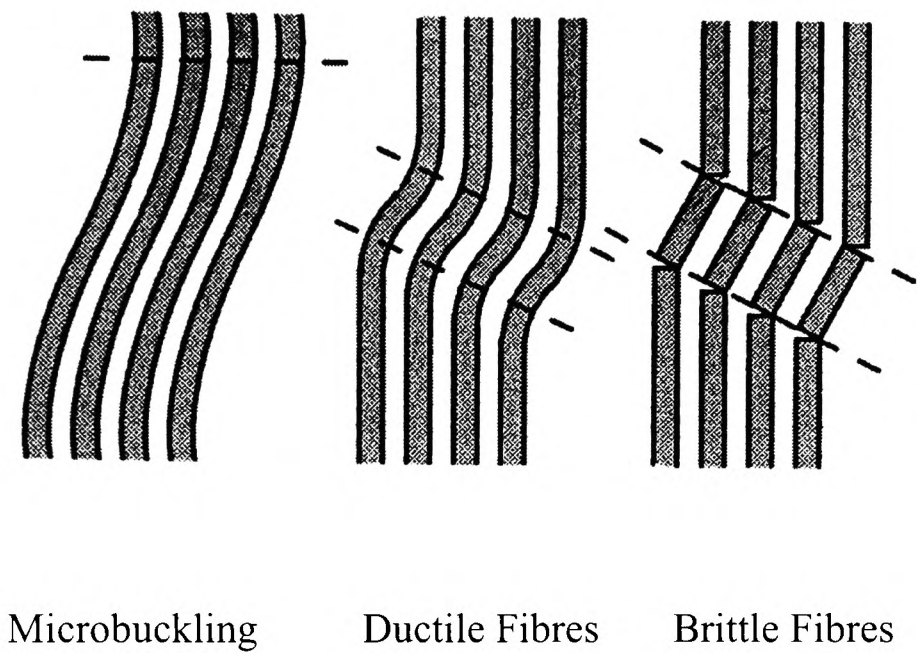


Figure 5.9

Soutis, Curtis and Fleck [41] maintain that compressive fracture of carbon fibre/epoxy laminates is always governed by plastic microbuckling of the 0° plies. They observed that for unnotched laminates the compressive failure strain is independent of the lay-up. With modern ductile and strong matrices they found that the compressive strength was typically 60% of the composite tensile strength.

Strong experimental and theoretical support was found for the Budiansky kinking analysis where microbuckling of unidirectional material is governed by the shear yield stress τ_y of an elastic-perfectly plastic composite, the shear yield strain γ_y and the misalignment angle of the fibres φ :

$$C_1 = \frac{\tau_y}{\varphi + \gamma_y} \tag{5.10}$$

By substitution of strength values for T800/924C CFRP Soutis et al. found that $\varphi = 3^\circ$, which was comparable with measured values of fibre misalignment.

To avoid the difficulties in modelling a compressive failure mode some authors have suggested

an empirical approach. One suggested empirical upper bound [40] for the compressive strength of the composite is 90% of the composite tensile strength:

$$C_1 = 0.9T_1 \quad 5.11$$

For a $0^\circ/90^\circ$ lay-up, where the effective tensile strength is 850 MPa, this implies a compressive strength of 765 MPa. The measured compressive strength of 692 MPa approaches this value.

5.6.3 50 x 50 mm, 29 ply Coupon

Seven undamaged compression strength tests were conducted with the 50 x 50 mm, 29 ply coupons. The pin ended compression test rigs were used. The range of failure stresses obtained was 252 to 292 MPa with a mean strength of 276 MPa. The failure mode was compressive shear across the horizontal mid section of the coupon accompanied by some longitudinal delamination splitting

5.6.4 140 x 85 mm, 21 ply Coupon

An undamaged compression strength test was also conducted on a full scale 21 ply coupon in order to determine the ultimate compressive strength and the degree of out of plane buckling. With an unsupported gauge length of 65 mm, and no impact damage to initiate failure at lower stresses, the coupon does buckle significantly. Two strain gauges were mounted on either face of the coupon to monitor strain, and the degree of buckling, throughout the test. The measurements made are shown in Figure 5.10. Both the applied stress and the measured back and front face strains are plotted against the elapsed time during the test. It can be seen that up to 60 seconds into the test, at a stress of 250 MPa, there is little divergence of the strain gauge readings due to structural buckling.

The maximum load obtained in the test was 324 MPa and this was taken as the undamaged compression strength for the 21 ply coupon.

5.6.5 Structural Buckling

A slender composite component may macro-buckle as a whole structure. This macro-buckling may be followed by a local failure at the point of maximum bending which will be either tensile failure or compression failure, as described above, depending on the stress distribution in the section.

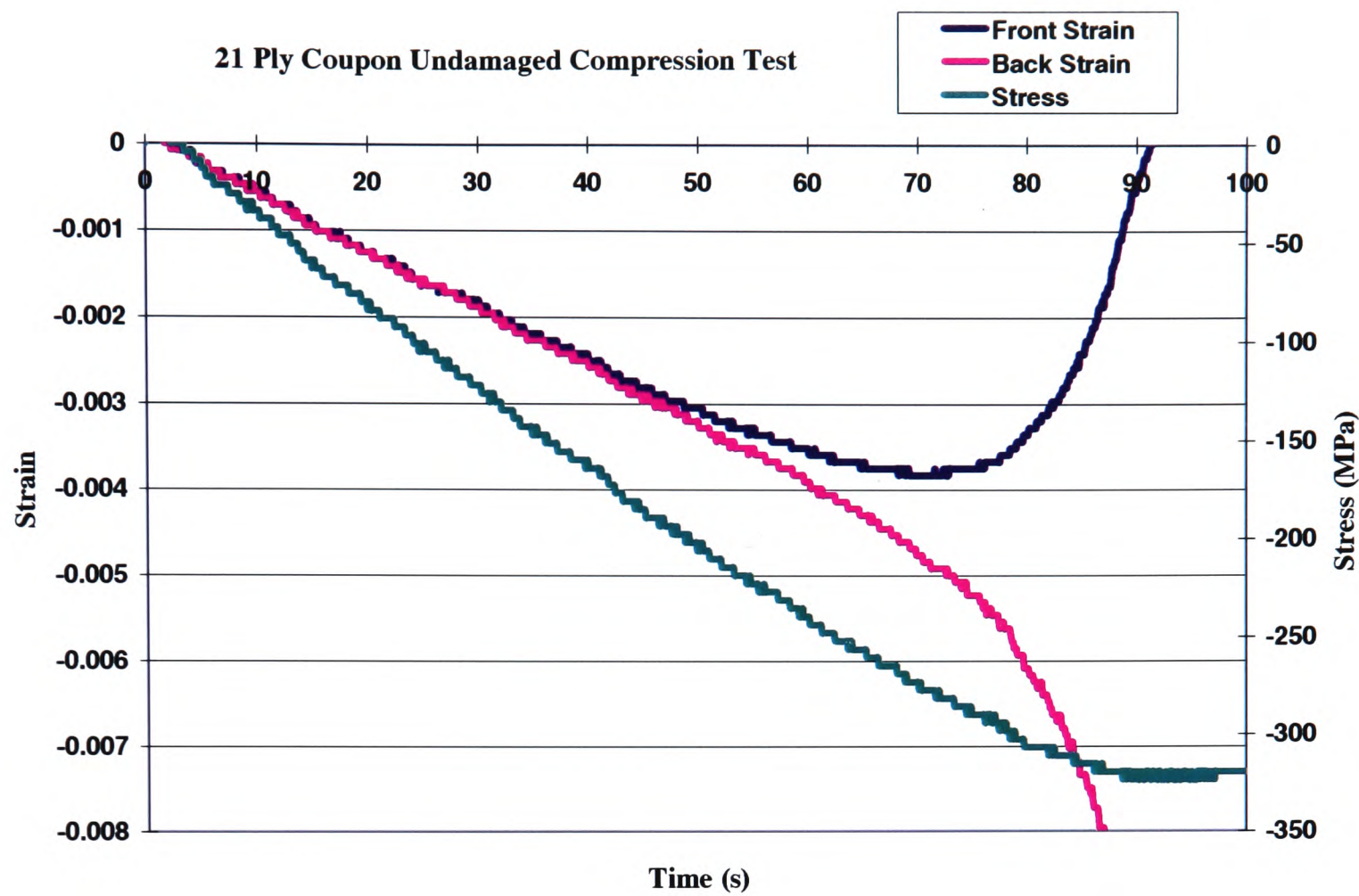


Figure 5.10

Macrobuckling involves overall instability of the specimen rather than just a material property. The critical Euler stress σ_E for buckling for a pin ended column is given by:

$$\sigma_E = \frac{\pi^2}{12} E \left(\frac{d}{l} \right)^2 \tag{5.12}$$

For the $0^\circ/90^\circ$ lay-up coupons we may take an approximate value for E as the mean of E_l and E_2 (the transverse and longitudinal Young’s Moduli). Therefore $E = 74.2$ GPa. The predicted Euler buckling load for the 50 x 50 mm, 29 ply coupon is therefore $\sigma_E = 316$ MPa.

The 21 ply coupon had clamped ends and the predicted Euler buckling load is greater:

$$\sigma_E = \frac{\pi^2}{3} E \left(\frac{d}{l} \right)^2 \tag{5.13}$$

For the fixed-end 140 x 85 mm, 21 ply coupon $\sigma_E = 453$ MPa.

The transverse shear modulus of a composite laminate is approximately that of the matrix and therefore proportionally much lower than the in-plane moduli. Consequently shear deformation

may be significant in buckling. The critical stress when shear deformation is included is given by:

$$\sigma_{cr} = \frac{\sigma_E}{1 + n \frac{\sigma_E}{G}} \quad 5.14$$

where σ_E is the critical Euler buckling stress, G is the shear modulus of the composite and n is a constant depending on the cross section. For a rectangular cross section n is 1.2. The critical buckling values for the compression coupons are therefore:

29 ply, 50 x 50 mm Coupon	$\sigma_{cr} = 279$ MPa	(c.f. experiment 275 MPa)
21 ply, 140 x 85 mm Coupon	$\sigma_{cr} = 380$ MPa	(c.f. experiment 324 MPa)

We can conclude from these results that, in the undamaged state the 29 ply coupon is on the point of elastic buckling failure when compressive fracture occurs. The 21 ply coupon, however, fails at a stress below the Euler buckling load when undamaged.

It is worth appreciating at this point that any real composite component will not be perfectly straight and any imperfection will allow macrobuckling at a lower load than the Euler buckling load. Indeed no matter how stocky the specimen, there will always be some bending component which arises from either an initial out-of-straightness or the misalignment of the compressive load.

5.7 Compression After Impact

5.7.1 Literature Results

Residual compressive strength after impact has for some time been regarded as a serious limitation to the application of CFRP laminates. Consequently the problem has attracted many researchers. Some experimental findings from the literature are described here.

Prichard and Hogg [42] used a compression after impact (CAI) strength test to investigate the effect of impact energy and the differing CAI behaviour of thermoset (toughened epoxy) and thermoplastic (PEEK) matrices.

The CAI test rig was based on a Boeing design but with slightly modified anti-buckling guides.

Materials tested were Ciba-Geigy Fibredux 924c (a toughened epoxy with Torayca T800H carbon fibres) and ICI APC-2 (PEEK with AS-4 carbon fibres). Specimens were 89 x 55 mm and of a 16 ply quasi-isotropic lay-up giving a thickness of approximately 2 mm. For impact the specimens were clamped between plates with a 40 mm diameter window and struck by a 20 mm diameter impactor weighing 3.96 kg with impact energies up to 15 J. The compression test was conducted with direct end axial loading and with anti-buckling guides providing restraint against out of plane deflection, 5 mm from the specimen edges.

For APC-2 an approximately linear reduction in CAI strength with increasing impact energy was observed with a 50% reduction in strength at a 15 J impact. The T800/924c, however, gave a much sharper reduction in strength. Impacts over 2 J resulted in a CAI strength of approximately 50% of the undamaged strength. Prichard and Hogg also correlated the CAI strength with the delamination damage width measured by C-scanning. When the CAI strength was plotted against damage width they found that both materials quite closely followed the net section strength reduction line; that is a delamination of half the specimen width gave a residual strength of approximately half the undamaged strength. Consequently the difference in behaviour between T800/924c and APC-2 was due to a difference in resistance to delamination in impact. APC-2 (PEEK) was much more resistant to delamination at these low-energy impacts.

Ishikawa et al. [43] investigated the sequence and propagation of delamination buckling in compression after impact (CAI) tests in both carbon fibre/epoxy and CF/PEEK flat plates.

The NASA CAI testing method was used with a 48 ply quasi-isotropic lay-up plate 127 x 254 mm. The thickness was 7.3 mm for CF/epoxy and 6.2 mm for CF/PEEK. For impacting the specimens were clamped with a window 102 x 102 mm and struck by an instrumented drop weight. Impacts were fixed at 27 J and used to generate the intended delamination.

Ultrasonic scanning revealed that the delaminations followed a fan shaped pattern through the depth of the laminate. This characterisation of the damage was used in a finite element model of the laminate which showed good agreement with the experimentally observed delamination buckling. The process of delamination buckling was observed by Moiré topography and measurement of the central deflection of the plate. In CF/epoxy the delamination blister

increased in size and amplitude with increasing compressive load until overall structural buckling of the plate occurred in the opposite direction. CF/PEEK was observed to have greater resistance to the propagation of delamination, and failed at higher compressive strains. After impact the residual strength of CF/epoxy was approximately 50% and for CF/PEEK approximately 75%.

5.7.2 Experimental Results

A total of 42 compression after impact (CAI) tests were conducted, 37 on the 29 ply coupons using the pin ended compression test rig and 5 on the 21 ply coupons.

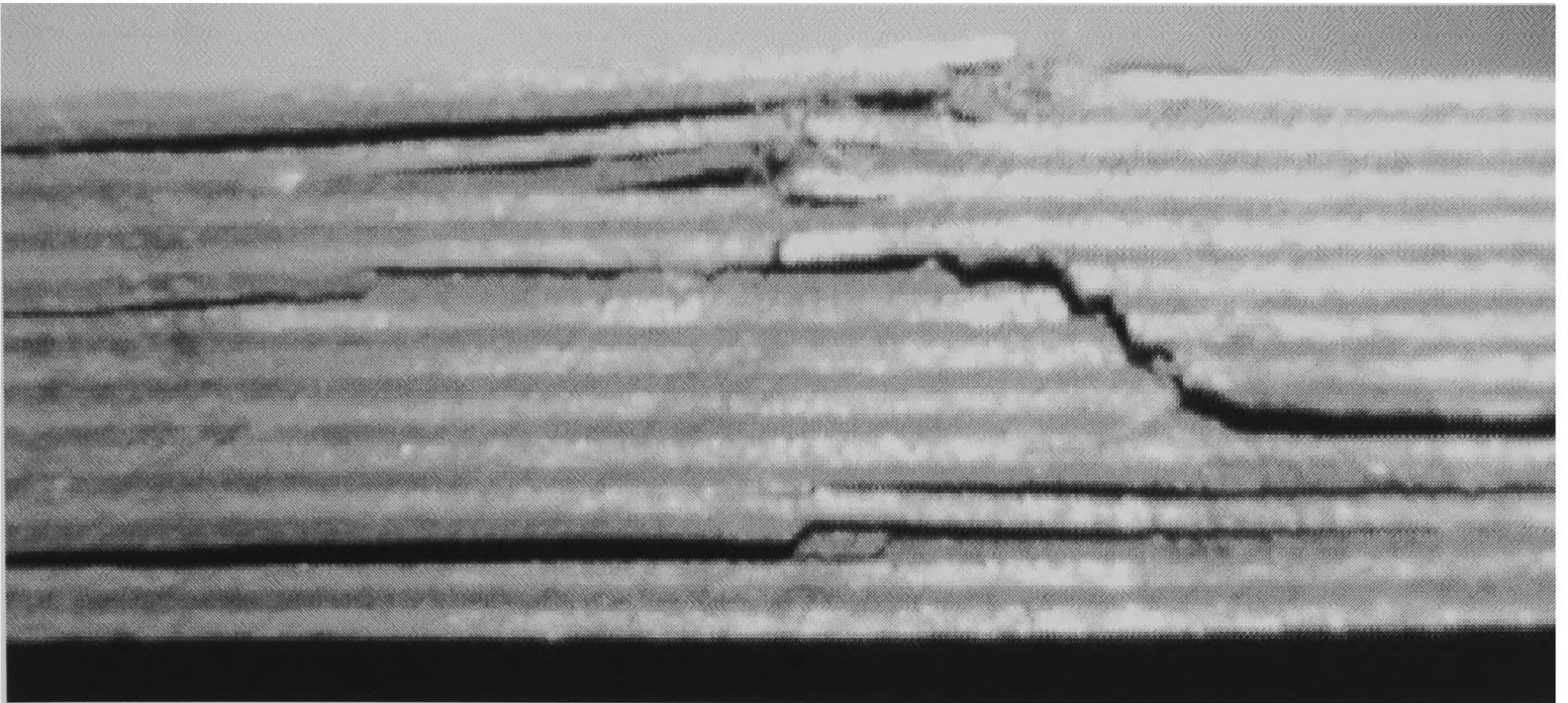
Table 5.7 details the results of the 21 ply CAI tests. These results are plotted in Figure 5.13. Also plotted is the maximum stress value obtained in the undamaged coupon test and the failure stress obtained in the pure compression strength test. Because the CAI strengths are so much lower than the buckling load for the coupon it is more useful to compare the strengths with the much higher material strength. The magnitude of the reduction in compression strength due to impact damage can then be seen.

The failure mechanism observed with the 21 ply CAI tests was generally compressive shear accompanied by some longitudinal delamination splitting. Figures 5.11 and 5.12 show typical compression after impact failures in the 21 ply coupons. The images are of the coupon edges. The failure crack started at the impact damage zone and propagated perpendicularly across the coupon width to the edges.

Compression After Impact Tests (21 Ply Coupon)

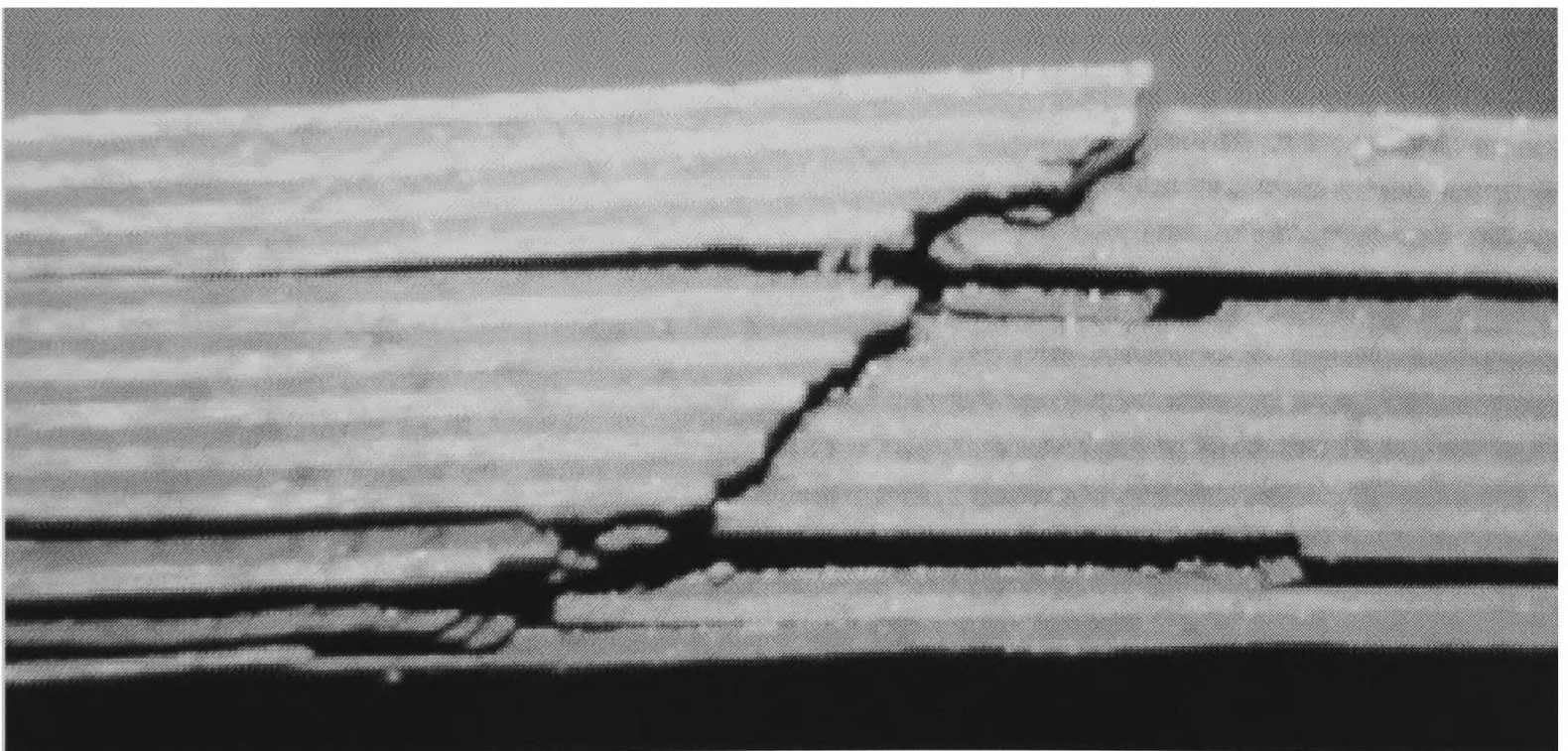
Impact Energy (J)	Indentation (mm)	Delamination Area (mm ²)	Failure Stress (MPa)
9.89	Not Measured	824	-161
9.79	0.88	767	-177
9.18	0.64	635	-182
4.3	0.1	312	-223
3.69	0.05	306	-244

Table 5.7



21 ply Coupon, Compression after Impact, Failure at -177 MPa,
(Impact Energy 9.79 J, Indentation 0.88 mm, Delamination Area 767 mm²)

Figure 5.11



21 ply Coupon, Compression after Impact, Failure at -222 MPa,
(Impact Energy 4.3 J, Indentation 0.1 mm, Delamination Area 312 mm²)

Figure 5.12

The results of the 42 CAI tests conducted on the 29 ply, 50 x 50 mm, coupons are shown in Figures 5.14 and 5.15. The residual compression strength has been plotted against both incident impact energy and delamination area measured by C-scan. The scatter observed in both presentations of the experimental data is significant. Best fit linear trendlines have been fitted to the results.

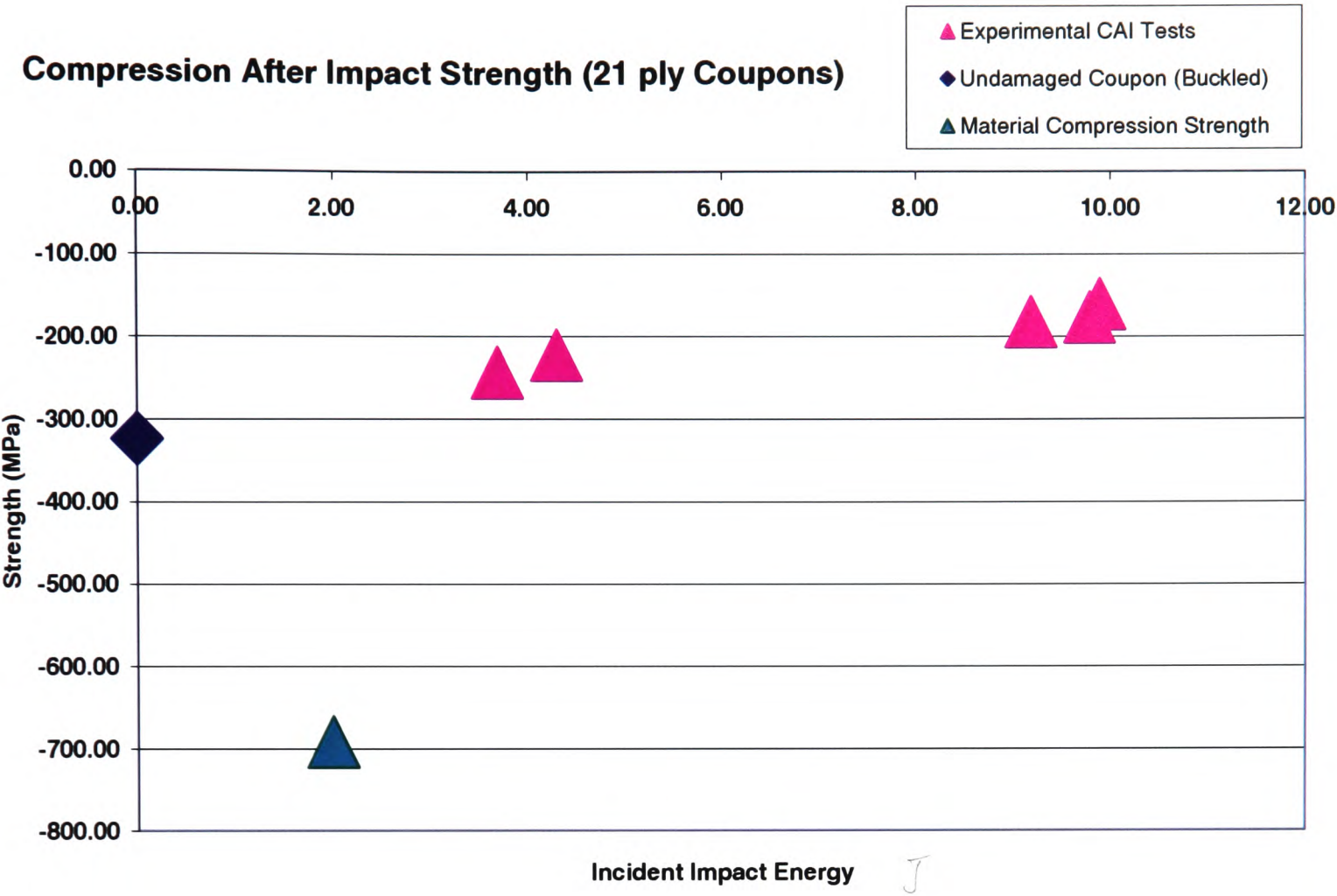


Figure 5.13

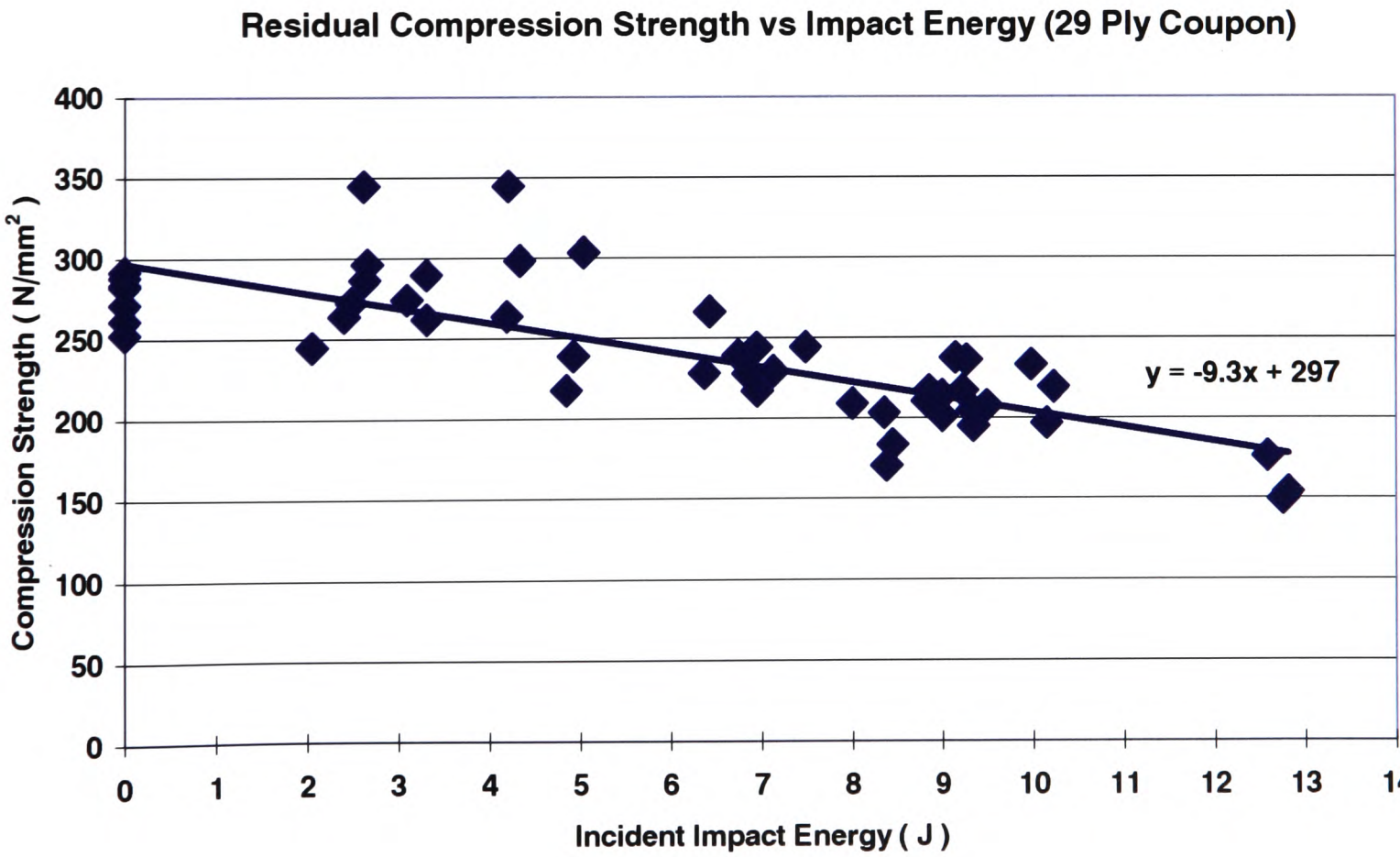


Figure 5.14

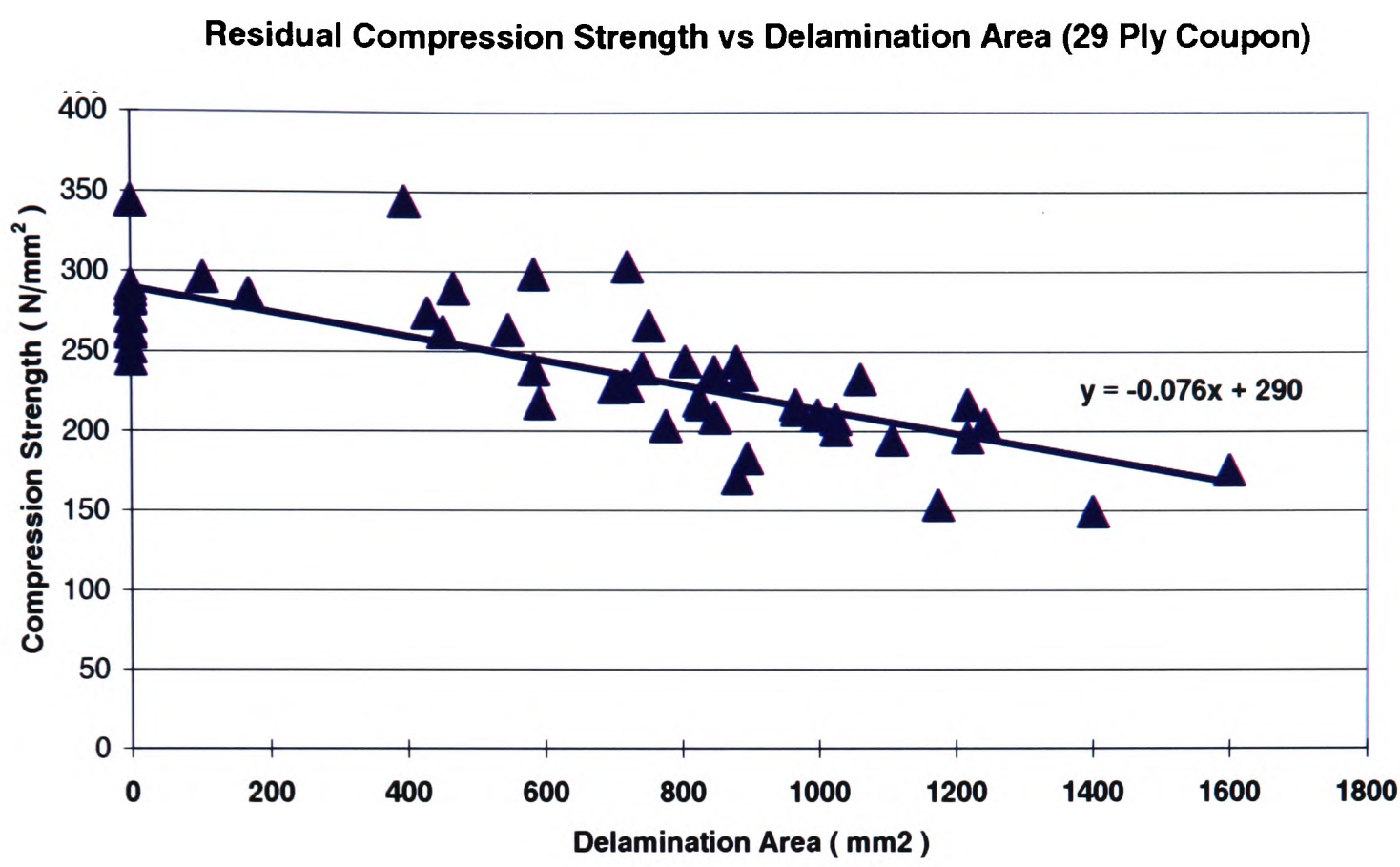
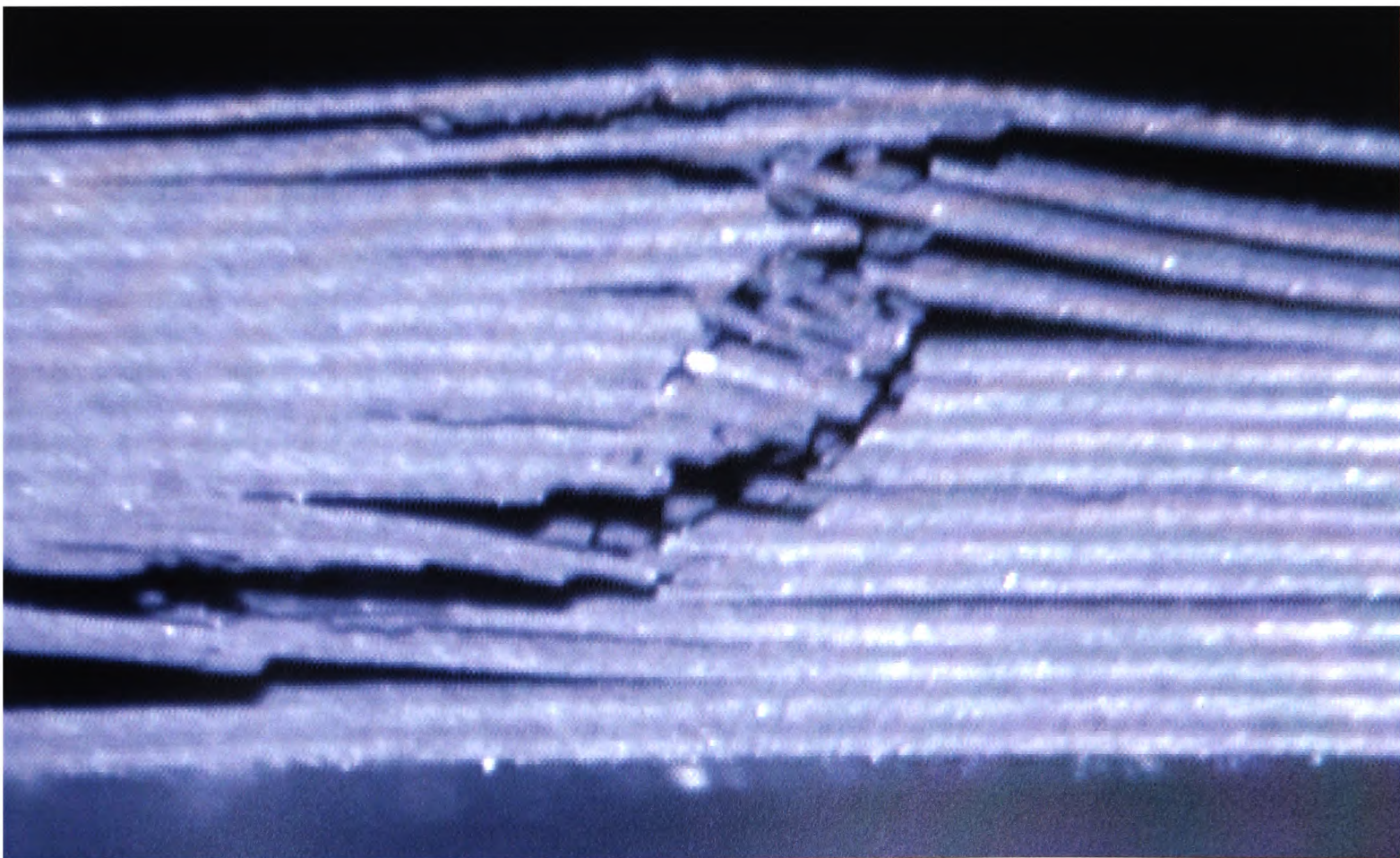


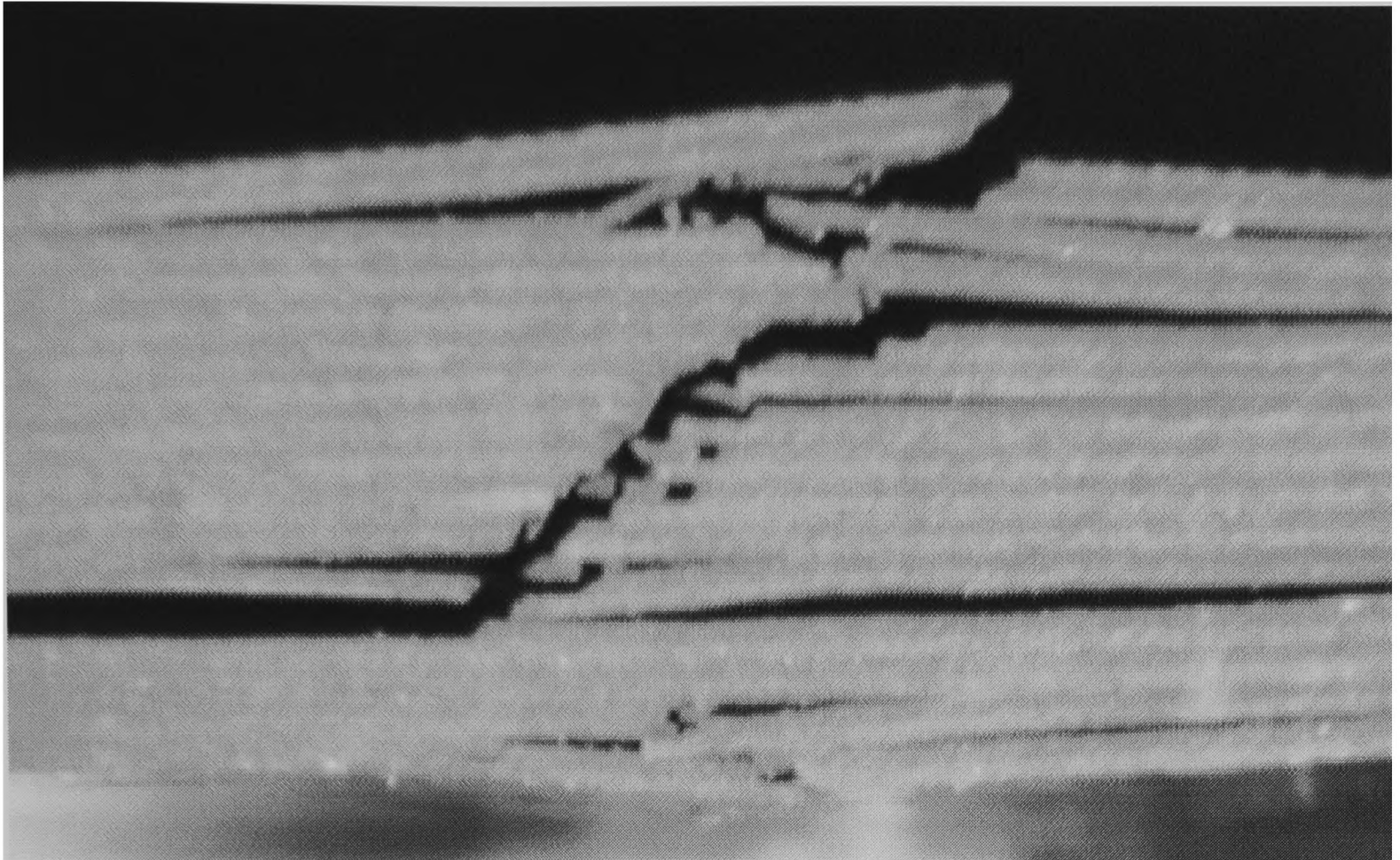
Figure 5.15

Figures 5.16 and 5.17 show typical compression after impact failures in the 29 ply coupons. The failure is a combination of compressive shear and longitudinal delamination splitting.



29 ply Coupon, Compression after Impact, Failure at -266 MPa,
(Impact Energy 6.43 J., Delamination Area 751 mm²)

Figure 5.16



29 ply Coupon, Compression after Impact, Failure at -197 MPa,
(Impact Energy 10.15 J, Delamination Area 1218 mm²)

Figure 5.17

5.7.3 Delamination Buckling

Much work on the reduction of compression strength after impact has focused on delamination damage and the phenomenon of delamination buckling. This work generally assumes a single delamination in an otherwise homogeneous material under compressive axial load.

Kachanov [44] was the first to analyse the problem. He assumed a thin surface delamination on a very thick body. The buckling load is determined by equating the axial compressive strain energy in the layer before buckling with the bending strain energy after buckling and work done in fracture. It is assumed that there is no axial strain in the buckled layer and this defines the amplitude of the buckle.

Simites et al. [45] made a much more detailed analysis. They considered a homogeneous plate with a single delamination through the full width under axial compression. Using small perturbation theory they determined the buckling loads for the general case of a delamination of length a , distance b from one end and at a depth h in a plate of thickness H and overall length L . Their results show that unless the delamination is sufficiently slender, that is long and close

to the surface, it will not affect the overall structural buckling of the plate. Only a slender surface delamination will buckle before overall structural buckling occurs.

The critical slenderness for a delamination in a plate clamped at both ends is given very approximately by:

$$\frac{a}{L} \frac{H}{h} > 1 \quad 5.15$$

Yin et al. [46] continued this work to consider the post buckled state of the delaminated beam plate. Their analysis allowed determination of the energy release rate for the delamination crack tip. Knowledge of the fracture energy for delamination should then allow prediction of the growth of the delamination buckle.

Kyoung and Kim [47] further advanced the work by considering the effect of shear deformation on the buckling loads and post buckling behaviour. The motivation for this direction of work is the low transverse shear modulus of composite laminated plates which is only the low shear modulus of the matrix. Consequently shear deformations may be significant and may affect the buckling loads for delamination buckling. The effect of shear deformation is only significant, however, for stocky beam plates. The greater the slenderness the more the behaviour is controlled by flexural bending and the less by shear deformation. Kyoung and Kim found their prediction to be in good agreement with experiments for the delamination buckling of slender surface delaminations.

Nilsson et al. [48] also considered a single surface delamination but in two dimensions. Using a commercial finite element analysis package they investigated the buckling, post buckling and subsequent growth of an initial circular surface delamination. Particular attention was paid to the crack growth fronts from the initial defect. The analysis was compared with experimental loading of a composite plate with an initial circular delamination formed by the inclusion of two sheets of PTFE between plies. The growing delamination blister was observed by a combination of C-scanning, X-ray radiography and displacement transducer measurement. The delamination was predicted and observed to grow from the edges of an initial circular defect and to spread across the plate, transverse to the direction of compressive loading.

The phenomenon of delamination buckling can be experimentally demonstrated [49] and it is possible to make accurate theoretical predictions of experimental behaviour. However, applying the models to real impact damage is very difficult. Real impact damage in a composite laminate will not be limited to a single delamination, but will include a distribution of delaminations throughout the thickness. There will also be other damage modes which complicate the damage state and prevent the simple application of the delamination buckling models: in particular fibre fracture and in-plane matrix cracks, both of which make the material highly unhomogeneous.

5.7.4 Damage Zone Model

Chen, Wu and Yeh [50] have modelled the residual strength and stiffness of an impact damaged laminate without considering the exact nature of the damage. They state that the CAI failure is caused by the reduction of stiffness and strength in the damaged region. Their approach is to consider the damaged area as a material with reduced strength and stiffness, separate from the surrounding undamaged material. The laminate is then modelled using finite element analysis as a plate with an inclusion, normally circular, of damaged material which is assigned a stiffness and strength, of X% and Y% respectively, of the undamaged material. The analysis gives the stress concentration around the damage zone under load and indicates that the stress redistribution is a local phenomenon. Residual strength is more dependent on the severity of the damage rather than the extent of the load redistribution.

This approach is similar to the “equivalent hole” model in which impact damage is considered to be equivalent to a drilled hole in terms of its effect on residual strength. Both these approaches fail to take account of local buckling under compressive load caused by the displaced plies in the impact damaged zone.

5.7.5 Equivalent Hole

The equivalent hole model has been applied to tension after impact tests in section 5.5.3. By use of the average stress criterion with an appropriate characteristic length a_0 it was possible to find the radius of holes that would give the same residual tensile strength as the two levels of impact damage used experimentally. We may apply the same method to the compressive residual strength of the 21 ply coupons.

Soutis, Curtis and Fleck [41] made compression strength tests of 245 x 50 mm CFRP coupons

containing drilled central circular holes of diameter 4 mm - 25 mm. Anti-buckling guides were used to prevent macro buckling of the coupons and the compressive failures were initiated from the points of stress concentration at the edge of the hole.

The experimental results obtained by Soutis et al. for a 24 ply coupon of T800/924c with a $[(0/90_2/0)_3]_s$ lay-up are shown in Figure 5.19. The radius of the hole is given as a fraction of the coupon half width, and the residual strength as a fraction of the notched strength to the unnotched strength. Also plotted in Figure 5.19 is the curve of the average stress criterion with a characteristic length $a_0 = 1.6$ mm. Although Soutis et al. propose an alternative predictive model for the notched compressive strength, it can be seen that the average stress criterion provides a good fit to the data. It is not necessary to use an adjustment to the residual strength to account for the reduction in net section of a finite width plate containing a hole.

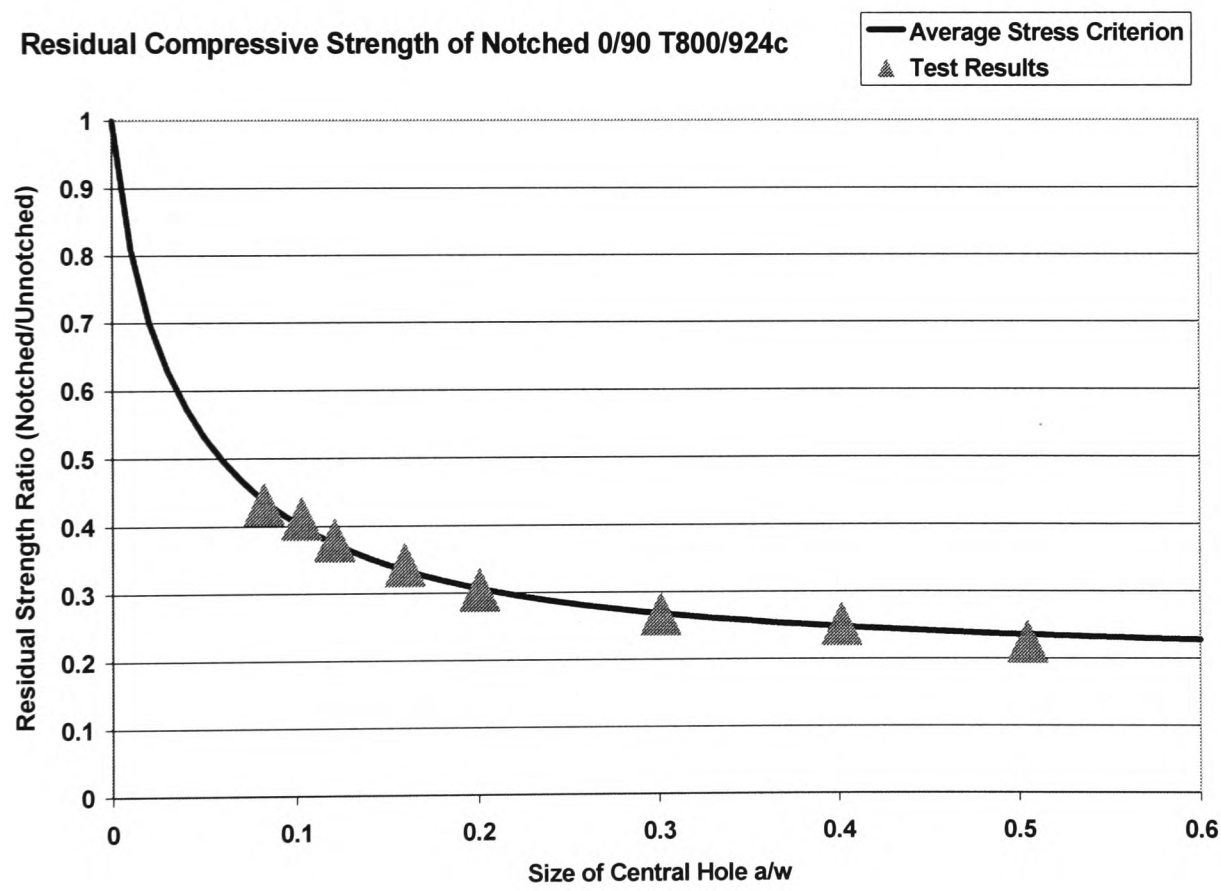


Figure 5.19

By reference to the experimental tests conducted by Soutis et al. it is possible to find the equivalent hole size for the two levels of impact damage tested. These are given in Table 5.8.

In compression, medium impact damage (BVID) is equivalent to a hole 12.8 mm in diameter. In tension the residual strength reduction is less and the damage is equivalent to a 9.8 mm

diameter hole. Severe impact damage is equivalent to a 33.2 mm diameter hole in compression and a 28.6 mm hole in tension. At the severe level of impact damage there is more fibre fracture damage and thus less difference between the strength reduction ratio in tension and compression.

Equivalent Hole Calculation for Compression After Impact

Impact Damage	Medium (BVID)	Severe
Incident Energy (J)	4.0	9.6
Indentation (mm)	0.075	0.9
Delamination Area (mm ²)	309	742
Nominal Failure Stress T_r (MPa)	-233	-173
Strength Reduction Ratio k_F	0.34	0.25
Equivalent Hole Diameter (mm)	12.8	33.2

Table 5.8

5.8 Strain Distribution at Impact Damage Zone

When an impact damaged coupon is loaded, the impact damage zone causes a stress concentration which initiates failure at lower levels than the undamaged state. Strain gauges were used to measure this stress concentration in an impact damaged coupon and evaluate the behaviour in both tension and compression loading.

The coupon was struck with a severe, low speed, impact of 9 J, resulting in an indentation of 0.43 mm, and an overall delamination area of 558 mm². Twelve strain gauges were used, six on either side of the coupon, on the coupon centre line next to the impact damage zone. Figure 5.3 shows the positions of the gauges.

Strain gauges on the coupon front face (the face struck by the impactor) were positioned right up to the edge of the impact indentation. On the back face the strain gauges were positioned up to the edge of the bulge behind the impact site.

The coupon was loaded in both tension and compression to a stress of ±151 MPa and the strain

gauge readings were recorded and are plotted in Figure 5.19. The position of the gauges is measured relative to the coupon edge. The point of impact was at the coupon centre, 42.5 mm from the edge.

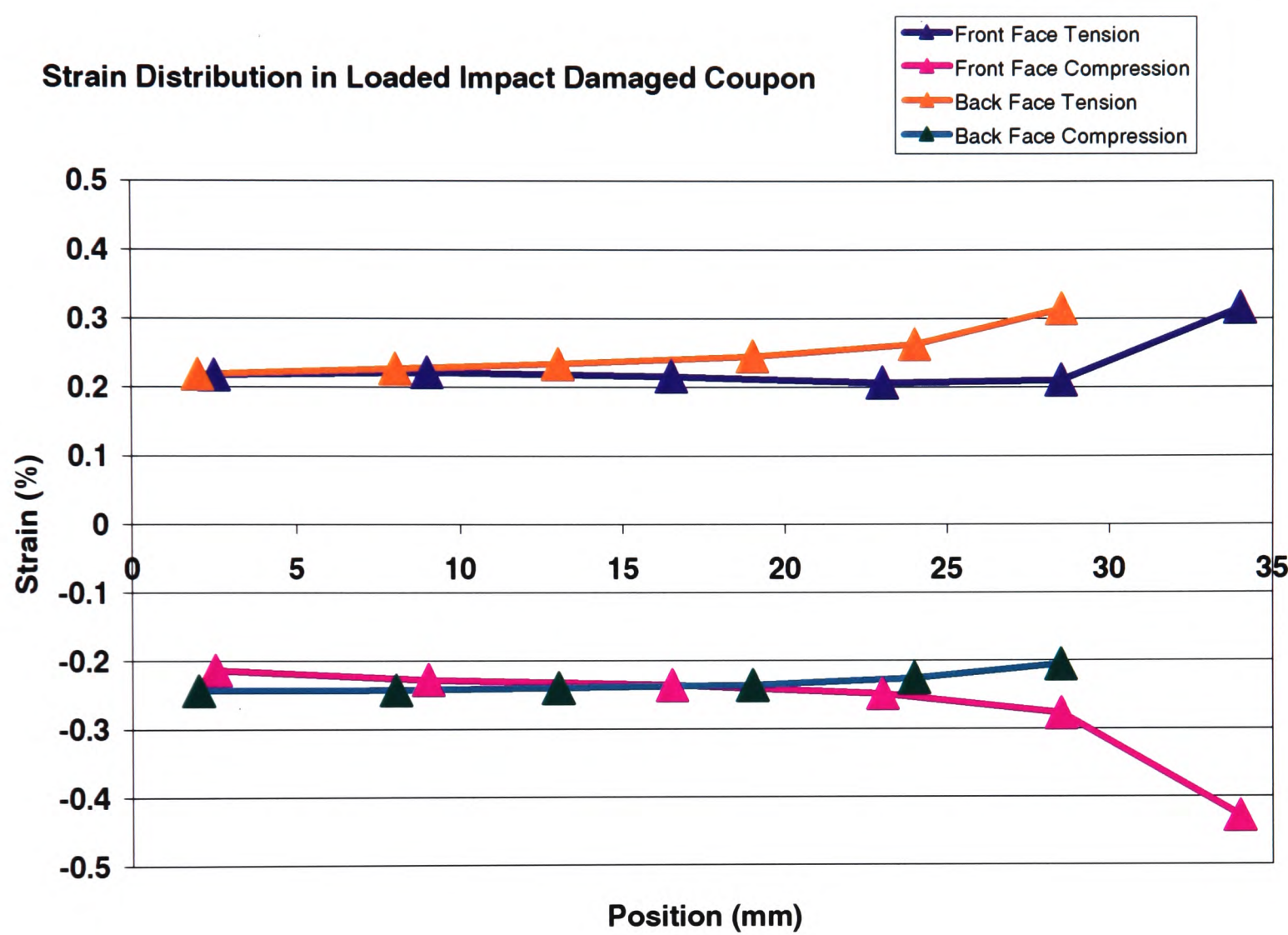


Figure 5.19

The difference in behaviour of the impact damage zone in tension and compression can be seen in Figure 5.19. Under tension there is a strain concentration of 1.45 at the edge of the impact damage zone on both the front and the back face. However in compression the behaviour is different. The front face strain concentration has increased to 1.9 while there is now a strain reduction at the edge of the impact damage on the back face of 0.9.

This difference in strains between front and back faces indicates a bending component in the strain distribution and this is confirmed by experimental observation. Under compressive load the displaced plies at the impact damage zone locally buckle outwards. This increases the compressive strain at the front face and decreases it on the back face.

5.9 Conclusions

Undamaged tension tests on the 21 ply, 140 x 85 mm coupon showed that the stress concentration at the end tabs was low (only 1.3) and the measured strength (688 MPa) approached that predicted by the small specimen tensile strength. In comparison the compressive strength for the coupon was only 47 % of the strength measured in a very short specimen (324 MPa compared to 692 MPa). This lower strength is due to structural buckling of the undamaged coupon.

However, in both tension and compression after impact tests, the coupon strength is considerably reduced and failure is initiated at the impact damage site. Impact damage considerably reduces the tension strength of the coupon: by approximately 50% for a 5 J impact; and by 70% at 10 J. Compared with the undamaged coupon strength, impact damage does not appear to have such a large proportional effect on the compression strength. However, the compression after impact (CAI) failures occur at stresses below the level of structural buckling of the coupon, and therefore it is more appropriate to compare the CAI strengths with the short specimen strength. If we do this we can see that an impact of 4 J has effectively caused a 60% reduction in compressive strength and a 10 J impact causes a 75% reduction.

Examining the stress distribution around the impact damage zone indicates the cause of the particularly low CAI strengths. Because the impact indentation leaves plies displaced and delaminated these locally buckle under a compressive load. This local buckling increases the compressive stress concentration, and thus failure is initiated earlier by a compressive shear failure.

Because at all levels of impact damage the compressive strength for the 21 ply coupon is lower than the tensile strength, the cyclic loads in fully reversed fatigue tests will be limited by the compressive strength.

CAI tests with 29 ply coupons show considerable scatter. Statistical variation in the level of damage resulting from impact (Figure 2.23) leads to even greater variation in the compression after impact strength.

The use of the equivalent hole model with the average stress criterion may be of use to

designers. Table 5.9 lists the equivalent hole diameters for drilled hole damage in T300/914 that correspond to the same residual tension or compression strengths as those occurring after impact damage from the two levels of impact energy considered.

Equivalent Hole Sizes

Damage Level	Medium (BVID ~5J)	Severe (~10J)
Incident Impact Energy (J)	4.6	9.5
Permanent Indentation (mm)	0.08	0.65
Delamination Area (mm²)	320	717
TENSION Equivalent Hole Diameter (mm)	9.8	28.6
Equivalent Hole Area (mm²)	75	642
COMPRESSION Equivalent Hole Diameter (mm)	12.8	33.2
Equivalent Hole Area (mm²)	129	866

Table 5.9

Chapter 6

Fatigue Behaviour

In this chapter the mechanism of fatigue failure in composites is discussed. In particular the problem of fatigue after impact damage is examined. Cyclic loading tests to failure were conducted on impact damaged coupons In order to quantify the effect of impact damage on the fatigue life of CFRP. Experiments were also made to investigate the mechanism of post impact fatigue failure.

6.1 Introduction to Fatigue in Composites

Fatigue and fracture processes in composites are quite different from those of metals. The mechanism of fatigue in metals depends on ductility. Metallic fatigue damage predominantly develops by the growth of cracks transverse to a tensile stress field through plasticity at the crack tip. In contrast composites are generally not ductile and are essentially linear elastic to the point of brittle failure. The mechanism of fatigue failure in composites is complex and still not well understood.

Some significant points may be made however:

- Fatigue damage in composites will develop as a distribution of cracks (rather than one single crack which grows to a critical size).
- The statistical nature of strength (of both fibres and matrix) allows progressive failure due to slight increments in strain distribution.
- Viscoelastic behaviour of the composite matrix may cause a varying strain distribution under cyclic loading.
- In general composites are fatigue resistant compared with metals.
- Cyclic compressive loading has been shown to be more damaging in composites than in metals.

A process of tensile fatigue in composites has been described by Matthews et al. [51]. The process is regarded as having two distinct stages. Initially matrix cracks form throughout the composite. This multiple matrix cracking builds up to a certain saturation level known as the characteristic damage state, or CDS. This constitutes the first stage. CDS results after sufficient

loading of the off-axis plies in a laminate. It is described as the state where the distance between two consecutive cracks in a particular ply is nearly uniform throughout a specimen. The secondary stage is characterised by the appearance of secondary cracks, the coalescence of previously formed cracks, delaminations and, ultimately, fibre fracture. This second stage is distinguished from the first by localisation, i.e. damage development at selected areas.

Bathias [52] made a general review of the problem of fatigue in composites. He observed that while compression-compression cyclic loading is damaging in composites, fully reversed compression-tension cyclic loading is more damaging. For compact specimens the no damage threshold for compression-compression fatigue is between 60% and 70% of the static strength of the material. However for tension-compression fatigue he observed that the no damage threshold was only 42% of the static strength.

Ratwani and Kan [53] developed a model to describe the process of damage growth in compression fatigue of a composite with a stress raiser such as a hole. Their model is based on the assumption that initial defects exist in composites between plies. These defects propagate due to the interlaminar stresses produced by applied fatigue loads. The mechanism of failure has been observed to be local progressive fatigue failure of the matrix near the stress raiser. This results in fibre splitting and progressive delaminations which lead to fibre buckling, and eventually causes laminate failure.

Impact damage in a composite plate is a stress raiser somewhat like a hole. However in Chapter 5 it has been observed that the stress state around an area of impact damage is much more complex than the simple case of a circular hole. It has also been observed that the stress distribution varies between tension and compression loading. In order to quantify the effect of impact damage on a typical aerospace CFRP material, fatigue after impact damage tests were conducted together with investigations into the mechanism of the fatigue process.

6.2 Fatigue After Impact

In this section experimental investigations into fatigue after impact damage are discussed, and the associated tests conducted for this thesis are described.

6.2.1 Literature Results

Work on fatigue after impact damage has generally focussed on experimental programmes to provide data for the qualification of aerospace components. The main issue of concern has been the problem of tools dropped onto the top surface of aircraft composite wings. In flight the upper wing skin is submitted to compression dominated fatigue loading. The concern has been that the BVID resulting from the tool drop may grow under the cyclic loads involved in flight to eventually result in a catastrophic failure.

Rosenfeld and Gause [54] conducted compression fatigue tests at between 1 and 5 Hz on graphite/epoxy quasi isotropic lay-up, 42 ply laminates with two types of stress raiser: an open hole; and hard body impact damage. Significant life reduction was found for compression fatigue with fully reversed tension-compression ($R = -1$) being the most damaging. The fatigue failure mechanism observed was a local progressive failure of the matrix near the stress raiser resulting in delamination and final failure by fibre buckling. However Rosenfeld and Gause suggest that the growth of matrix damage may not be an increase in the area of the same level of damage but actually an increase in the severity of damage in the original area. They found that for fatigue life reduction subvisual impact damage (~ 11 J) was at least as severe, and in general more severe, than a 6.4 mm diameter open hole. The subvisual damage would not propagate in the laminate at cyclic strains less than 0.3 %. This compares with the design ultimate strain used for thick skin primary aircraft structures of 0.4 %. Spectrum loading tests showed that Miner's rule for cumulative damage was extremely unconservative. A slight reduction in life was found when the average cyclic load frequency was increased from 1 Hz to 3 Hz, leading to the conclusion that compressive fatigue characteristics may be strain rate dependent.

Swanson et al. [55] also conducted compression fatigue tests on carbon/epoxy laminates with both open holes and impact damage as stress raisers. The tests were on standard Boeing quasi-isotropic plates and run at 7 Hz with a load ratio $R = 50$ (compression - compression). Temperature rise in the specimens was measured by thermocouple at this frequency of cycling and found to be negligible. CAI strength was observed to decrease with increasing impact energy but there was no significant growth in damage under cyclic loading and no measurable decrease in strength.

Clark and Van Blaricum [56] conducted spectrum loading fatigue tests on impact damaged carbon fibre composite coupons. They used 7 mm thick 300 x 100 mm plates with a 56 ply $[\pm 45/0_2]_7$ lay up. Impact damage was produced with a 12.2 J impact from a 12.7 mm diameter drop weight onto the coupons which were clamped with open window sizes of 20, 30 and 40 mm diameter. The coupons were loaded with grip plates which incorporated anti-buckling guides for the edges of the gauge area with dimensions 130 x 95 mm. The fatigue loading was based on a standard fighter aircraft sequence which was a compression dominated spectrum representing upper wing skin service conditions. The impact damage considered was found to reduce the static compression strength by up to 50%. With the fatigue threshold being approximately 60% of the static strength at any damage level, the design stress for cyclic loading of damaged components is therefore only 30% of the undamaged static failure stress. It was found that removing the low load cycles from a spectrum fatigue test did not affect the fatigue life, and therefore that only the high load cycles were propagating damage. The rate of damage propagation was found to be controlled by the damage width as measured by C-scan. Damage growth was perpendicular to the 0° fibre direction (the direction of applied load).

Beheshty & Harris [57] conducted fatigue tests on impact damaged CFRP at varying load ratios. The material tested was a 16 ply lay-up sequence of $[(\pm 45, 0_2)_2]_s$ of HTA/982A carbon-fibre. Before testing of specimens began impact damage of up to 2J was introduced with a drop weight. It was found that impact damage did not affect the fatigue response of the material in a purely tensile mode ($R = +0.1$). However the larger the compressive component of the cyclic load the greater the reduction in life, relative to the undamaged material.

Their experimental findings may be summarised as follows:

- Fully reversed tension - compression ($R = -1$) fatigue is the most damaging mode of cyclic loading for notched CFRP.
- In compression dominated cyclic loading BVID causes greater life reduction than the equivalent hole.
- Fatigue behaviour of CFRP is governed by high load cycles close to the static residual strength.
- Compressive fatigue response of CFRP may be strain rate dependent.

6.2.2 Damage Level

Two damage levels were used in fatigue tests. These levels were denoted as medium and severe impact damage, have been used in Chapter 5 for equivalent hole calculations, and corresponded to incident impact energies of approximately 5 and 10 Joules. Table 6.1 details measurements of the two levels of damage. The medium level of damage corresponds to BVID (barely visible impact damage) whereas the severe level may be classed as VID (visible impact damage).

Damage Levels in 21 Ply Coupons

	Medium (BVID ~5J)	Severe (VID ~10J)
Mean Incident Impact Energy (J)	4.62	9.54
Standard Deviation (J)	0.38	0.57
Mean Incident Impact Velocity (m/s)	6.26	8.99
Standard Deviation (m/s)	0.26	0.27
Mean Permanent Indentation (mm)	0.08	0.65
Standard Deviation (mm)	0.02	0.20
Mean Delamination Area (mm ²)	320	717
Standard Deviation (mm ²)	30	113

Table 6.1

6.2.3 Load Ratio

Bathias [52] observed that compression - tension fatigue was the most damaging cyclic loading mode for composites in general. Rosenfeld and Gause [54] also found this to be true for impact damaged CFRP. Beheshty & Harris [57] found that the compressive component of cyclic load needed to be significant for observation of any reduction in fatigue life due to impact damage. Consequently the fatigue tests conducted for this thesis were under fully reversed compression and tension cyclic loading, with a load ratio of $R = -1$.

6.2.4 Frequency

The fatigue tests were conducted with a sinusoidal load signal at 5Hz. It has been suggested that higher frequencies than 5 Hz may cause heating in the specimen. This could result in undesirable affects on the composite mechanical properties and the fatigue behaviour.

Effects of frequency on the fatigue lifetime of some composites have been measured although Joseph et al. [58] observed that frequency of loading does not influence the lifetime of carbon/epoxy laminates. By contrast, increasing frequency increases the lifetime of glass fibre/epoxy laminates.

6.2.5 Testing Machine

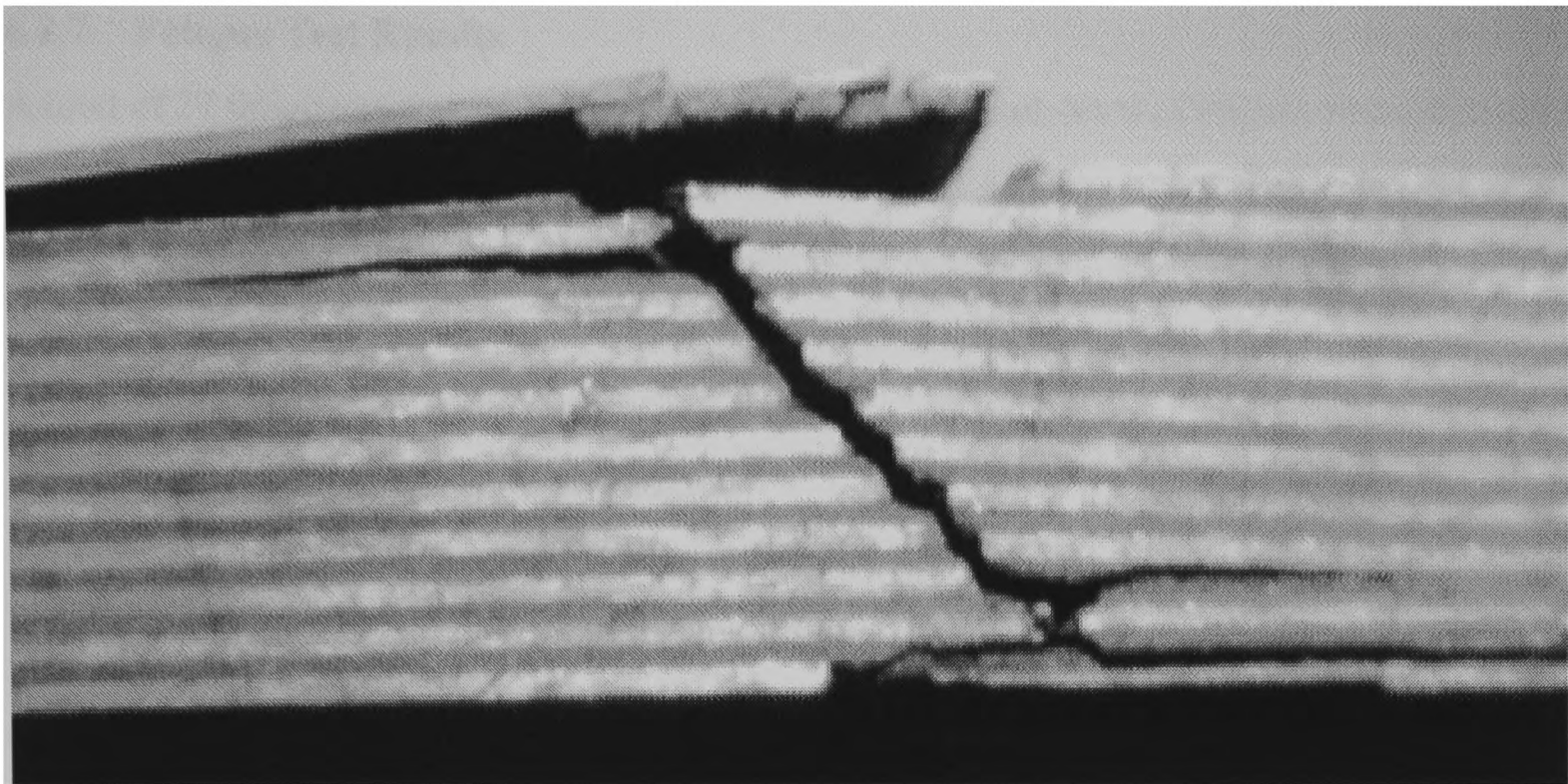
Fatigue tests were conducted on a Dartec servo-hydraulic testing machine (Chapter 5, 5.2.2). For cyclic loading tests the feedback computer control loop gains were adjusted in operation in order to give stable response of the machine.

The point of specimen failure was ascertained by the use of computer controlled “trips”. Both displacement limit trips and the servo valve control signal limit trip were used to detect failure. It was found that carefully set displacement limit trips could be used to preserve the specimen just after the point of failure. If the trips were insensitively set then the unresisted displacement of the machine piston after failure would be such that the coupon would be completely destroyed. However it was found that “spikes” in the displacement measuring circuit could prematurely trigger the displacement trips. Therefore it was usually impractical to leave the machine unsupervised on long fatigue tests using closely set displacement trips, and the less sensitive servo valve trip was used instead.

The Dartec trips were set so that, when triggered, they would turn off the machine pressure and also stop the function generator. This prevented further damage after failure and caused the number of cycles to failure to be recorded on the computer control screen.

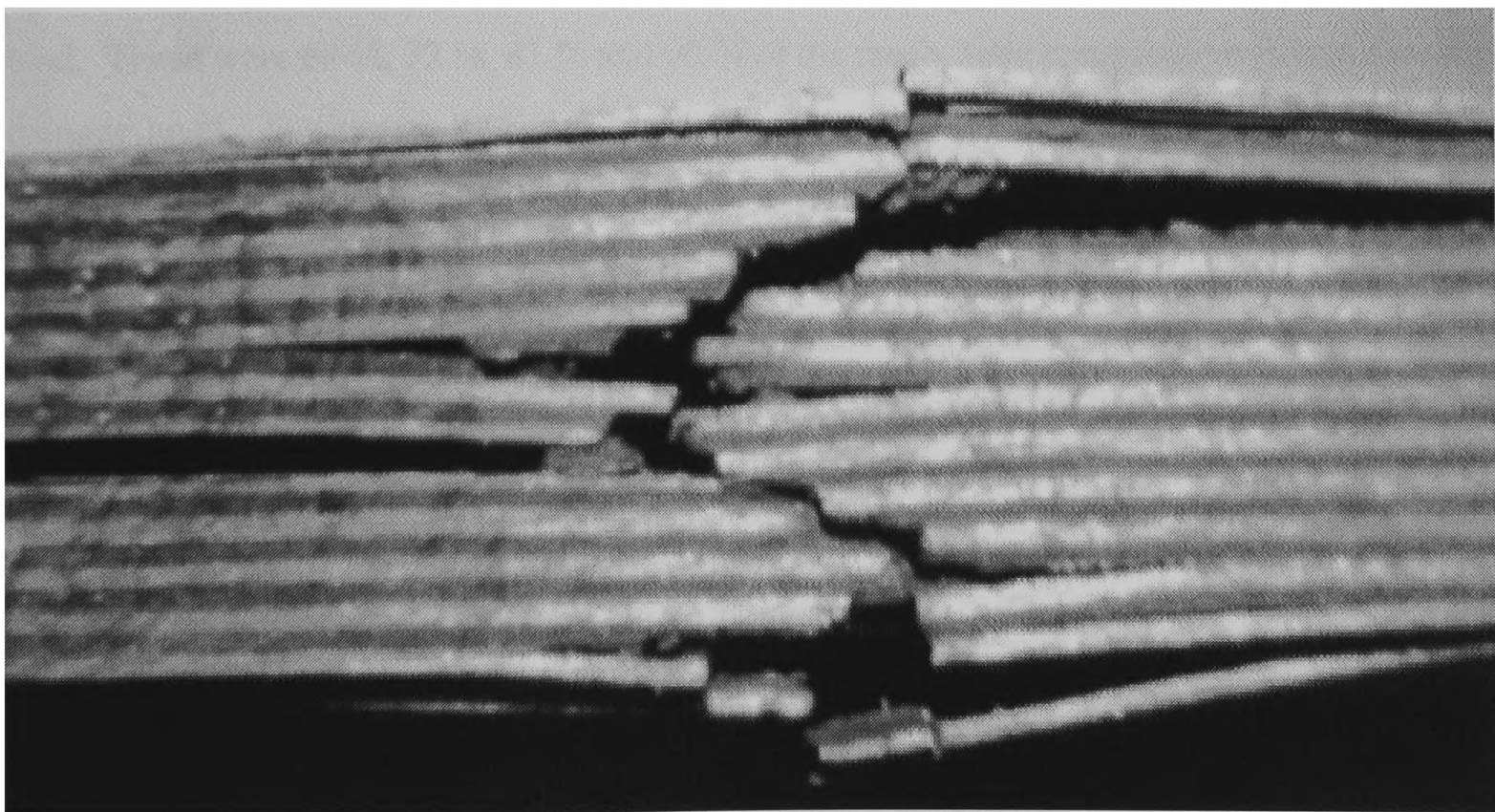
6.2.6 Failure Mode

Final failure of the coupons under fatigue loading was always a compressive failure initiated at the impact damage site. The final failure crack would propagate perpendicularly across the coupon to both edges. Typical failures can be seen in Figures 6.1 and 6.2 which show the edges of failed coupons. The failure mode is very similar to that observed in compression after impact tests of the 21 and 29 ply coupons (Figures 5.11, 5.12, 5.16 and 5.17).



Fatigue after Impact, $\pm 35\text{kN}$ ($\pm 147\text{ MPa}$) Failure @ 13000 cycles
Severe Impact Damage (10.42J, Indentation 0.74mm, Delamination Area 838 mm²)

Figure 6.1



Fatigue after Impact, $\pm 40\text{kN}$ ($\pm 168\text{ MPa}$) Failure @ 18000 cycles
Medium Impact Damage (4.98J, Indentation 0.12mm, Delamination Area 354 mm²)

Figure 6.2

6.2.7 Fatigue Test Results

A total of 27 fatigue tests were conducted, 13 with the medium level of impact damage, and 14 with severe damage. The test results are detailed in Table 6.2. With each fatigue test requiring the following stages it was possible to conduct up to two fatigue tests per week.

- i) Diamond saw coupon from CFRP sheet
- ii) Impact coupon in gas gun
- iii) Measure indentation
- iv) Ultrasonic C-scan to determine overall delamination area
- v) Prepare and glue aluminium tabs to coupon
- vi) Align grips and mount coupon on testing machine
- vii) Set testing machine load limits and trips
- viii) Conduct test
- ix) Record cycles to failure
- x) Clear testing machine of broken coupon

For the medium level of damage, fatigue tests were conducted at four different levels of cyclic load. These were 68 %, 72 %, 81 % and 90 % of the mean static compressive strength for that damage level. For severe impact damage, the fatigue tests were conducted at six different levels: 67 %, 73 %, 85 %, 87 %, 90 % and 97 % of the static compressive strength, respectively.

The shortest fatigue lifetime observed was 2 cycles, for a coupon with severe damage, tested at a 97% load. The longest test was of a coupon with severe damage, tested at a 67 % load. This test lasted 781321 cycles before failure and took 43 hours.

The fatigue results are presented graphically in Figure 6.3. Each fatigue after impact damage test is plotted as a single point showing the nominal cyclic stress and the number of cycles to failure. Nominal cyclic stress is the average applied stress defined as the maximum and minimum load divided by the undamaged coupon cross sectional area. Best fit (least squares) linear - log trendlines have been fitted to the experimental results and these show the increase in lifetime observed with reduction in the magnitude of the applied loading. For a given stress level the medium damage coupons will survive a larger number of cycles before failure.

The results are also presented in Figure 6.4 with the lifetimes plotted against the applied stress in the form of a fraction of the static strength. It can be seen that, in reference to the static compression strength, medium impact damage gives a greater reduction in fatigue life.

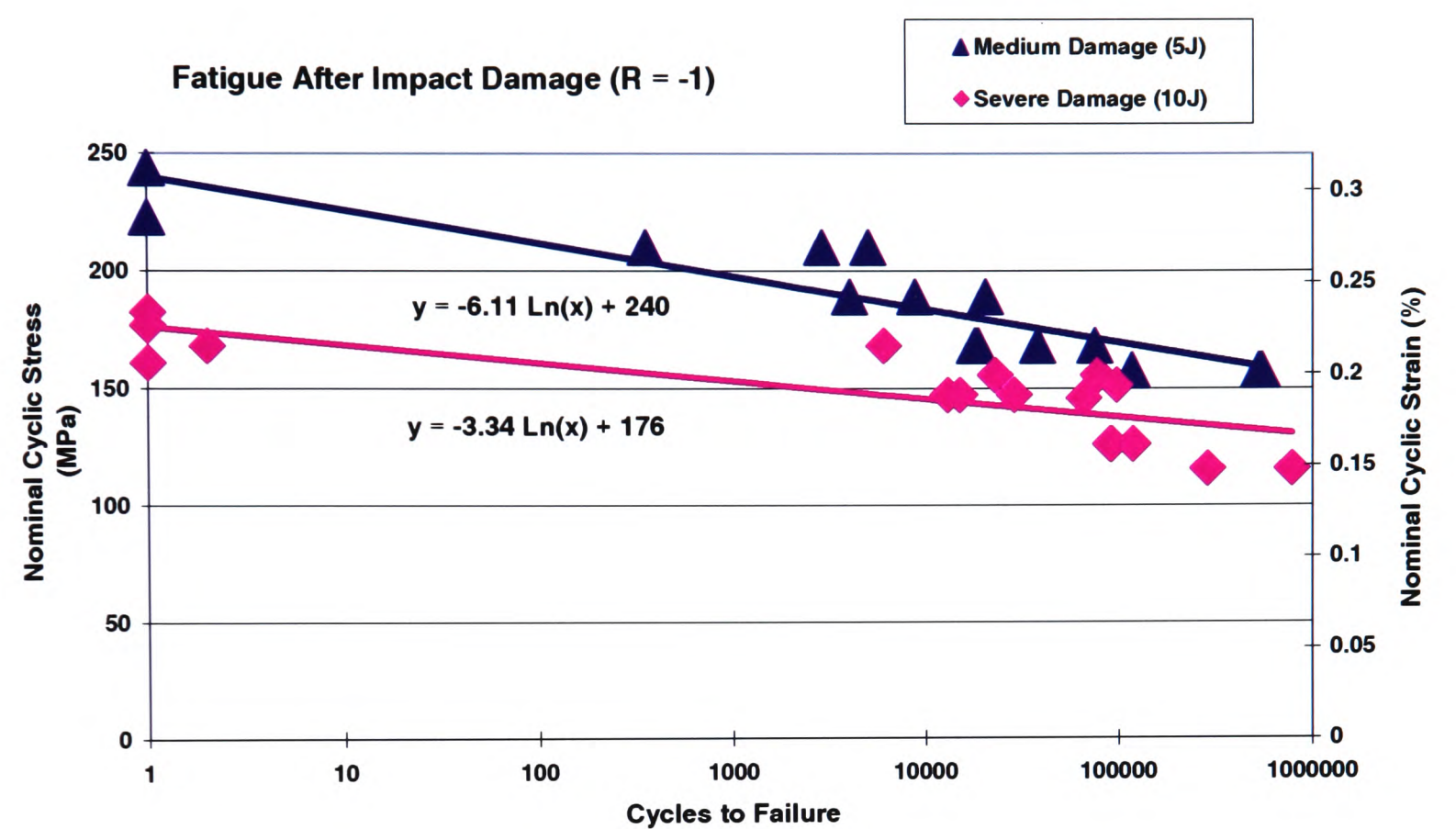


Figure 6.3

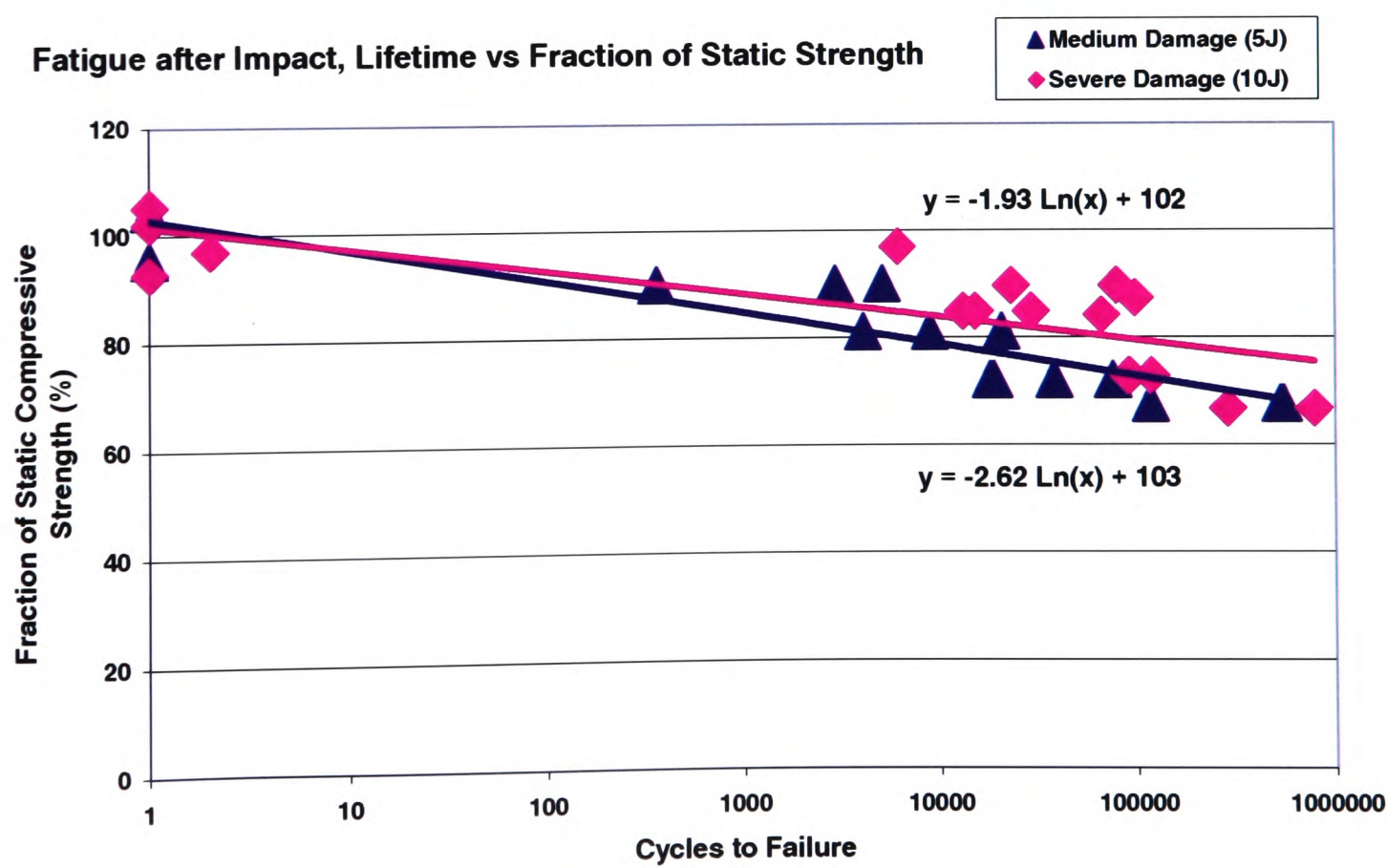


Figure 6.4

Fatigue After Impact Damage Test Results

Impact Level	Impact Energy (J)	Inden-tation (mm)	Delam. Area (mm ²)	Cyclic Load (± kN)	Cyclic Stress (±MPa)	Cyclic Strain (± %)	Fraction of Static Strength	Cycles to Failure
Medium	4.20	0.07	286	38	158	0.201	68 %	530239
	4.59	0.08	330	38				116794
	4.78	0.06	348	38				549340
	4.98	0.12	354	40	168	0.215	72 %	18000
	5.01	0.06	350	40				75359
	5.05	0.10	287	40				18755
	5.08	0.08	358	40				37947
	4.35	0.08	270	45	189	0.242	81 %	20481
	4.52	0.07	274	45				8841
	4.83		330	45				4039
	~ 5.00	0.11	310	50	210	0.269	90 %	358
	4.44	0.08	363	50				5052
	4.68	0.08	318	50				2911
Severe	9.60	0.63	805	28	116	0.148	67 %	286950
	9.70	0.56	893	28				781321
	9.73	0.70	724	30	126	0.161	73 %	90000
	9.88	0.85	728	30				117645
	9.97	0.65	861	30				91361
	9.85	0.73	740	35	147	0.188	85 %	15016
	9.11	0.59	642	35				65561
	10.29	0.93	881	35				28753
	10.42	0.74	838	35				13000
	~9.00	0.43	558	36	151	0.194	87 %	97238
	8.84	0.63	571	37	155	0.199	90 %	78181
	9.61		645	37				22776
	9.44	0.90	787	40	168	0.215	97 %	2
	9.58	0.22	580	40				6032

Table 6.2

6.2.8 Fatigue Life Models

Various attempts have been made to develop predictive models for the fatigue lifetime of composites. The models are generally empirical, or semi-empirical, in nature and use a power law or logarithmic law that matches experimental stress/life data.

To predict the fatigue life of composites Harris et al. [59] have proposed a “constant-life model”. They observe that the strength-life equal-rank assumption is not valid for composites; that is, if a series of materials are ranked according to their tensile strengths, then their fatigue responses will not necessarily be similarly ranked. The constant-life model is a parametric analysis that may be used as a method of life prediction. A limited number of tests should provide data which may then be sufficient to predict lifetimes for other loading conditions. It appears that for the same fatigue lifetime there is a general relationship between the alternating and mean components of stress of the form:

$$\frac{\sigma_{alt}}{T_X} = f \left(1 - \frac{\sigma_m}{T_X} \right)^u \left(\frac{C_X}{T_X} + \frac{\sigma_m}{T_X} \right)^v \quad 6.1$$

where σ_{alt} is the alternating component of stress and σ_m is the mean stress as follows:

$$\sigma_{alt} = \frac{1}{2}(\sigma_{max} - \sigma_{min}) \quad 6.2$$

$$\sigma_m = \frac{1}{2}(\sigma_{max} + \sigma_{min}) \quad 6.3$$

T_X and C_X are the monotonic tensile and compressive strengths respectively. The constant life equation gives a bell shaped curve showing the dependence of the alternating stress, to give a particular fatigue lifetime, on the mean stress. The exact shape of the bell curve for a particular lifetime is controlled by the parameters f , u and v . While f was initially treated as being independent of the particular fatigue life considered, in later papers Harris et al. [60], [61] consider a linear function of log life to better fit their experimental data. Linear functions of log life were also found to be appropriate for the exponents u and v :

$$f, u, v = A + B \log N_f \quad 6.4$$

where N_f is the number of cycles to failure. Harris et al. found that their constant life model gave a good fit to a large database of fatigue tests of undamaged composite components at varying load ratios and mean stress levels.

Beheshty and Harris [60] examined the application of the constant life model to fatigue after impact damage of CFRP coupons. Fatigue tests at four different load ratios gave results showing reasonable agreement with the model. Fatigue tests for this thesis were conducted at a single load ratio ($R = -1$) and consequently it is not possible to fully evaluate the suitability of the constant - life model to these results. However, some of the model parameters may be calculated and discussed for the fatigue data obtained here.

Since the mean stress in the fatigue tests was zero the constant life model may be simplified to:

$$\frac{\sigma_{alt}}{T_X} = f \left(\frac{C_X}{T_X} \right)^v \quad 6.5$$

This indicates that the level of alternating stress that can be sustained as a proportion of tensile strength, is a direct function of the ratio of compressive strength to tensile strength. Substitution of a log-life form of f and v indicates that the stress/life behaviour should have the following relationship:

$$\frac{\sigma_{alt}}{T_X} = (A + B \log N_f) \left(\frac{C_X}{T_X} \right)^{(C + D \log N_f)} \quad 6.6$$

Fatigue test results for the two levels of damage are plotted as the cyclic load fraction of static tensile strength against the number of cycles to failure in Figure 6.5. It can be seen that the ratio of static compressive strength to tensile strength (C_X/T_X) is a very significant factor in the fatigue behaviour. The coupons with severe damage ($C_X/T_X = 0.98$) were under cyclic stresses which were close to the static tensile strength as well as the static compressive strength, and the fatigue lifetimes were reduced considerably relative to the medium damaged coupons ($C_X/T_X = 0.63$). Medium damaged coupons were loaded well below their static tensile strength and it can therefore be presumed that the behaviour was compression dominated.

Equation 6.6 is used to plot best fit curves to the data in Figure 6.5. The fit for medium damage fatigue results is good but is less accurate for the severe damage. The appropriate coefficients A , B , C and D in the parameters f and v have been evaluated as follows:

$$f = 1.00 - 0.05 \log N_f \tag{6.7}$$

$$v = 1.00 + 0.04 \log N_f \tag{6.8}$$

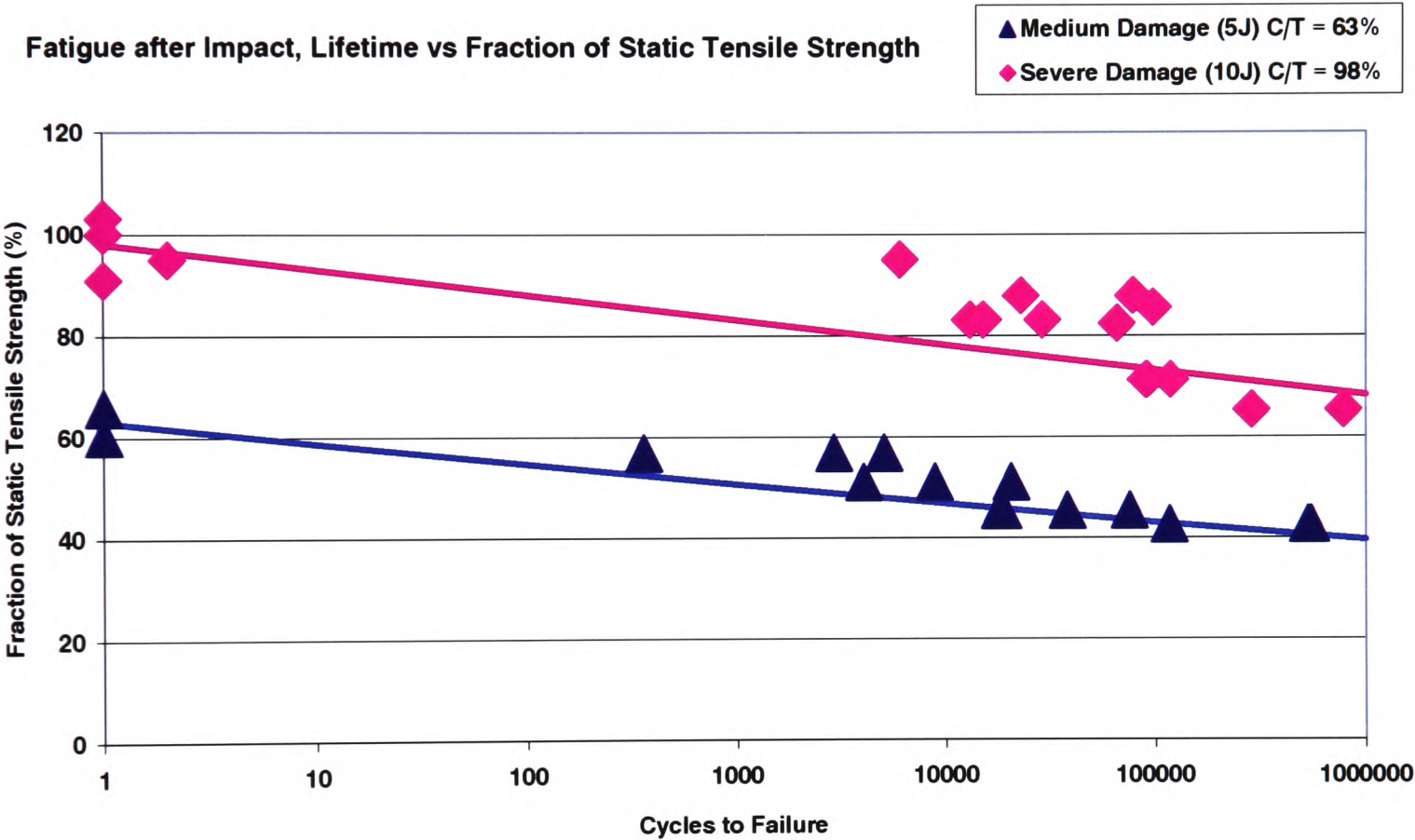


Figure 6.5

The stress - life relationship suggested by Harris et al. (Equation 6.6) provides an approximate fit to the fatigue test results obtained if appropriate coefficients are used. Fawaz and Ellyin [62] have proposed a semi-empirical model for prediction of fatigue failure of fibre reinforced materials. Their model assumes a linear - log relationship between applied cyclic stress σ and number of cycles to failure N_f at that stress.

Linear - log best fit trendlines have been plotted for the fatigue data in Figures 6.3 and 6.4 and give a reasonable match. However all trendlines are still only rough approximations to the experimental data. A more accurate fit to the data, suitable for design, can perhaps be obtained with an “S” shape curve, or the fatigue life diagram of Talreja [63].

The fatigue life diagram proposed by Talreja [63] is represented in Figure 6.6. Cyclic stress (or maximum strain in the first load cycle) is plotted against the logarithm of the number of cycles to failure. The fatigue stress-life behaviour of composites is divided into three regions.

Region I is the scatter band of static failure, which extends horizontally, and is dominated by the non-progressive (i.e. cycle independent) fibre failure mechanism. Region II consists of the sloping scatter band, representing the dominant, progressive (i.e. cycle dependent) damage mechanism. Region III is the fatigue limit. This is the lower limit below which the rate of progression of the damage mechanism is too low to be measured, or the mechanism is effectively arrested.

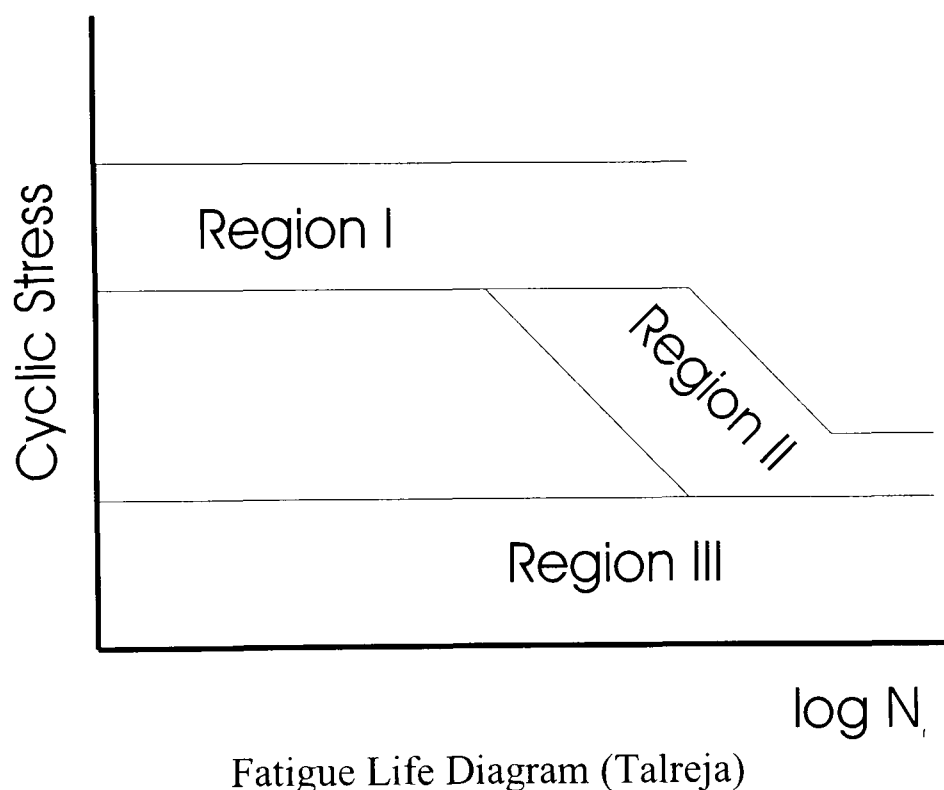


Figure 6.6

Beaumont's [64] measurements of the rate of increase of a "damage parameter" (proportional to the reduction in specimen modulus) support Talreja's fatigue life diagram. Beaumont observed three regimes of behaviour in tensile fatigue tests of CFRP:

- i) an apparent threshold stress below which no damage was detectable
- ii) between this threshold stress and cyclic stresses approaching the tensile strength of the composite laminate the damage rate follows a power law relationship:

$$\frac{dD}{dN} = C \left(\frac{\sigma_{alt}}{T_X} \right)^\alpha \quad 6.9$$

- iii) at stresses close to the tensile strength T_x the damage rate is much higher than can be explained by the power law. Failure is now essentially static.

The fatigue life diagrams proposed by Talreja provide a better fit to the fatigue test results obtained here than single power law, or log-life, S-N curves. In Figures 6.7 and 6.8 Talreja diagrams are fitted to the experimental scatter obtained for the fatigue after impact damage tests with medium and severe levels of damage.

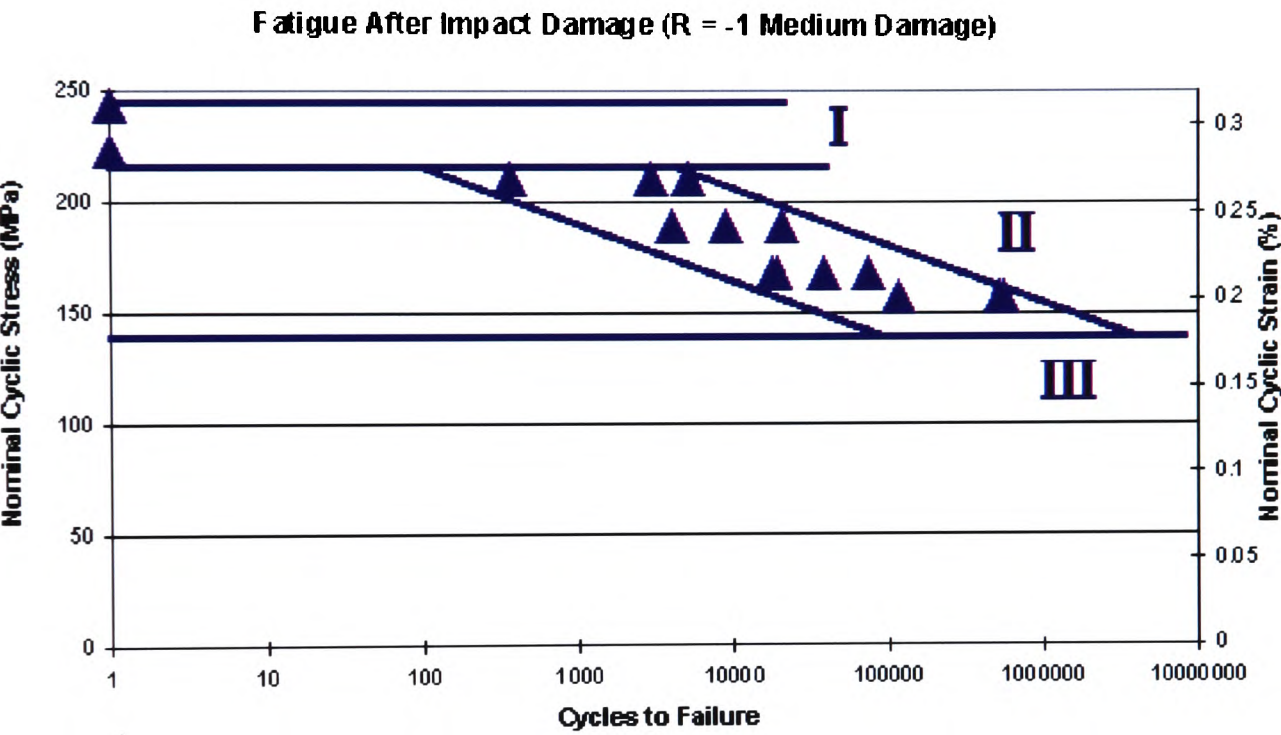


Figure 6.7

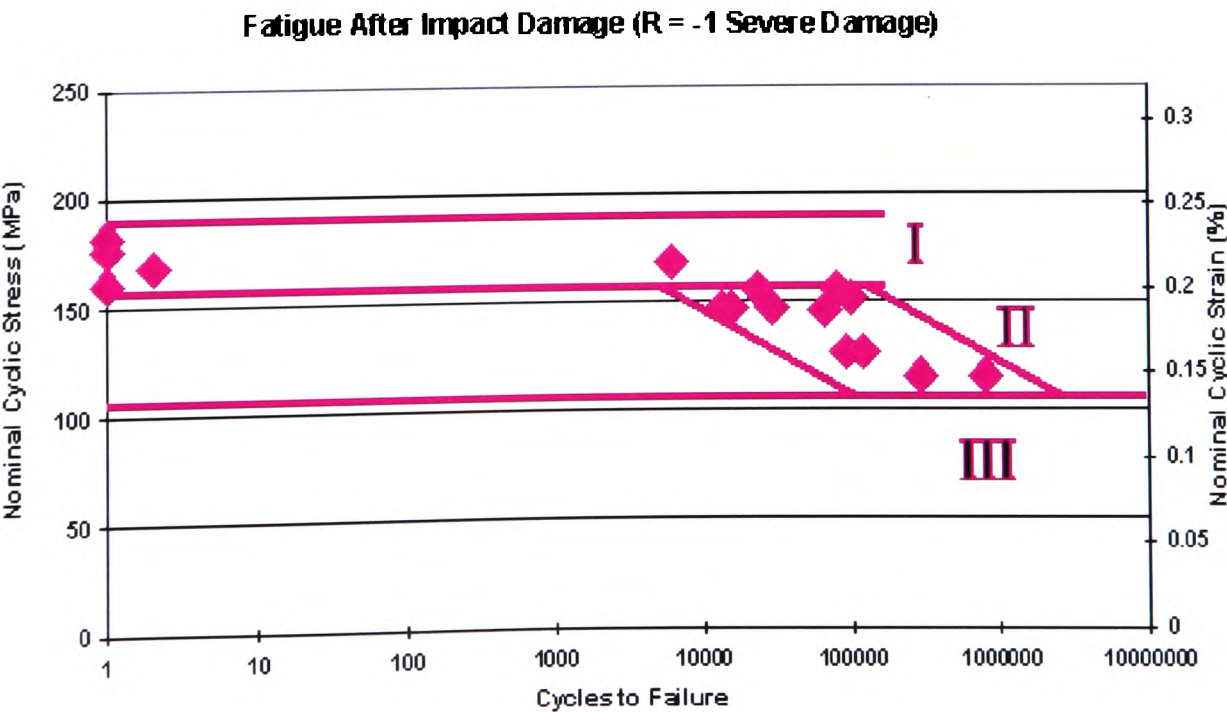


Figure 6.8

6.3 Growth of Damage During Fatigue Lifetime

During cyclic loading of impact damaged CFRP there appears to be a development, or growth, of damage which results in a reduction of the residual strength of the specimen. After a number of load cycles, the residual strength drops to the level of cyclic loading at which final fatigue failure occurs.

The mechanism of damage progression in CFRP is still poorly understood. With the aim of investigating the process of damage development that leads to fatigue failure in impact damaged coupons observations and measurements of coupons were made during the fatigue process. Three methods were used to monitor damage progression. These were:

- Visual observation
- Ultrasonic C-scan
- Continuous monitoring of coupon extension during cyclic loading

6.3.1 Literature Results

Behesty and Harris [57] used ultrasonic C-scanning to monitor the growth of delamination area under cyclic loading in impact damaged CFRP laminates. The material tested was a 16 ply laminate of $[(\pm 45, 0_2)_2]_s$ lay-up CFRP subjected to low velocity impact. Under compression fatigue tests with a load ratio $R = +10$ there was no measurable change in the delamination area measured by C-scan. However, the C-scan images of tension-compression fatigue with a load ratio of $R = -0.3$ showed some limited growth of delamination.

Changes in modulus of a composite component under fatigue have been reported by a number of authors. Brønstedt et al. [65] have proposed a model which suggests a linear decrease in modulus with fatigue cycles N . This model was supported by experimental measurements of changing modulus in constant load amplitude tensile fatigue tests of glass fibre/epoxy specimens.

Beaumont [64] also proposed using changes in stiffness during cyclic loading to monitor the accumulation of fatigue damage in composites. His tensile fatigue experiments on quasi-isotropic CFRP coupons also showed an approximately linear reduction in modulus with number of cycles N .

6.3.2 External Damage Development

During fatigue tests external observations were made of the progression of fatigue damage.

In each fatigue cycle the compressive load would caused local buckling and displacement of the impact damage site. This was visible from the front and back faces as a “fluttering” movement at the speed of fatigue cycling (5 Hz) . The amplitude of movement did not significantly change throughout the fatigue tests. It was in the range of 1 to 3 mm but was greater in those coupons with “severe” impact damage.

Some changes were visible externally during fatigue tests. The first change generally seen was the bridging of two cracks on the coupon front face across the impact indentation. Usually a surface compression crack was visible on each side of the impact indentation. In the tension phase of the fatigue cycle, these cracks could be seen to open very slightly and after a few hundred cycles the two cracks would join across the impact indentation.

On the back face some small changes were also visible in the early stages of a fatigue test. The delamination “blister” around the back face bulge would very gradually extend. However this growth was limited and did not necessarily continue throughout the test.

In the later stages of a fatigue test some larger, and discrete, changes were observed. Strips of surface plies would delaminate from the lower plies and buckle, during the compression phase of the cyclic loading. This delamination buckling would occur predominantly on the back face, though occasionally on the front face, close to the impact damage site, as well. Significant levels of this delamination buckling would develop in the final stages of the fatigue test and would indicate that final failure was near.

Although incremental changes were observed during the fatigue test the final failure was always a brittle fracture propagated across the full specimen width. The changes observed could not be directly related to the final failure mechanism (a compressive shear failure) other than by assuming that they were part of a gradual mechanism of stress redistribution. It would appear that gradual, incremental, failures of matrix and fibre around and within the impact damage zone would lead to sufficient redistribution of stress so that eventually final fracture occurred. The final failure mechanism was essentially the same as that observed in the static CAI tests.

6.3.3 Progressive C-Scans

Figure 6.9 shows C-scans taken at intervals during a fatigue test. The colour density of the images is not significant, they are intended only to show the area of delamination.

This experiment aimed to determine if the area of delamination measured by C-scan would grow as a function of fatigue cycles. The initial impact damage level was severe (incident impact energy 8.8 J, indentation 0.63 mm and initial delamination area of 571 mm²). Cyclic loading was at a frequency of 5 Hz with an amplitude of ± 37.5 kN (158 MPa). At intervals the coupon was removed from the testing machine jaws and C-scanned. It was then replaced in the machine and the test was continued.

It can be seen from the C-scan images that there was very little growth of overall delamination area before the final failure occurred. However some incremental changes are visible in the scans.

In the first 3000 cycles the delamination blister “filled out” slightly and this can be seen in the change from a very concave diamond shape of overall delamination to a more regular diamond shape. There was then little change observed externally until approximately 13000 cycles. At this point some delamination buckling of strips on the back face occurred. This can just be seen in the C-scan image taken shortly afterwards. Further delamination buckling of strips developed on the back face, on both sides of the impact site though it can be seen that there was no significant increase in the overall delamination area.

Final failure occurred at 78181 cycles. The testing machine was paused temporarily for adjustment at a constant compressive load of - 37.5 kN and the coupon failed within ten seconds.

6.3.4 Change in Coupon Modulus

Observation of fatigue tests to failure suggested that the fatigue mechanism and failure mode may involve a reduction in the overall stiffness of the coupon. Since the fatigue tests were conducted at constant load cycling a reduction in stiffness would increase the overall strain until a critical failure strain was reached. To investigate this hypothesis a clip gauge displacement

C-scans at Intervals During Fatigue Test

Severe Impact Damage

(Incident energy 8.8 J, Indentation 0.63 mm, Delamination area of 571 mm²).

Cyclic Load ± 37.5 kN (± 158 MPa, R = -1)

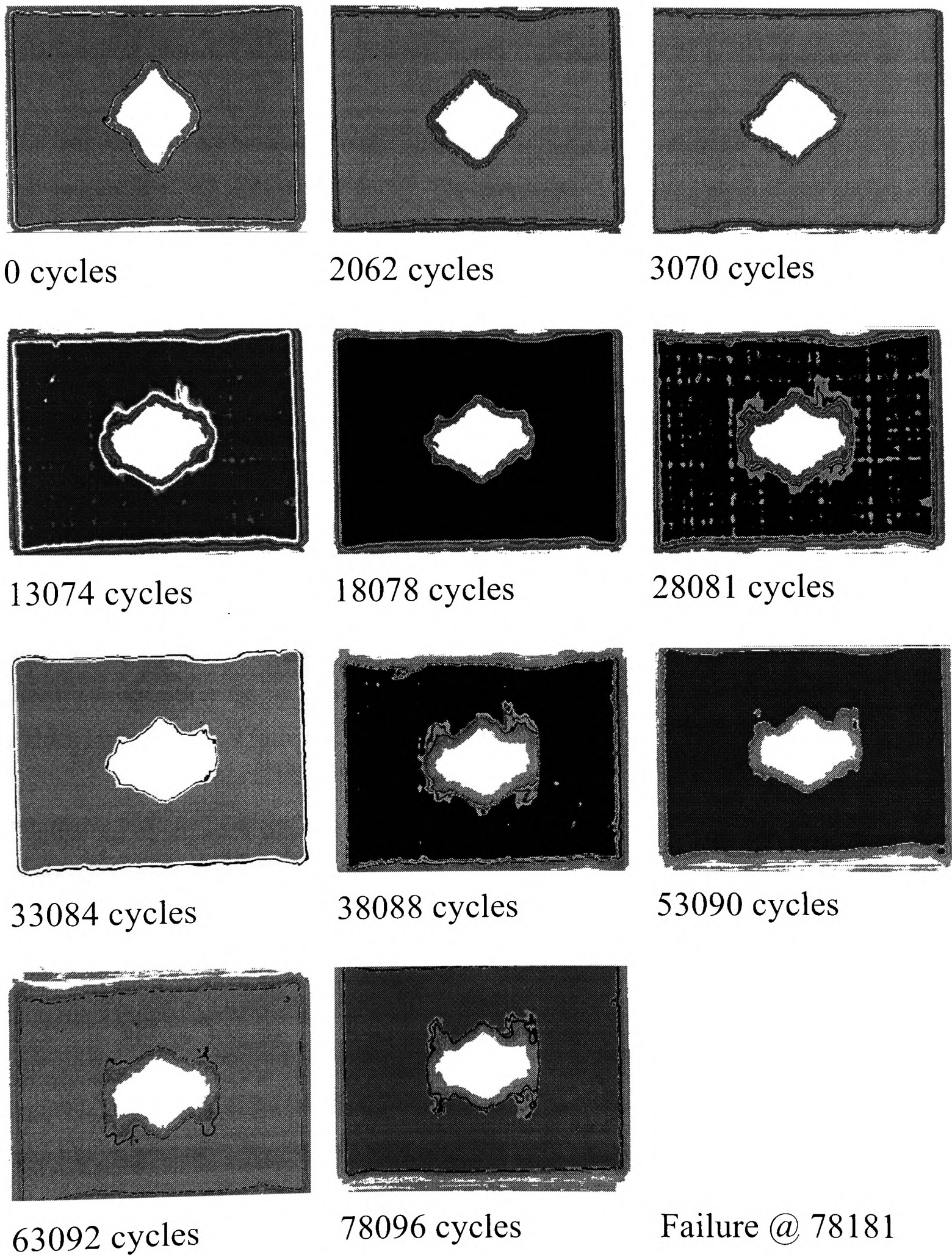


Figure 6.9

transducer was used to monitor the strain range in a coupon during two fatigue tests.

The coupons tested (denoted here as A and B) had been submitted to severe impact damage ($\sim 10\text{J}$). The resulting damage was measured as an indentation of 0.43 mm and an overall delamination area of 558 mm² for coupon A, and 0.59 mm and 642 mm² for coupon B.

The fatigue tests were conducted under constant load cycling, coupon A at $\pm 36\text{ kN}$ ($\pm 151\text{ MPa}$) and coupon B at $\pm 35\text{ kN}$ ($\pm 147\text{ MPa}$). The average strain experienced by the coupons was measured with the displacement transducer. Figure 6.8 shows how the maximum and minimum strain in each load cycle changed under the constant load cycling due to changes in the coupon modulus. Final failure was at 97238 cycles for coupon A, and 65561 cycles for coupon B.

The “spikes” seen in Figure 6.10 were instrument (voltage) spikes and do not represent real strains. It can be seen that for the first 50 % of the fatigue life of coupon A there was very little change in the overall strain measured in the coupon. However in the final half of the fatigue life gradual development of damage caused a reduction in the overall modulus of the coupon and the strain increased. In contrast the damage development in coupon B occurred later in the fatigue life and much more rapidly. In this case there was a sudden development of significant delamination buckling around the damage zone at approximately 58000 cycles. This caused the sudden large reduction in compression modulus.

Continuous observation of some fatigue tests revealed that the damage development in many of the tests of impact damaged coupons featured large, discrete, and sudden local failures such as delamination buckles. Generally these failures were first heard as audible cracking and then observed visually. Such discrete events were, presumably, caused by a gradual background incremental development of distributed damage.

Figure 6.10 shows that the tensile (maximum) strains in coupons A and B increased much less than the compressive (minimum) strains. This result most probably occurs because of the progression of the delamination “blister” on the coupon back face.

Both coupons A and B were impact damaged to the severe level and therefore the coupons were

being cycled at load limits close to both the static tensile and compressive strengths. Consequently tensile behaviour is also likely to be significant in the fatigue progression. The lesser, but significant, increase in the tensile strains shows that there must have also been an increase in fibre breakage.

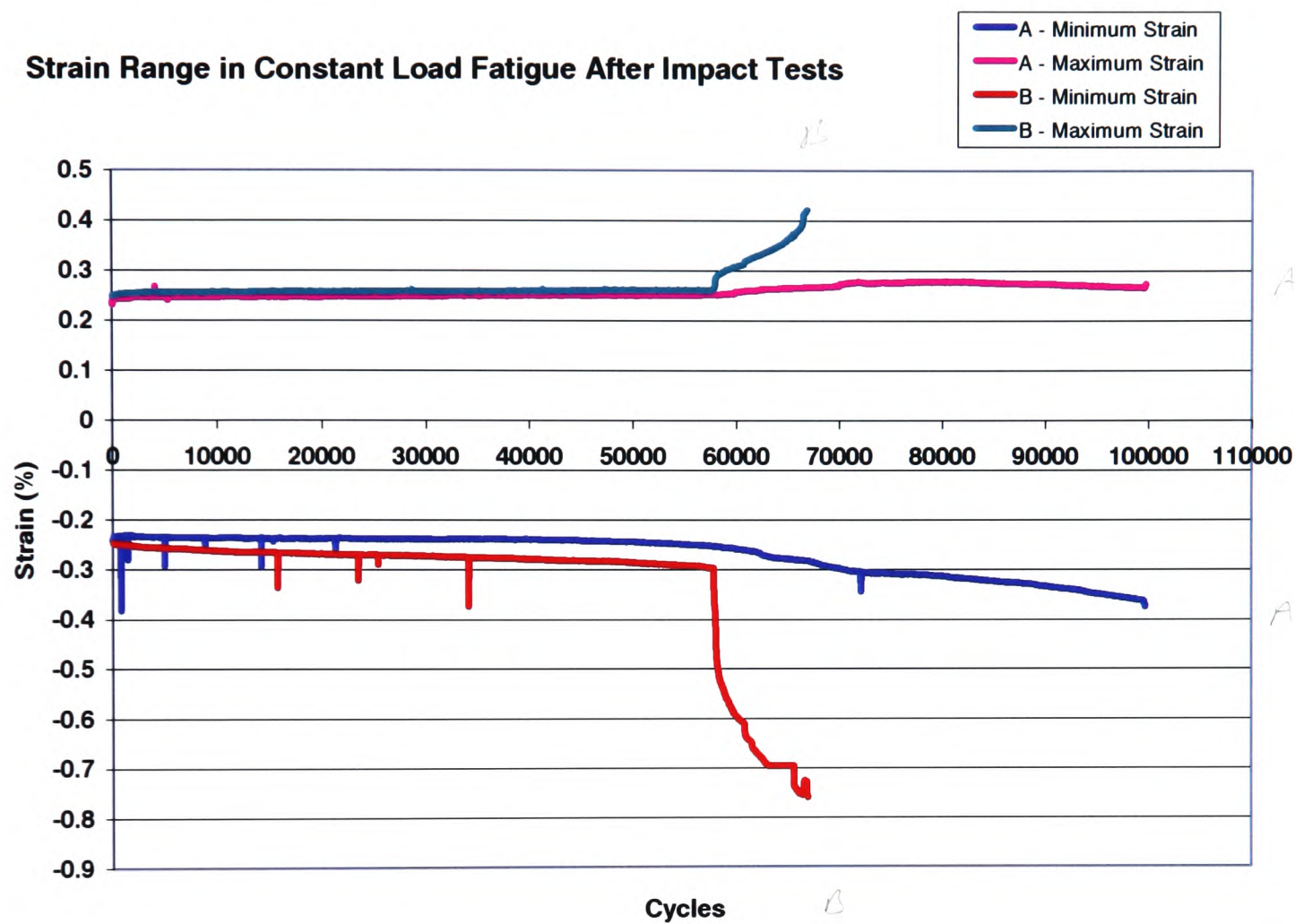


Figure 6.10

6.3.5 Interpretation of Results

It can be seen from Figure 6.10 that a straightforward linear decrease in modulus is not an appropriate model for the compression fatigue of CFRP coupons with severe impact damage. There is some gradual change in coupon stiffness but the progression towards final failure may also include large step changes in overall modulus.

Ultrasonic C-scanning at intervals reveals that the progression of damage is much more complicated than a simple, steady, increase in delamination area. Fatigue test observations confirm this, as delamination buckle strips appear as discrete events.

The fact that relatively large, discrete, events feature in the damage development means that any model based on gradual damage development is unlikely to be accurate. It is clear, however, that there must be a relatively gradual, microscopic, background development of damage which

acts as the initiator to the larger, macroscopic failure events which are most significant in the progression to failure.

6.4 Hysteresis in Impact Damaged Coupons

For fatigue damage to progress within any material there must be a mechanism of dissipation of energy. Dissipation of energy requires that unloading force/displacement (or stress/strain) paths must diverge from the loading paths; i.e. there must be hysteresis. Energy dissipation observed in the quasi-static indentation tests on the CFRP coupons has been described in Chapter 2, Section 2.9. It was expected that the fatigue process in impact damaged CFRP would be driven by a similar or related mechanism of energy absorption.

Measurements taken from strain gauges and extensometers fixed to impact damaged coupons indicated time dependent and viscoelastic behaviour in the impact damaged CFRP coupons. Creep under constant load and hysteresis under quasi-static and dynamic (high frequency) cyclic loads was observed. The measurements and observations made are given in this section.

6.4.1 Literature Results

Kennedy and Serdinak [66] made measurements of hysteresis in the impact damage zone of woven CFRP composite laminates under tension-compression fatigue and found this to be a reliable indicator of the onset of final failure.

The material studied was woven CFRP with carbon fibre through thickness reinforcement. Panels were tested for CAI strength and in tension-compression fatigue with a load ratio $R = -5$ (compression dominated) and a frequency of 4 Hz. Global buckling of the specimens was prevented by knife edge supports. Damage was introduced by high speed (160 m/s) impact from aluminium ball projectiles.

X-radiograph images of the fatigue tested coupons showed that most of the damage growth (extension of matrix cracks and delaminations) occurred in the last 3 to 4% of the fatigue life.

Damage development was also monitored by the use of extensometers attached across the impact damage site. This allowed comparison of the average displacement response of the damaged zone with far field stress at different stages of the fatigue life. It was found that for

approximately 95% of the fatigue life there was no measurable difference in the loading and unloading paths of the stress/displacement plots. However in the last 5% of the fatigue life significant hysteresis was measured in the impact damage zone. The increase in the magnitude of this hysteresis was a reliable indicator of the onset of final failure.

Orth et al. [67] proposed using dynamic hysteresis measurements of the force/extension behaviour of composite components during fatigue tests as an indicator of the development of fatigue damage. They maintain that the suitability of a composite component cannot accurately be described just by reference to an S-N (load-life) diagram. Additional information about the fatigue life can be obtained from hysteresis measurements during the fatigue process. As an example, they give results for fatigue tests on a CFRP engine valve component. Significant changes in damping ratio were measured after 10^4 cycles in a test where failure occurred at 10^5 cycles.

6.4.2 Creep Under Constant Compressive Load

Changes observed in strain distribution in an impact damaged coupon under constant load may be attributed to creep (viscoelastic behaviour) of the CFRP. The magnitude of this creep was greatest in the areas of higher strain close to the impact damage site.

Strain gauges were positioned on the horizontal centre line of an impact damaged coupon. The gauges were positioned next to the impact damage site on both the front (impacted) face and back face of the coupon. The positioning of the gauges is shown in Figure 5.3. The coupon had been subjected to a “severe” impact (10J) resulting in an indentation of 0.43 mm and an overall delamination area of 558 mm². Strain gauges on the front face (the impacted side of the coupon) were positioned at distances from the coupon edge as detailed in Table 6.3.

Front Face (Impact Side) Strain Gauge Positions

Strain Gauge	1	2	3	4	5	6
Distance from Coupon Edge (mm)	2.5	9	16.5	23	28.5	34

Table 6.3

A compressive load profile was applied quasi-statically to the coupon. The load profile is shown in Figure 6.11. Load was increased at a steady rate, held constant, and then decreased at a steady rate back to zero. The test lasted approximately 7 minutes.

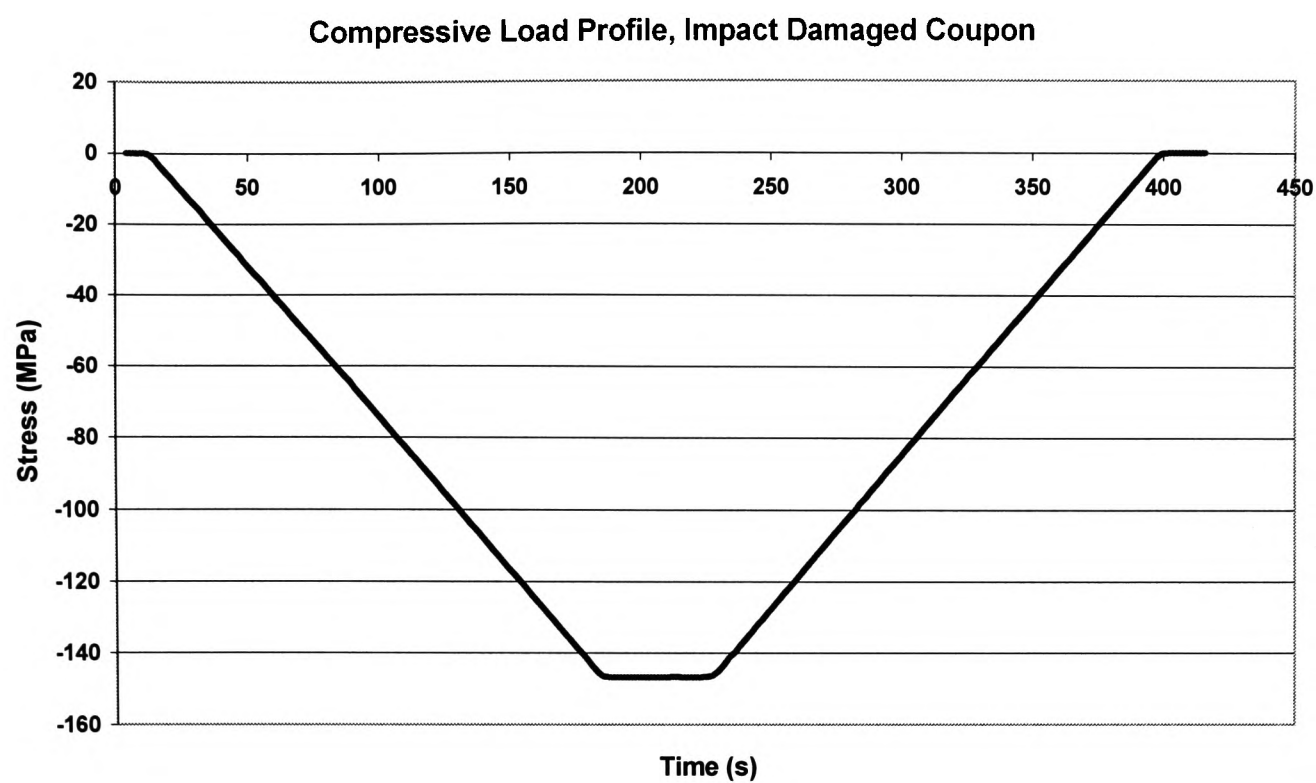


Figure 6.11

The strains recorded during the compressive test are plotted in Figure 6.12. The compressive strain distribution is as discussed in Section 5.8. Local buckling at the impact damage site causes non-linear behaviour and an increase in the compressive stress concentration on the front face next to the impact indentation.

The strain distribution under the constant load is not constant. Strain gauge No. 6, which was the gauge closest to the impact indentation, shows an increase in compressive strain under the constant compressive load. When the strains are plotted against the average applied stress (Figure 6.13) a marked hysteresis loop can be seen for gauge 6. Lesser, but observable, hysteresis was measured for the gauges closer to the coupon edge (1 to 5).

Measurements of strain distribution were also made from the coupon back face. The positions of the gauges on the coupon back face are given in Table 6.4.

Strain gauge readings for the static compression test are plotted in Figure 6.14. No creep was measured by the back face gauges under the constant load. However, upon unloading, the gauge readings do not return to zero, indicating that there is a residual compressive strain left in the coupon back face. The residual strain is most significant for the gauge furthest from the

coupon edge and closest to the impact damage site (No. 7).

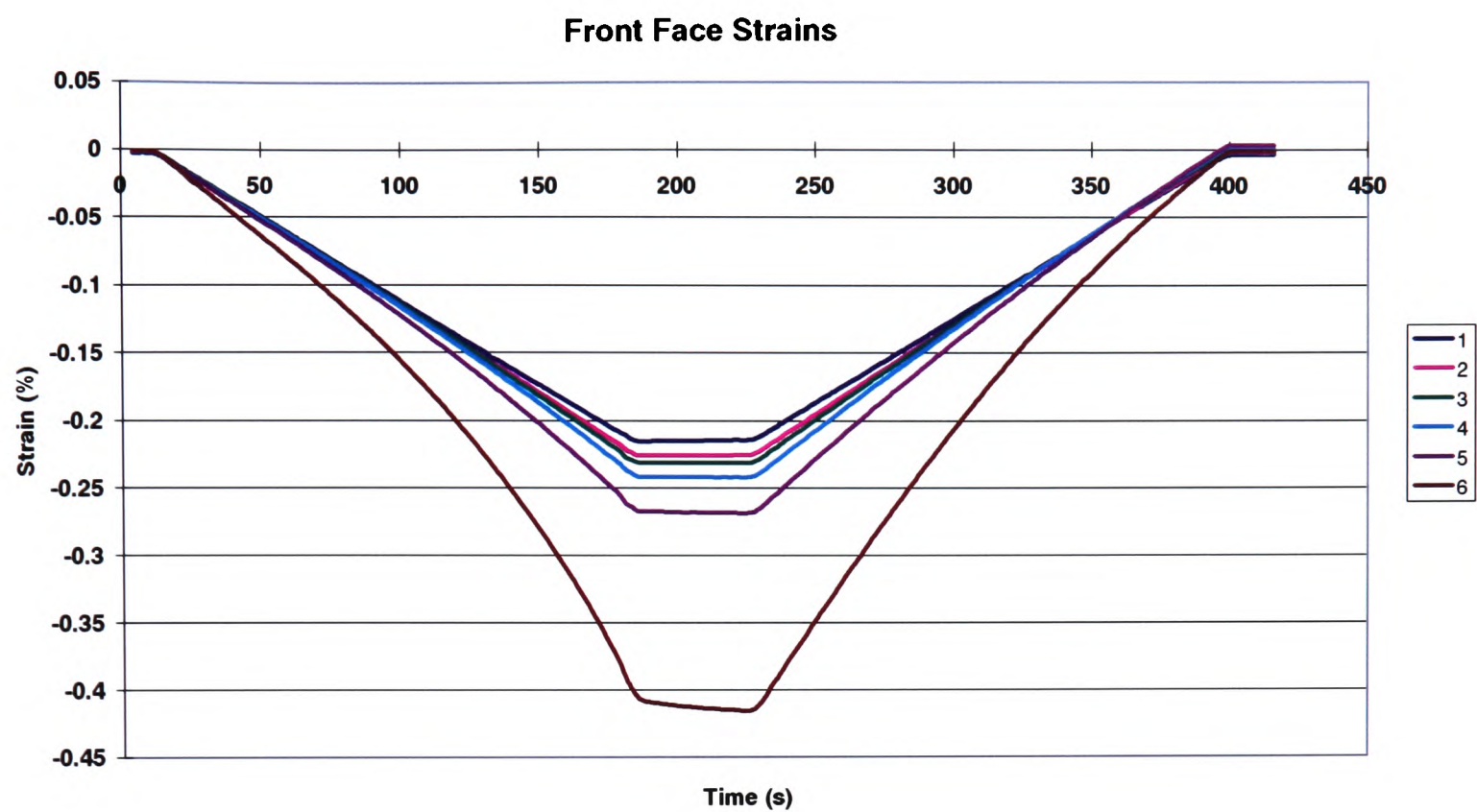


Figure 6.12

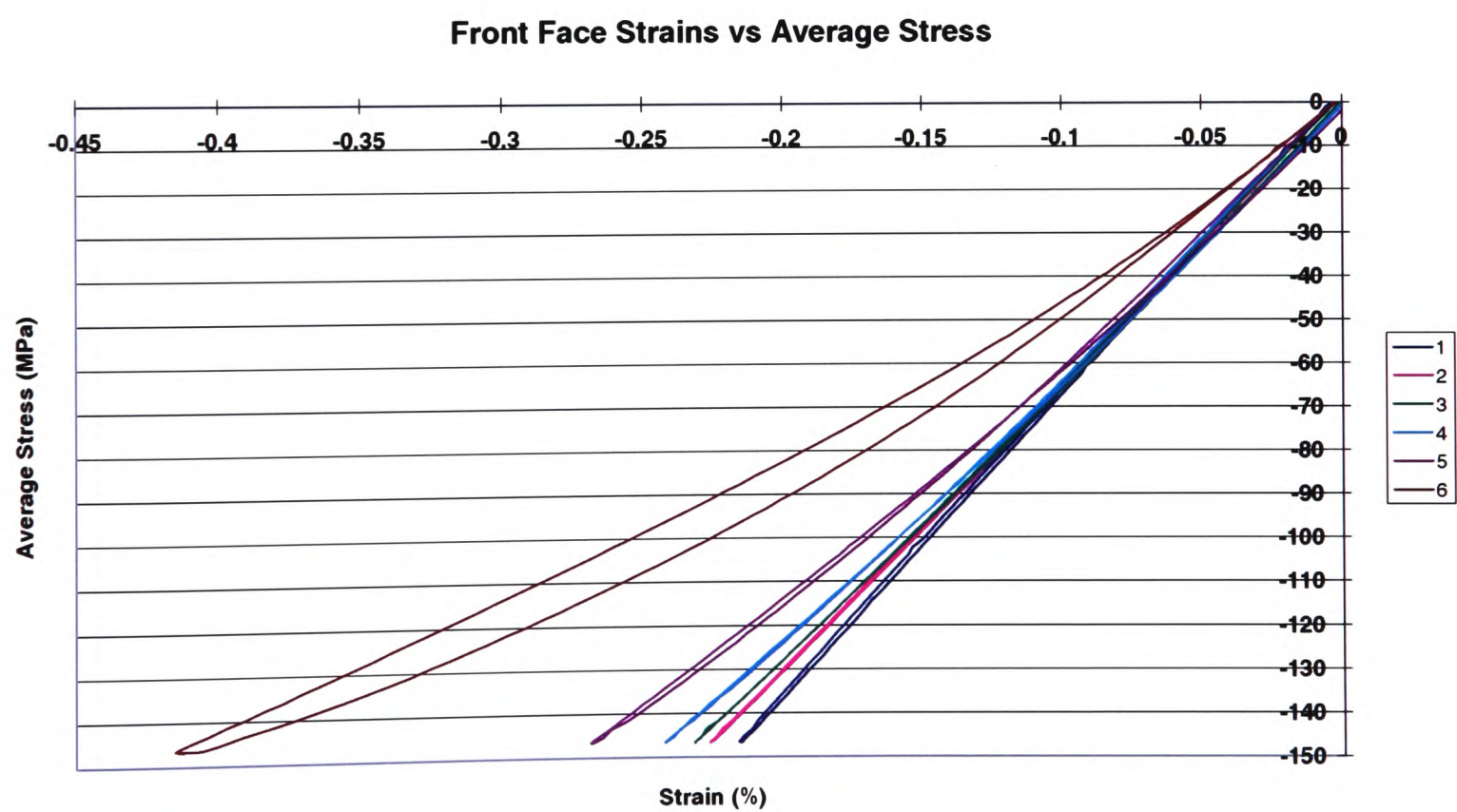


Figure 6.13

This residual strain can also be seen in Figure 6.15 where the strain gauge readings are plotted against the average applied stress. The non-linear path of gauge 7 at the higher loads is due to the local buckling of the impact damage site.

The combination of time dependent creep and residual strain in the damaged CFRP suggests that both viscoelastic and frictional effects are being observed. Viscoelastic properties of composites generally arise from time dependent behaviour of the polymer epoxy matrix. In the case of impact damage, where the distribution of matrix cracks and delamination will cause areas of high stress and strain in the matrix, it is reasonable to expect that matrix creep will become more significant. Frictional effects are likely to arise from the sliding of broken fibres within the matrix, where failure of the fibre/matrix interphase has caused the fibre to de-bond. Friction may also arise between delaminated plies as they slide over each other.

Back Face Strain Gauge Positions

Strain Gauge	7	8	9	10	11	12
Distance from Coupon Edge (mm)	28.5	24	19	13	8	2

Table 6.4

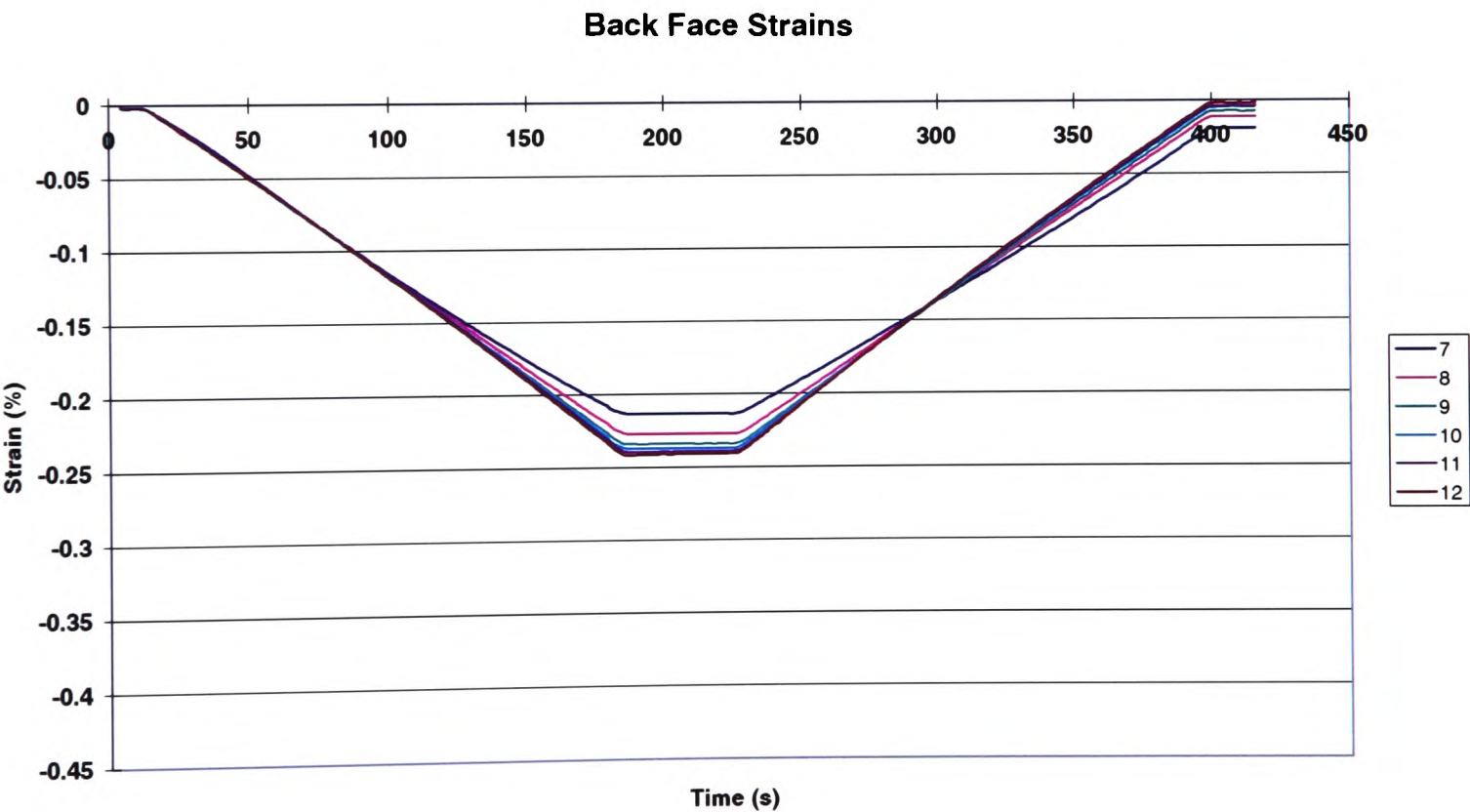


Figure 6.14

6.4.3 Hysteresis in Quasi-static Cyclic Loading

Strain gauges themselves eventually fail under the cyclic loading and for this reason it was not possible to accurately record changes in strain levels in the coupon throughout a fatigue tests because the. The high strain levels in the coupon, 0.2 to 0.4% strain (2000 to 4000 microstrain), were such that the strain gauges would typically fail after 20000 - 80000 cycles.

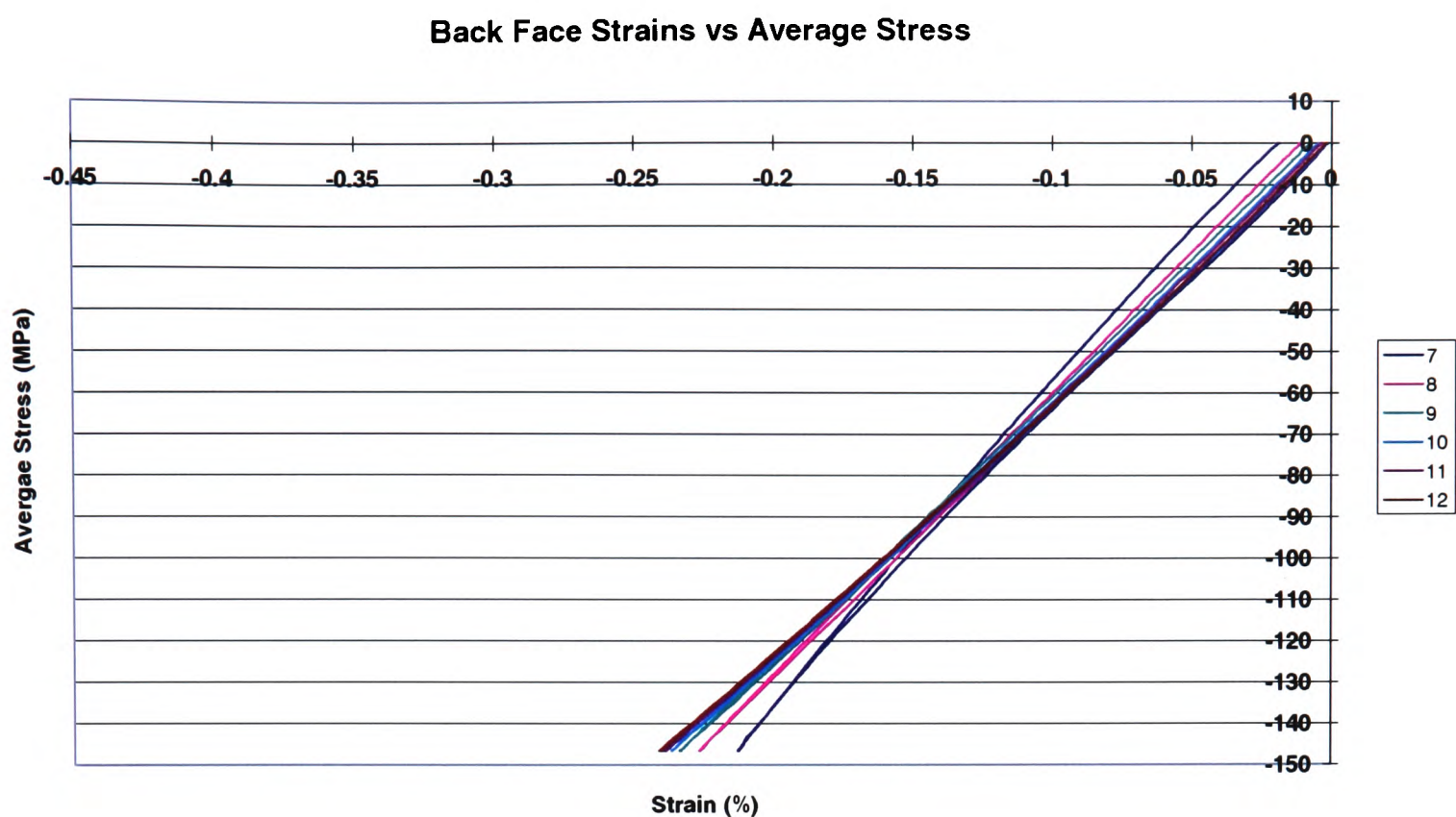


Figure 6.15

It was possible, however, to accurately measure the strain response of an impact damaged coupon over a single cycle. Measurements of strain for a coupon with severe impact damage were for a very low frequency (0.005 Hz) sinusoidal load input. The cyclic load profile used is shown in Figure 6.16.

The strains measured on the coupon front face are plotted against average stress in Figure 6.17. Significant hysteresis can be seen for gauge 6 which was closest to the impact damage site. Slight hysteresis is also observed for the gauges further from the damage zone. Strains measured on the back face are plotted in Figure 6.18. The most significant hysteresis was measured by gauge 7 which was closest to the back face impact damage bulge.

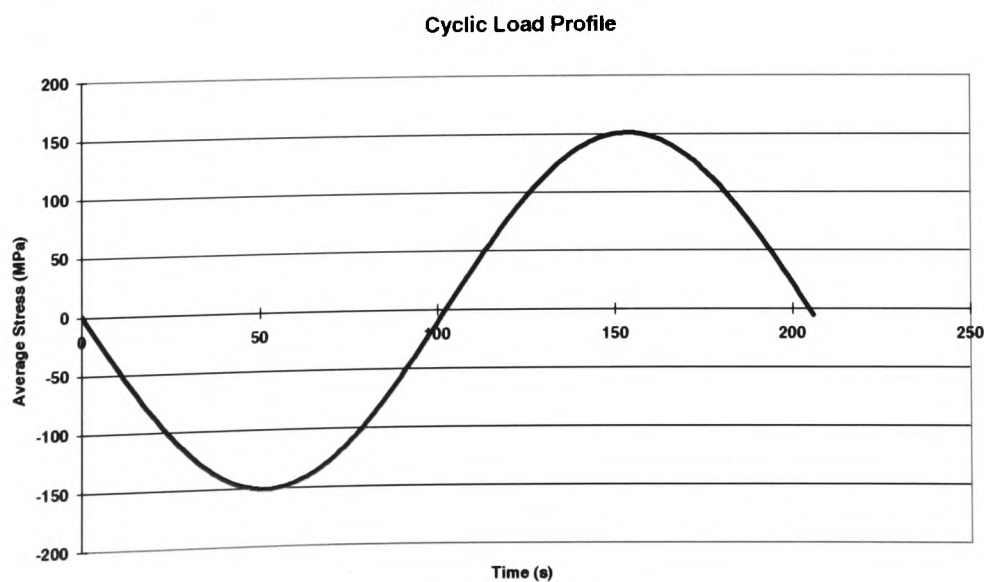


Figure 6.16

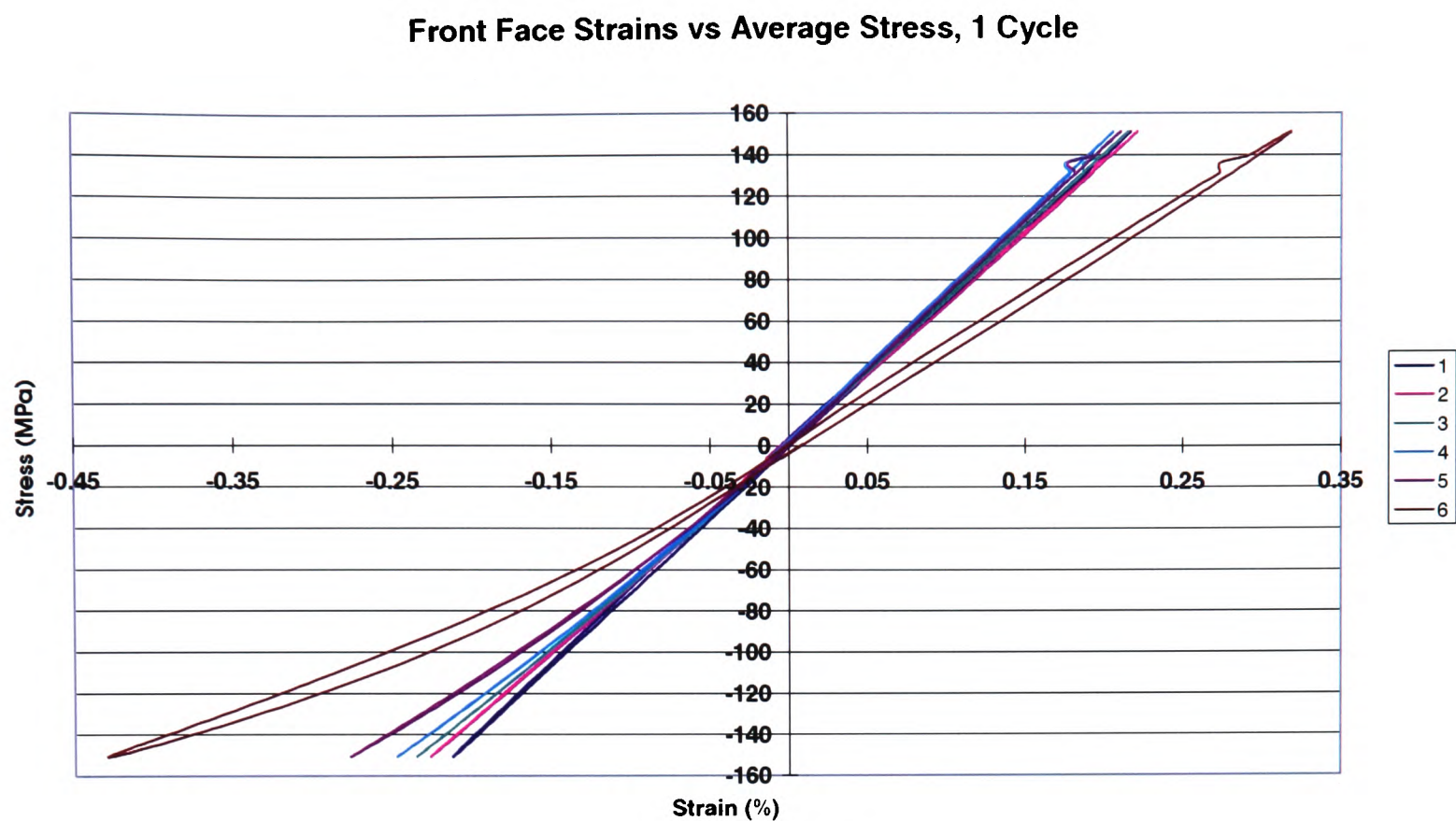


Figure 6.17

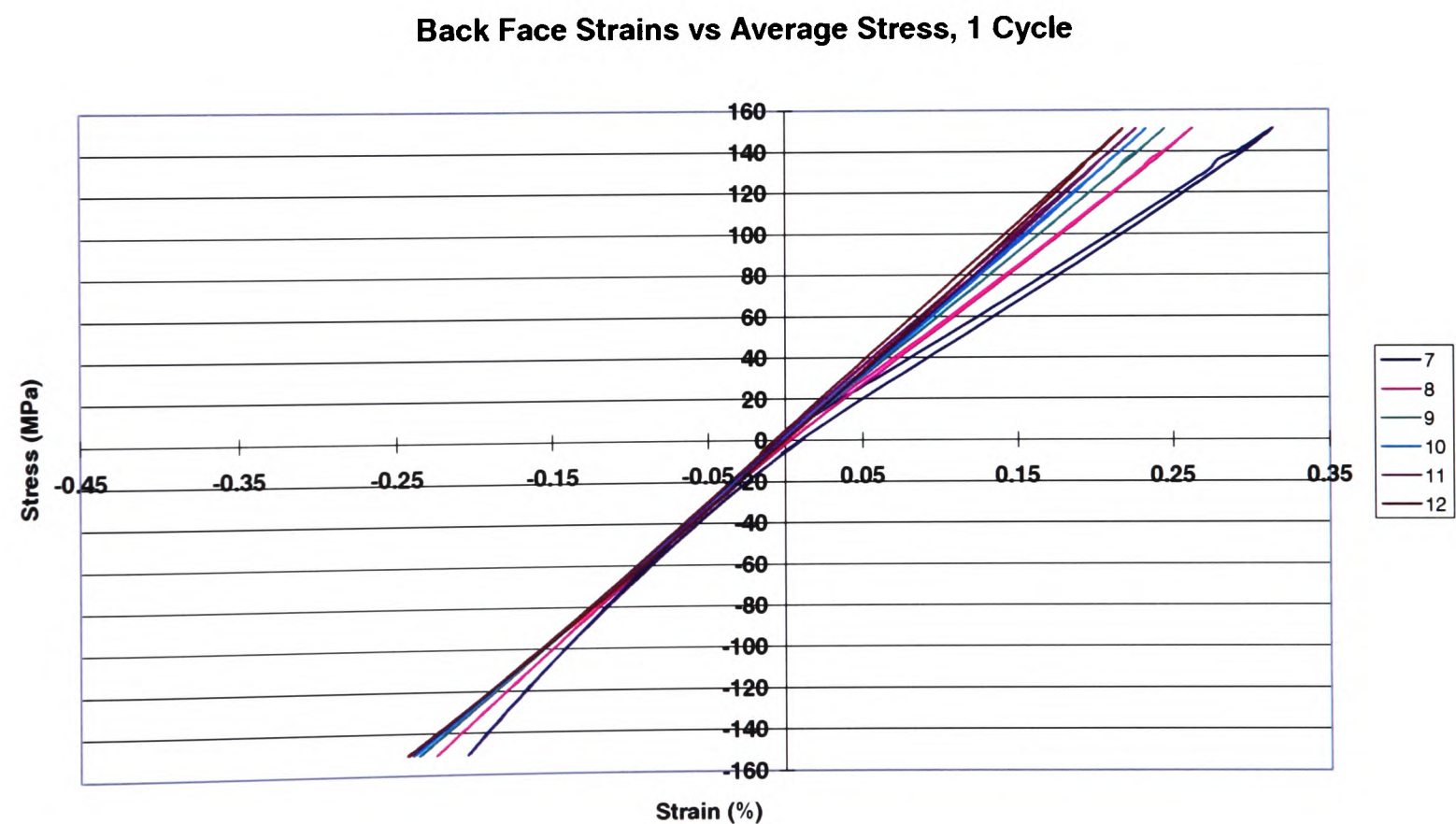


Figure 6.18

6.4.4 Hysteresis under Dynamic Loading

During fatigue tests of the impact damaged coupons A and B (Section 6.4.3) measurements were made, at intervals, of the overall force/extension behaviour through 10 cycles at the loading frequency. Force measurements were obtained directly from the Dartec testing machine load cell while extension measurements were from a clip gauge extensometer attached to the

coupon. It was observed that there was a phase lag between the coupon extension and the sinusoidal input load and this causes an energy dissipating hysteresis loop.

Hysteresis loops measured in this way at five different stages of the fatigue life of coupon B are plotted in Figure 6.19. The first four were taken at one hour intervals during the fatigue test; the final loop shows the force/extension response for the final eleven cycles before failure. Force and extension of the coupon are normalised as “average applied stress” and “coupon average strain” for plotting. It can be seen that the point of failure of the coupon occurred just before maximum compression (when the stress drops very suddenly at approximately -0.7% strain).

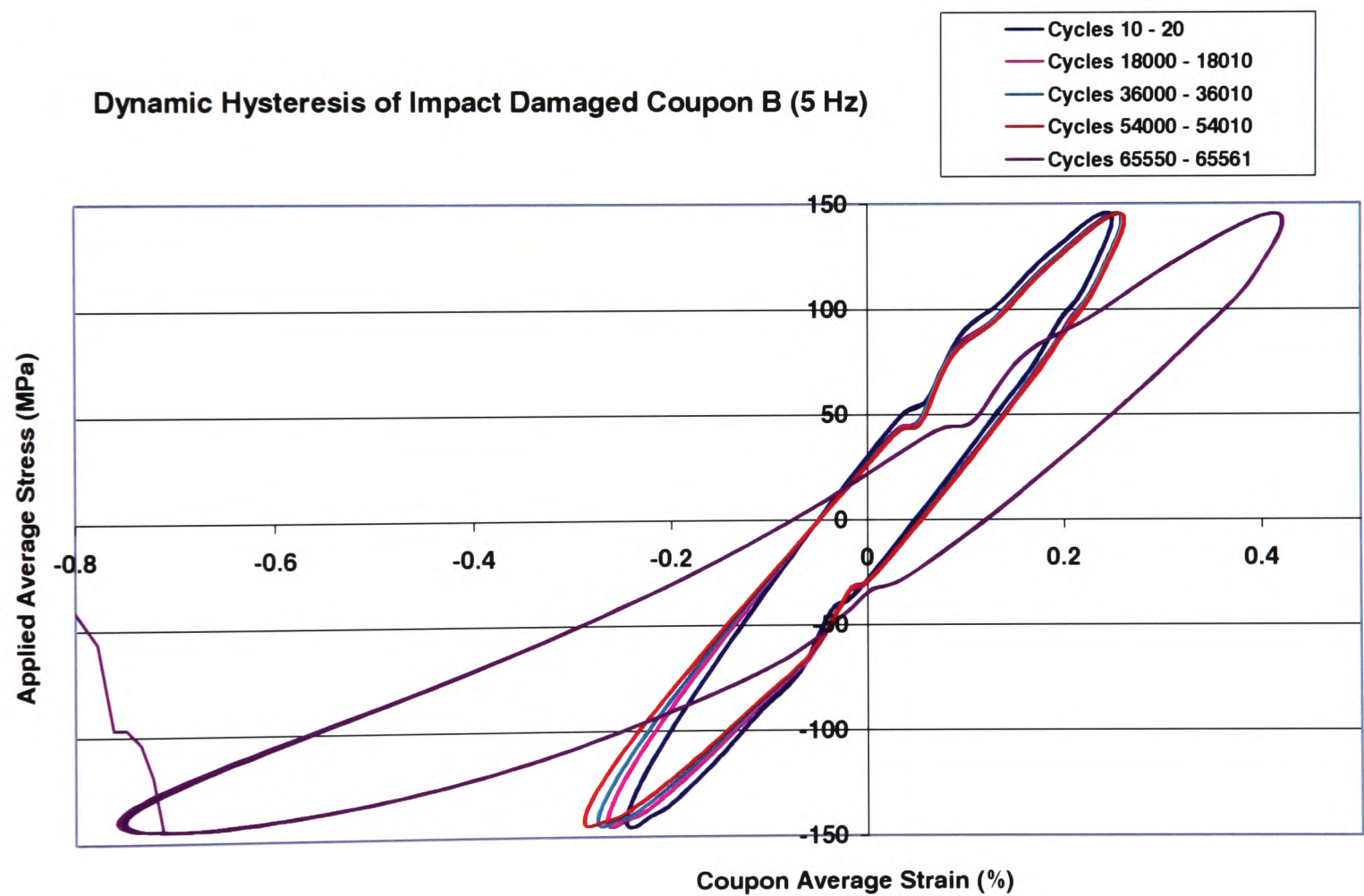


Figure 6.19

Over the first four hours of the fatigue test there was a slight increase in the overall coupon modulus. This has already been shown in Figure 6.8. There was little change in the overall energy absorption characteristics of the coupon however. We may define a damping ratio Δ for the coupon as the ratio of the specific energy absorbed per cycle u_a to the maximum strain energy stored per cycle u_m .

$$u_a = \int \sigma d\varepsilon \tag{6.10}$$

$$u_m = \frac{1}{2} \sigma_{\max} \epsilon_{\max} + \frac{1}{2} \sigma_{\min} \epsilon_{\min}$$

6.11

$$\Delta = \frac{u_a}{u_m}$$

6.12

The variation of the damping ratio throughout the fatigue test of coupon B is calculated in Table 6.5. Although the energy absorbed in the hysteresis loop of each cycle gradually increased throughout the test the strain energy stored in each cycle also increased (because of the decrease in coupon modulus). Consequently there was no significant change in the damping ratio.

Energy Absorption and Damping in Fatigue Test of Coupon B

Cycle	Average energy absorbed per cycle u_a (kJ/m ³)	Average maximum strain energy per cycle u_m (kJ/m ³)	Damping ratio Δ	Rate of energy absorption in coupon (W)
10	231	352	0.66	17.9
18000	242	373	0.65	18.7
36000	244	378	0.65	18.9
54000	250	387	0.65	19.3
65550	573	827	0.69	44.3

Table 6.5

Hysteresis measurements at different frequencies were made in the fatigue test of coupon A. The force / extension behaviour is normalised as an average stress and strain in the coupon and plotted in Figure 6.20. The damping ratio at different frequencies is calculated in Table 6.6. The initial five cycles of coupon A were conducted at a very low frequency of 0.005 Hz. At this speed of loading the overall force / extension behaviour of the coupon was almost linear elastic with a damping (energy absorption) ratio of only 0.025. Next, the loading frequency was increased briefly to 2 Hz, before being set at the testing frequency of 5 Hz. At 2 Hz and 5 Hz, damping in the coupon causes significant hysteresis in the force / extension response. The response of the coupon during the last eleven cycles up to failure is also plotted in Figure 6.20. Damage development has caused a decrease in the overall coupon modulus. Final failure

occurs just before the point of maximum compression in the last cycle.

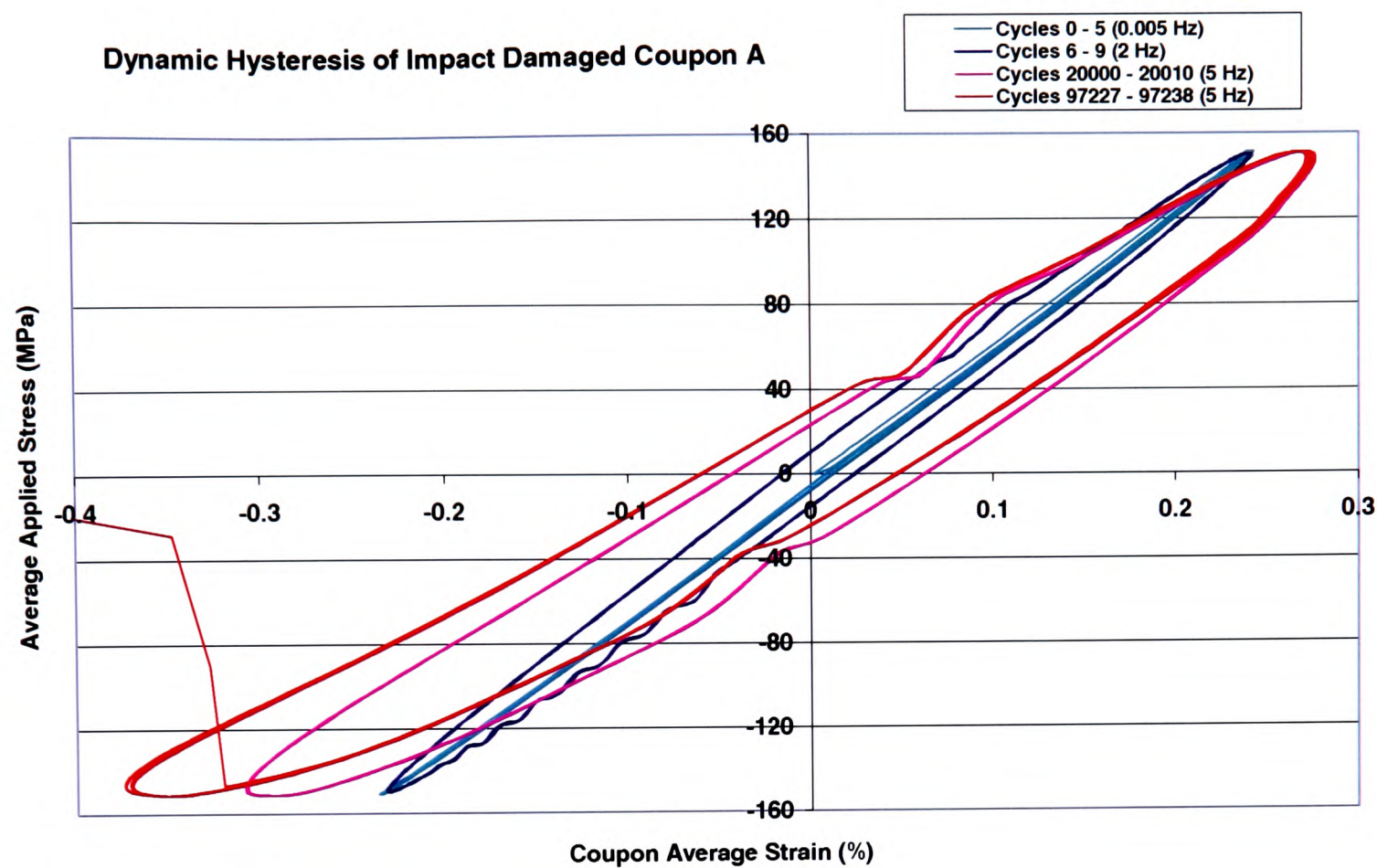


Figure 6.20

Energy Absorption and Damping in Fatigue Test of Coupon A

Frequency (Hz)	Cycle	Average energy absorbed per cycle u_a (kJ/m ³)	Average maximum strain energy per cycle u_m (kJ/m ³)	Damping ratio Δ
0.005	1	8.8	352	0.025
2	6	95.4	352	0.27
5	20000	270	435	0.62
5	97238	283	471	0.60

Table 6.6

6.4.5 Hysteresis in Undamaged CFRP

Measurements of the force/extension response of an undamaged 21 ply coupon were made in order to compare the magnitude of hysteresis of the impact damaged coupons with the virgin, or undamaged, material. Sinusoidal cyclic load tests were made at a range of frequencies from 0.005 Hz to 5 Hz.

Figure 6.21 shows the force/extension (normalised as stress and strain) behaviour of an

undamaged 21 ply coupon under quasi-static (0.005 Hz) and dynamic (5 Hz) loading. At 5 Hz very little hysteresis is measured. However at 0.005 Hz there is significant hysteresis. These results are evidence of creep in the undamaged composite. Viscoelastic behaviour of the epoxy matrix allows relaxation of the composite under load. At the high frequency, however, there is insufficient time in each cycle for the creep to be significant and therefore the damping ratio is lower.

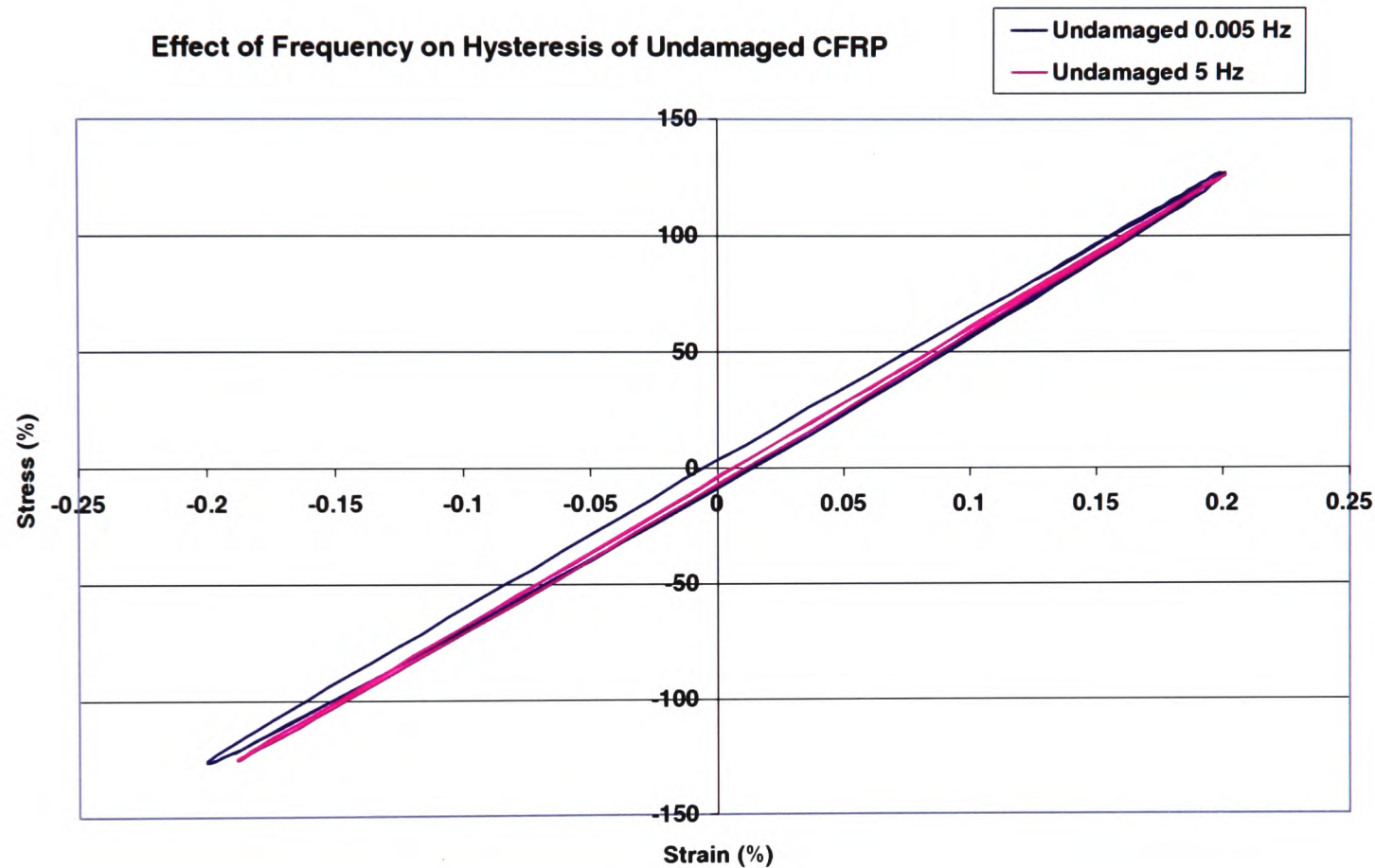


Figure 6.21

The reduction in damping ratio with increase in frequency is the opposite effect to that observed in cyclic loading of the damaged coupons, where the damping ratio increased with frequency. In Figure 6.22 and 6.23 the response of the undamaged coupon is compared to an impact damaged coupon under quasi-static and dynamic cyclic loading.

Figure 6.22 shows the response of the CFRP coupons at quasi-static rates of loading (<0.01 Hz). A number of observations may be made:

- There is a larger hysteresis loop in the response of the undamaged coupon than in the impact damaged coupon.
- Under tensile loading the overall modulus of the damaged coupon is slightly reduced due to the presence of the central impact damage.
- After the first tensile cycle a residual tensile strain is introduced into the

damaged coupon which is not recovered.

- With each cycle the overall modulus of the damaged coupon is slightly reduced and the maximum strain under constant load gradually increases. This trend continues very gradually throughout the fatigue life (Section 6.3.4, Figure 10).

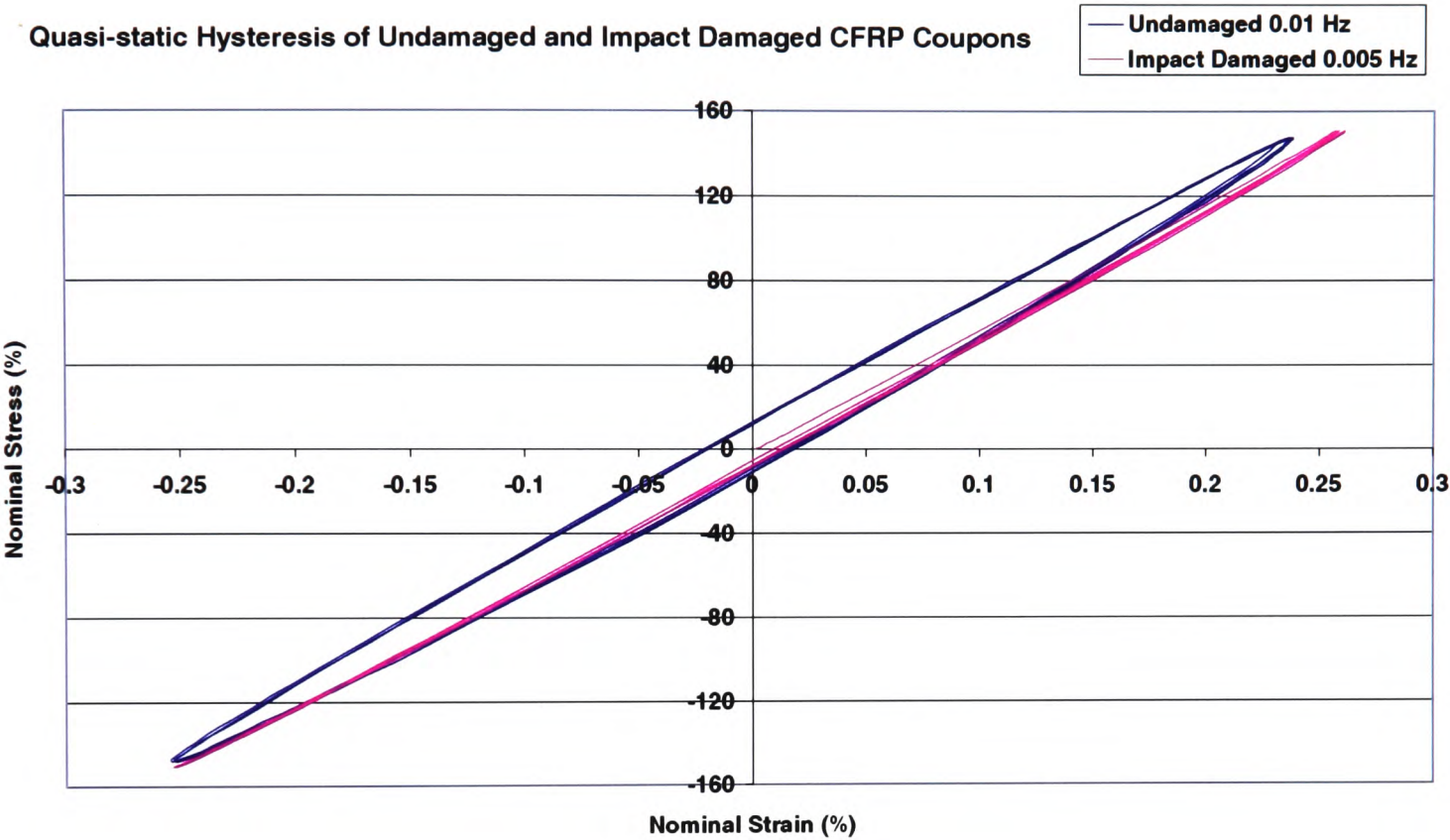


Figure 6.22

Figure 6.23 shows the very significant influence of impact damage on the dynamic hysteresis of a CFRP coupon. There is very little hysteresis in the undamaged coupon at 5 Hz, while in contrast there is very significant hysteresis in the impact damaged coupon.

The damping ratios for undamaged and impact damaged coupons at dynamic and quasi-static loading rates are given in Table 6.7.

Damping (Energy Absorption) Ratio

Cyclic Loading	Quasi-static (0.005 - 0.01 Hz)	Dynamic (5 Hz)
Undamaged Coupon	0.24 (0.01 Hz)	0.057
Impact Damaged Coupon	0.025 (0.005 Hz)	0.66

Table 6.7

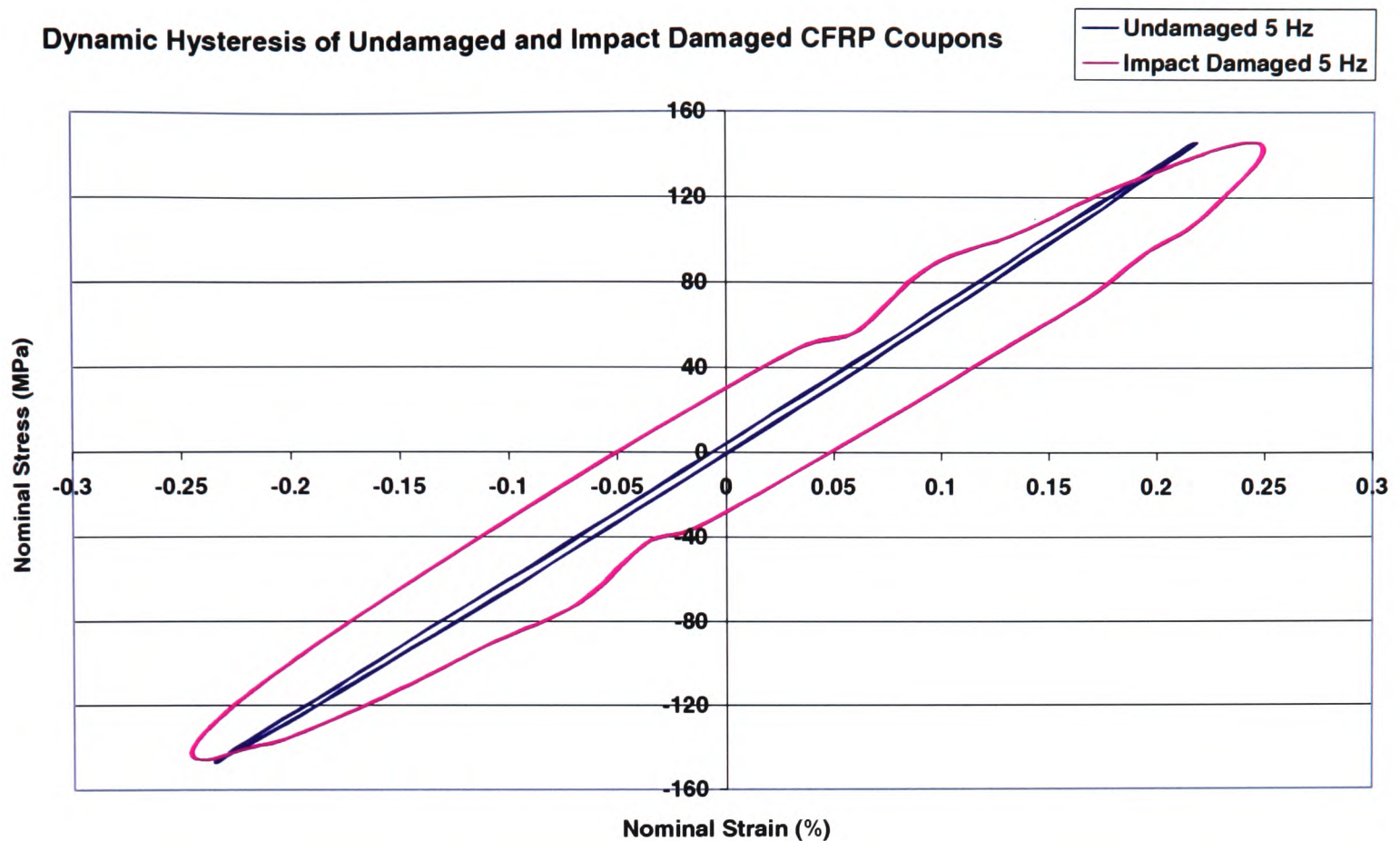


Figure 6.23

6.4.6 Viscoelastic Model

Much of the observed time dependent behaviour of undamaged and damaged CFRP may be explained by simple viscoelasticity. Viscoelastic behaviour of structural materials results in four important physical manifestations: firstly creep, where strain increases under constant stress; second, relaxation, in which stress decreases under constant strain; third, under oscillatory loading the resulting stress-strain curve will describe a hysteresis loop; fourth, under varying strain rates, stress-strain curves will exhibit a strain-rate dependence.

Models for viscoelastic behaviour based on linear springs and dashpots have been examined by Gibson [39]. A dashpot is a device which provides a resisting force directly proportional to velocity. The simplest viscoelastic models are the Maxwell model, which consists of a spring and dashpot in series, and the Kelvin-Voigt model, which is a spring and dashpot in parallel. The standard linear solid, or Zener model, is made up of one spring in parallel with a Maxwell model. The Zener model is the simplest spring-dashpot model which correctly describes all expected features of experimentally observed creep and relaxation behaviour in linear viscoelastic materials. However, it has often still been found inadequate because real polymer behaviour includes a distribution of relaxation times. To cope with this problem

improved Zener models have been used, consisting of either n Maxwell elements in parallel or n Kelvin-Voigt elements in series.

A relatively simple viscoelastic model was proposed and tested here which can be used to explain much of the observed behaviour of the undamaged and impact damaged CFRP coupons presented in this thesis. The model is shown diagrammatically in Figure 6.24 and consists of two Kelvin-Voigt elements in series.

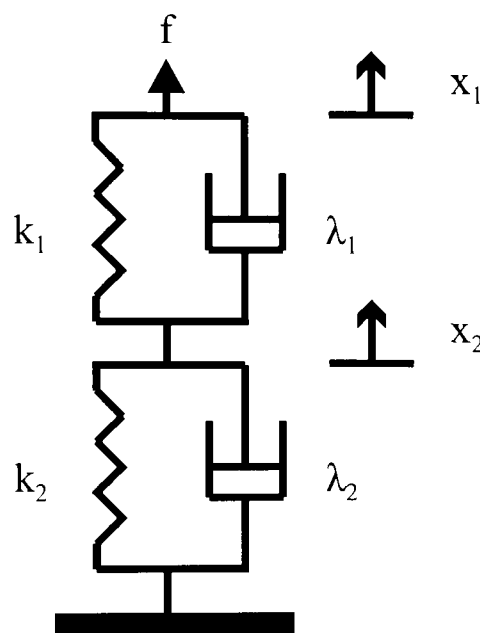


Figure 6.24

The response of the viscoelastic model to an external force f is defined by Equation 6.13. If f is a sinusoidally oscillating input (Equation 6.14) of amplitude F and frequency ω then the response x_1 and x_2 will be of the same frequency, but with amplitudes X_1 and X_2 and phase angles ϕ_1 and ϕ_2 (Equations 6.15 and 6.16). The phase angles and amplitudes of response may be calculated from Equations 6.17 to 6.20. The energy loss damping ratio Δ may be directly calculated from the output phase angle ϕ_1 (Equation 6.21).

$$f = k_1(x_1 - x_2) + \lambda_1(\dot{x}_1 - \dot{x}_2) = k_2x_2 + \lambda_2\dot{x}_2 \quad 6.13$$

$$f = F \sin(\omega t) \quad 6.14$$

$$x_1 = X_1 \sin(\omega t + \phi_1) \quad 6.15$$

$$x_2 = X_2 \sin(\omega t + \phi_2) \quad 6.16$$

$$\tan \varphi_2 = \frac{-\lambda_2 \omega}{k_2} \quad 6.17$$

$$\tan \varphi_1 = \frac{\lambda_2 \omega (\lambda_1^2 \omega^2 + k_1^2 + \lambda_1 \lambda_2 \omega^2) + \lambda_1 \omega k_2^2}{\lambda_2 \omega (\lambda_1^2 \omega^2 + k_1^2 - k_1 \lambda_2 \omega) - k_1 k_2^2} \quad 6.18$$

$$X_2 = \frac{F}{k_2 \cos \varphi_2 - \lambda_2 \omega \sin \varphi_2} \quad 6.19$$

$$X_1 = \frac{F}{k_1 \cos \varphi_1 - \lambda_1 \omega \sin \varphi_1} \left(1 + \frac{k_1 \cos \varphi_2 - \lambda_1 \omega \sin \varphi_2}{k_2 \cos \varphi_2 - \lambda_2 \omega \sin \varphi_2} \right) \quad 6.20$$

$$\Delta = \pi \tan \varphi_1 \quad 6.21$$

By choosing appropriate values for the spring and dashpot constants this model can be used to give reasonably accurate predictions of the experimental hysteresis response of the undamaged coupon as a function of frequency (Figure 6.21). It is necessary to include at least two Kelvin-Voigt elements in the model because hysteresis was measured at both high and low frequencies, implying a distribution of relaxation times. Figure 6.24 shows the prediction of the model.

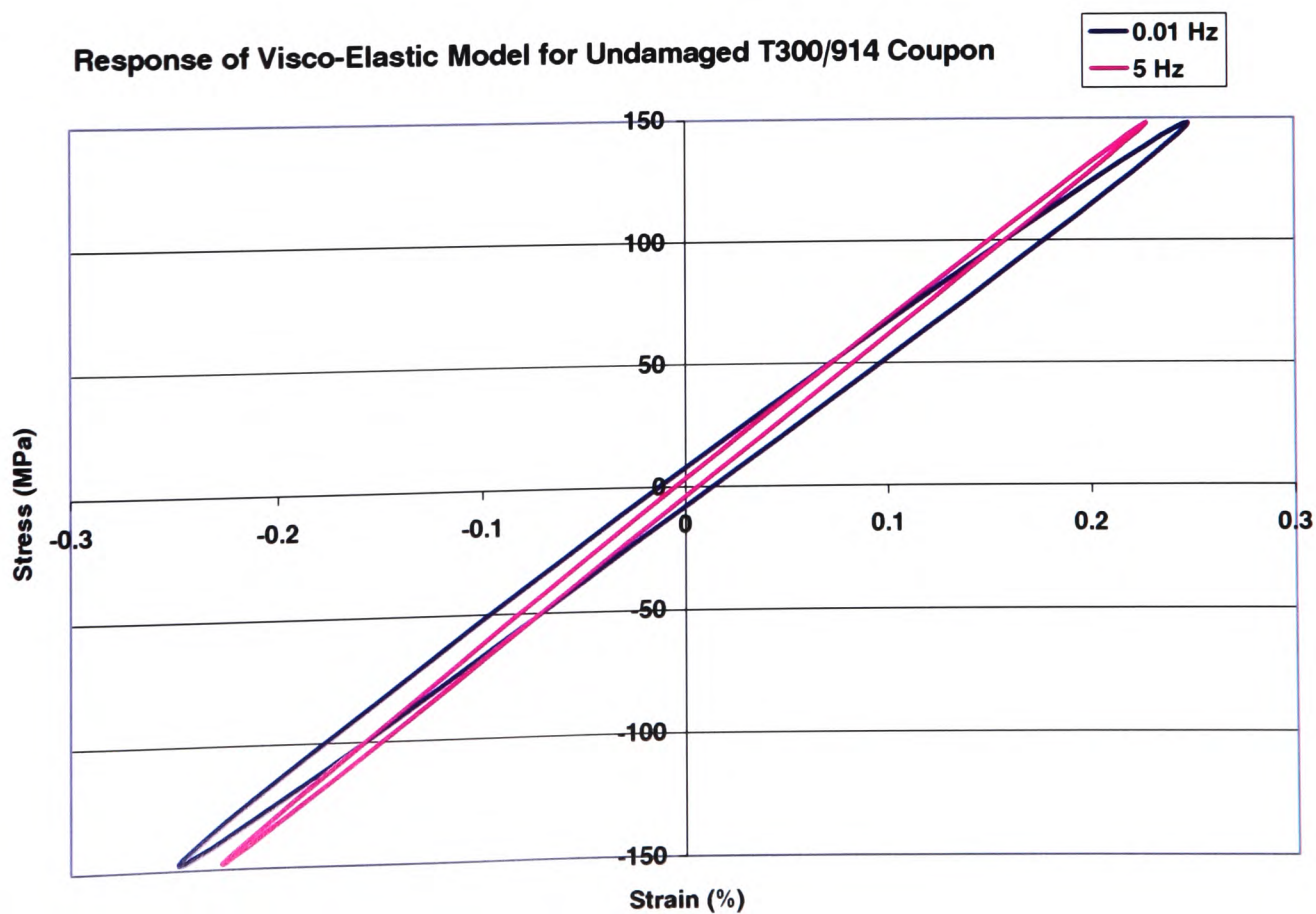


Figure 6.24

Modification of the model coefficients allows prediction of the behaviour of the coupon after impact damage has been introduced. Figure 6.25 shows the prediction of the model for the impact damaged coupon.

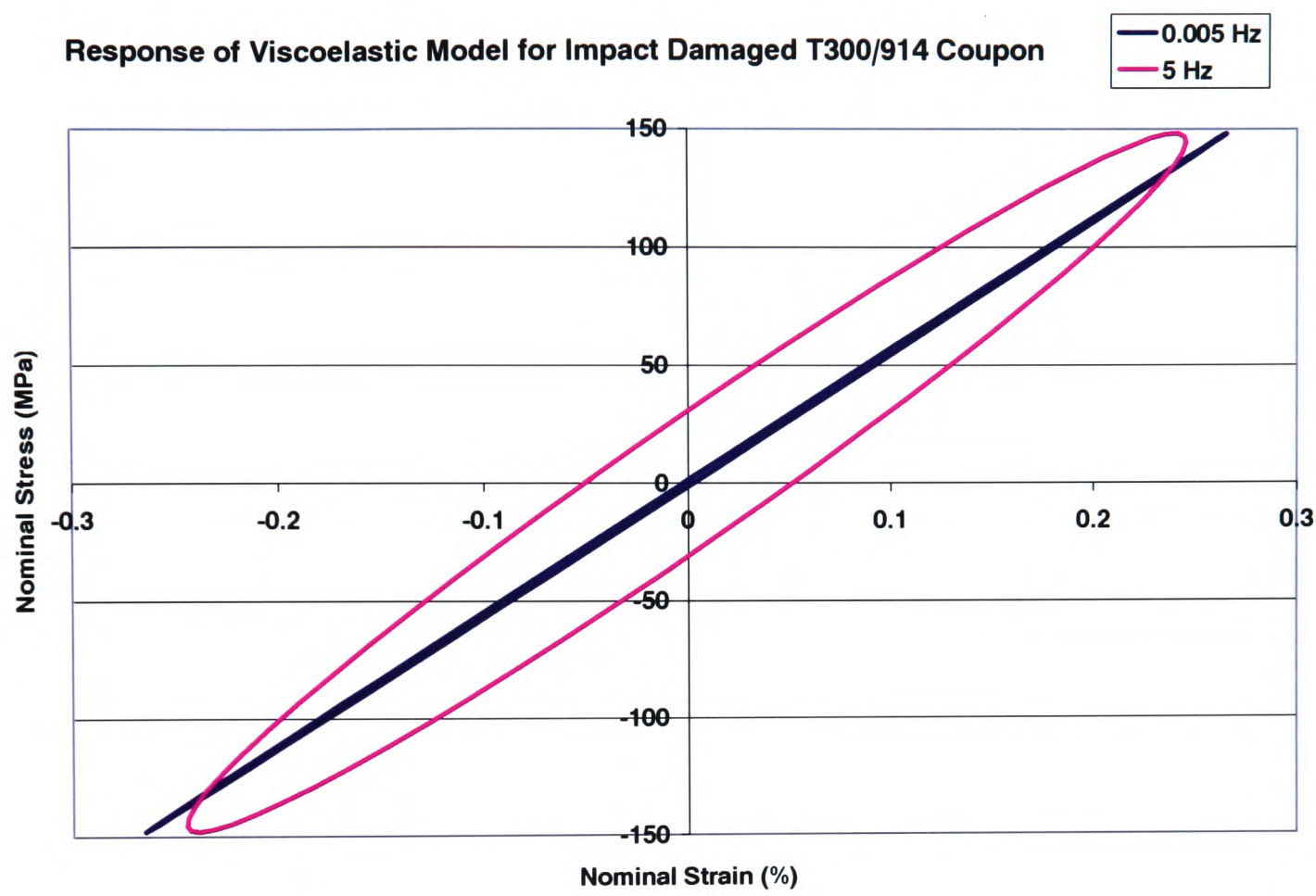


Figure 6.25

Table 6.8 gives the viscoelastic model spring and dashpot constants that produced the approximate hysteresis response for the undamaged and impact damaged coupons. Spring and dashpot constants have been normalised to give strain output from a stress input. The undamaged coupon model has a main stiffness k_1 which is close to the measured Young’s Modulus of the $0^\circ/90^\circ$ lay-up coupon (64 GPa, Section 5.6.1). Creep at very low frequency cyclic loading is accounted for by the extra stiffness and high resistance damper k_2 and λ_2 . In addition there is a low resistance damper λ_1 to account for the small amount of hysteresis at 5 Hz.

To account for the changed response when impact damage is introduced, the viscoelastic model constants had to be modified. The main stiffness k_1 was reduced slightly and the damper constant λ_1 was increased by a factor of eight to produce the significant hysteresis at 5 Hz. However, the undamaged coupon also showed less creep at the very low speed cyclic loading and so the stiffness k_2 had to be increased to give the correct approximate response.

Viscoelastic Model Coefficients and Calculated Damping Ratios

Coupon	k_1 (GPa)	k_2 (GPa)	λ_1 (GNs/m ²)	λ_2 (GNs/m ²)	Quasi-static Damping Δ	Dynamic Damping Δ 5 Hz
Undamaged	65	550	0.05	5000	0.17 (0.01 Hz)	0.077
Severe Impact Damage	59	1000	0.4	5000	0.029 (0.005 Hz)	0.67

Table 6.8

6.4.7 Discussion

It is difficult to provide a simple justification for the apparent reduction in creep and introduction of residual strain in the impact damaged coupon although a number of additional factors may be significant. Some of these are discussed here.

i) Transverse Ply Cracking

Because the transverse failure strain of a unidirectional ply is lower than the longitudinal failure strain (Section 5.4), tensile or bending loading of a laminate will generally result in a distribution of matrix cracks in the 90° plies, before fibre failure occurs in the 0° plies. We may therefore assume that there will be a significant distribution of matrix cracks in the 90° plies around the impact damage site. If we further assume that the creep behaviour of the undamaged coupon is largely a function of load transfer to the 0° plies from relaxation of the 90° transverse plies, then it is possible that cracking in the transverse plies may influence the creep behaviour, and therefore reduce the low speed hysteresis seen in the impact damaged coupon.

ii) Friction

The loading and unloading cycle of an impact damaged coupon will result in friction between de-bonded fibres and matrix and also between delaminated plies. Frictional effects can not be accounted for in a simple viscoelastic model.

iii) Stick-slip

In push-out tests of a mono-filament SiC fibre in a polymer PMMA matrix Cherouali et al. [68] observed a “seal-skin” effect in the resistance to the fibre motion. In one direction the fibre showed continuous alternating stick and slip but in the other direction steady sliding was measured. There was also a “seating drop” as the fibre was returned to its original position. It is probable that similar behaviour will occur in the pull-out

and push-back of de-bonded carbon fibres in an epoxy matrix. Such behaviour would provide energy dissipation (consistent with damping and hysteresis) and changes in the stick-slip behaviour during cycling would provide a fatigue mechanism.

iv) Differences in Tension/Compression Behaviour

Differences in tension and compression behaviour of the impact damaged coupon may originate partly at the microscopic level of the stick-slip of de-bonded fibres, as discussed above. However at the macroscopic level it is clear visually that local buckling of the delaminated plies around the impact site was significant.

6.5 Conclusions

Tension - compression fatigue tests on impact damaged CFRP coupons give stress life failure results which may be described by Talreja's fatigue life diagram. At high stresses the failure is essentially static and dominated by a fibre failure mechanism. At lower stresses there is a sloping scatter band of points which represent failure following a progressive, cycle dependent, damage mechanism. There is also a fatigue limit, below which there is no measurable progression of damage and "infinite" fatigue lifetimes are achieved.

For the T300/914 CFRP coupons tested a doubling of incident impact energy from 5 J to 10 J reduced the fatigue life at a given stress by approximately one decade. Fatigue limits were measured at approximately 0.18 % strain (150 MPa) for a 5 J impact (BVID) and 0.13 % strain (110 MPa) for a 10 J impact. A linear - log law can be used to approximately describe the fatigue life data at higher strains.

It is concluded that the process of damage development in an impact damaged coupon is complex and involves both gradual processes and, larger discrete failure processes. The gradual processes include a slow decrease in the stiffness of the impact damage zone and, therefore, the overall coupon modulus, as well as very slight increases in the size of the back face delamination blister. Large, discrete events in the fatigue process include the growth of front face tensile cracks early in the fatigue life and the delamination buckling of strips of surface ply around the damage zone. In most tests rapid progression of damage indicated imminent failure.

Measurements of hysteresis in impact damaged coupons gave an indication of the basis of the fatigue failure mechanism. Both viscoelastic creep under constant load and dynamic damping

were measured.

Viscoelastic creep under compressive load, and resulting residual stresses upon unloading were measured by strain gauges in the impact damage zone. These experimental results can be attributed to a combination of viscoelastic behaviour of the epoxy matrix together with frictional behaviour between de-bonded fibres and the matrix and between delaminated plies.

At dynamic rates of cyclic loading (2 - 5 Hz) significant hysteresis was measured for the overall force/extension response of the impact damaged coupons. Damping ratios of 0.65 (increasing to 0.7 before failure) were measured for a severely damaged coupon cycled at 5 Hz.

It is probable that the fatigue behaviour of impact damaged CFRP is governed by both cycle dependent and time dependent mechanisms. That is to say, there are damage progression mechanisms which depend only on the number of cycles and other, creep controlled mechanisms which depend only on the length of time under load.

Chapter 7

Conclusions and Recommendations for Further Work

7.1 Measurement and Characterisation of Impact Damage

Investigations in the measurement and characterisation of impact damage have identified appropriate techniques for the rapid assessment and characterisation of impact damage, and quantified the effect of hard body impacts on coupons fabricated from a typical aerospace CFRP material.

The most rapid technique for assessment of impact damage in CFRP is visual inspection. Severity may be quickly quantified in-situ by measurement of the permanent indentation.

Surface inspection, however, does not reveal the extent of internal damage. Ultrasonic C-scanning is an efficient method for measuring the extent of overall delamination. For assessment of impact damage in a coupon test, X-ray computer microtomography is an effective method that does not require sectioning of the damaged zone.

Low speed impact (up to ~ 10 m/s) on laminated CFRP results in a complex damage distribution of fibre fracture, matrix cracking and ply delamination which leaves a permanent surface indentation. However, in the case of barely visible impact damage (BVID) in a 21 ply laminate a surface indentation of only 0.1 mm may be accompanied by internal delamination cracks of up to 25 mm in breadth.

Most of the damage resulting from a low speed impact will be in the form of delamination. Increasing impact energy will cause greater areas of delamination but will also introduce significant fibre fracture. The brittle nature of CFRP causes considerable scatter in measurements of severity of impact damage.

High speed impact damage (~ 100 m/s) causes more localised damage in CFRP with more fibre fracture than occurs in low speed impacts. The permanent indentation is greater and the damage distribution tends towards a shear out failure at higher impact energies.

Indentation tests have shown that the damage produced in low speed impacts is essentially quasi-static and have also shown how energy is absorbed in an impact. There is evidence that there are three components of absorbed energy in impact damage in CFRP: fracture energy; frictional work between fractured surfaces; and inelastic behaviour of the matrix.

7.2 Numerical Modelling of Impact

An accurate predictive numerical modelling technique for composite impact is not presently available.

Attempts have been made to simulate the impact of a projectile with a composite coupon. A dynamic FEA model using brick elements and stress based failure criteria modified from that proposed by Chang & Chang was used. Material data for the failure criteria were obtained from small specimen tests. Although the model is moderately successful in predicting the type and location of damage that may arise in an impact, it is numerically inaccurate in predicting failure loads and extent of damage.

At present the model simulates delamination failure by reduction of the shear stiffness of a whole ply to zero. This causes an unrealistic decrease in shear stiffness of the laminate. Much improvement to the simulation could be obtained by the use of tied interfaces between plies that could be broken to simulate delamination. This would leave the delaminated ply stiffness intact. An important mode of energy absorption could be accounted for if friction on the contact surface between delaminated plies was calculated.

Entirely stress based failure criteria are, however, unlikely to solve the fundamental problems of dynamic fracture mechanics that arise here. Damage results of the model will always be highly dependent on the density of the finite element mesh discretisation. In the long term a combination of stress based criteria for damage initiation and energy based (fracture mechanics) criteria for damage propagation is more likely to be successful.

7.3 Effect of Impact Damage on Residual Strength

Experimental investigations have quantified the effect of impact damage on the residual tensile and compressive strength of a typical aerospace CFRP laminate.

For low impact energies, where the predominant damage type is delamination, local buckling of the delaminated plies in the impact damaged area causes greater reduction of the residual compression strength than of the residual tensile strength. For higher energy, or more penetrating impacts where there is considerable fibre fracture, the residual tensile strength will also be considerably reduced by impact damage.

Models of delamination buckling are too simple to accurately predict the residual compressive strength of an impact damaged coupon. Real impact damage consists of delaminations throughout the laminate thickness as well as a complex distribution of matrix cracks and fibre fracture. Specifying an equivalent hole size that would result in the laminate having the same residual strength as impact damaged material may be of use to designers. However, because compression of an impact damaged laminate is generally more severe than tension, the size of equivalent hole depends on the mode of loading. Equivalent hole sizes for tensile and compressive loading at two levels of impact damage are given in this thesis.

7.4 Effect of Impact Damage on Fatigue Life

Fatigue after impact damage tests conducted here provide stress-life data that may be used for design. Measurements and observations obtained from impact damaged coupons under cyclic loading indicate the mechanism of fatigue failure.

Tests of 21 ply coupons of T300/914 CFRP under fully reversed tension-compression cyclic loading at 5 Hz indicated a fatigue limit of 0.18 % strain for BVID (barely visible impact damage) (5 J) and 0.13 % strain for severe (10 J) impact damage.

The fatigue stress - life test data obtained can be modelled using a Talreja fatigue life diagram divided into three regions. Region I is the static failure limit, region II is a scatter band that may be modelled with a linear - log rule, and region III is the fatigue limit.

Although undamaged CFRP is generally considered to be linear elastic in behaviour up to the point of failure, and little strain rate dependence is measurable in test to failure of undamaged material, the mechanism of fatigue failure is influenced by time dependent, viscoelastic behaviour.

Viscoelastic behaviour in impact damaged CFRP is characterised by creep in the impact damage zone (which in some tests led to failure under a constant stress), low frequency cyclic hysteresis in the damage zone, and dynamic, high frequency hysteresis in the overall force/extension response of an impact damaged coupon. Although some creep is also measurable in very low speed cyclic loading of undamaged CFRP, the energy absorption in an impact damaged coupon at high frequency is significantly greater.

The origin of the non-linear behaviour probably lies in viscoelastic properties of the epoxy matrix. Much of the hysteresis behaviour measured in undamaged and damaged coupons may be simulated using a simple viscoelastic model with two degrees of freedom. However, the effect of friction between de-bonded fibres and matrix, and between delaminated plies must also be significant in the hysteresis; and also, therefore, in the fatigue mechanism. Further research is needed to more accurately characterise and measure this non-linear behaviour.

Hysteresis in the impact damage zone is the mechanism of energy dissipation that drives damage progression, leading to eventual fatigue failure. Damage progression in CFRP includes both gradual processes and larger, discrete failure events. Under tension - compression cyclic loading the stiffness of an impact damaged coupon gradually decreases in compression and, to a lesser extent, in tension. This progressive change in modulus will, at intervals, be accompanied by the fast fracture growth of delamination buckles of surface plies, or limited tensile cracks. Delamination area, however, will not necessarily increase significantly before final failure occurs.

Experimental observations suggest that the fatigue damage progression in impact damaged coupons is controlled by both cycle-dependent and time-dependent mechanisms. The relative importance of these mechanisms could be investigated by fatigue tests at varying frequencies and cyclic load profiles.

References

- [1] Howatson, A.M., Lund, P.G. & Todd, J.D., "Engineering Tables and Data - 2nd Edition," Chapman & Hall, 1991
- [2] Cantwell, W.J. and Morton, J., "The Significance of Damage and Defects and Their Detection in Composite Materials: A Review," *Journal of Strain Analysis*, Vol. 27, No. 1, 1992, pp. 29-42
- [3] King, J., "Composites take off without a parachute," *Materials World*, Vol. 5, No. 6, June 1997
- [4] Whitehead, R.S. and Mullineaux, J.L., "Qualification of Primary Composite Aircraft Structures," *Advances in Composite Materials and Structures*, AMD - Vol. 82, ASME, December 1986, pp. 97-103
- [5] Cantwell, W.J. and Morton, J., "The Impact Resistance of Composite Materials - A Review," *Composites*, Vol. 22, No. 5, September 1991, pp. 347-362
- [6] Fibredux 914, Information Sheet No. FTA 49f, Hexcel Composites, October 1989
- [7] Robinson, P. and Davies, G.A.O., "Impactor Mass and Specimen Geometry Effects in Low Velocity Impact of Laminated Composites," *International Journal of Impact Engineering*, Vol. 12, No. 2, 1992, pp. 189-207
- [8] Airbus Industrie, "Fibre Reinforced Plastics - Determination of Compression Strength After Impact," *Airbus Industrie Test Method 1.0010*, Issue 1, February 1993
- [9] Boeing, "Advanced Composite Compression Tests," *Boeing Specification Support Standard*, BSS 7260, Rev. C 1988
- [10] Curtis, P.T., "CRAG Test Methods for the Measurement of the Engineering Properties of Fibre Reinforced Plastics," *Royal Aerospace Establishment Technical Report 88012*, February 1988
- [11] Tropis, A., Thomas, M., Bounie, J.L. and Lafon, P., 1993, "Certification of the Composite Outer Wing of the ATR72," *Proceedings of the Institute of Mechanical Engineers Part G: Journal of Aerospace Engineering*, Vol. 209, 1994
- [12] Birt, E.A. "The Applicability of Radiography to the Inspection of Composites," *Structural Materials Centre, DRA, Farnborough*
- [13] Davis, G.R. and Elliott, J.C., "X-ray microtomography scanner using time-delay integration for elimination of ring artefacts in the reconstructed image," *Nucl. Instr. Meth. Phys. Res. A* 394, 157-162, 1997
- [14] Yuan, Q. & Karger-Kocsis, J., "Effects of Interleaving on the Impact and Impact Fatigue Responses of Cross-ply Carbon Fibre/Epoxy Laminates," *Polymers & Polymer*

Composites, Vol. 3, No. 3, 1995

- [15] Delfosse, D. and Poursartip, A., "Energy-based approach to impact damage in CFRP laminates," *Composites Part A*, 28A, pp. 647 - 655, 1997
- [16] Sierakowski, R.L. and Newaz, G.M., "Damage Tolerance in Advanced Composites," Technomic Publishing Co., 1995
- [17] Timoshenko, S.P. and Woinowsky-Krieger, S., "Theory of Plates and Shells - 2nd Edition," McGraw-Hill Book Company, 1959
- [18] Clark, G., "Modelling of Impact Damage in Composite Laminates," *Composites*, Vol.20, No. 3, May 1989, pp. 209-214
- [19] Murray, Y., and Schwer L., "Implementation and verification of fibre composite damage models", *NASA Technical Report AD-A216600*, December 1989
- [20] Tsai S.W. and Wu E.M., "A general theory of strength for anisotropic materials", *Journal of Composite Materials*, Vol. 5, 1971, pp.58-80
- [21] Hashin Z., "Failure criteria for unidirectional fibre composites", Transactions of the ASME, *Journal of Applied Mechanics*, Vol. 47, June 1980, pp.329-334
- [22] Chang, F. & Chang, K., "A Progressive Damage Model for Laminated Composites Containing Stress Concentrations", *Journal of Composite Materials*, Vol. 21, September 1987
- [23] Sun, C.T., "An Analytical Method for Evaluation of Impact Damage Energy of Laminated Composites," *Composite Materials: Testing and Design (4th Conference)*, *AM STP 617*, ASTM, 1977, pp. 427-440
- [24] Lifshitz, J.M., Gov, F., & Gandelsman, M., "Instrumented Low-Velocity Impact of CFRP Beams," *Int. J. Impact Engng*, Vol. 16, No. 2, 1995, pp. 201-215
- [25] Wu, H.-Y.T. and Springer, G.S., "Impact Induced Stresses, Strains and Delaminations in Composite Plates," *Journal of Composite Materials*, Vol. 22, June 1988, pp. 533-560
- [26] Liu, S. & Chang, F., "Damage Development in Cross-Ply Laminated Composites Due to Transversely Concentrated Loading," AIAA-92-2554-CP, pp. 3046-3053
- [27] Finn, S.R. and Springer, G.S., "Delaminations in Composite Plates Under Transverse Static or Impact Loads - A Model," *Composite Structures*, 23, 1993, pp. 177-190
- [28] Robinson, P. and Davies, G.A.O., "Impactor Mass and Specimen Geometry Effects in Low Velocity Impact of Laminated Composites," *International Journal of Impact Engineering*, Vol. 12, No. 2, 1992, pp. 189-207
- [29] Davies, G.A.O., Zhang, X., Zhou, G. and Watson, S., "Numerical Modelling of Impact Damage," *Composites*, Vol. 25, No. 5, 1994

- [30] Majeed, O., Worswick, M.J., Straznicky, P.V. and Poon, C., "Numerical Modelling of Transverse Impact on Composite Coupons," *Canadian Aeronautics and Space Journal*, Vol. 40, No. 3, September 1994, pp.99-106
- [31] Whirley, R.G. & Hallquist, J.O., "DYNA3D - User Manual," Lawrence Livermore National Laboratory, May 1991
- [32] OASYS DYNA3D Theory, Ove Arup & Partners Lecture Notes
- [33] Hallett, S.R., "Small Specimen Impact Testing and Modelling of Carbon Fibre T300/914," *D.Phil. Thesis*, Department of Engineering Science, University of Oxford, 1997
- [34] Hughes, D.A., "Introductory Manual for DYNA3D," *UTC Report No. 73, Rolls Royce University Technology Centre for Solid Mechanics*, Department of Engineering Science, University of Oxford, 25th December 1995
- [35] Curtis, P.T., Gates, J. and Molyneux, C.G., "An Improved Engineering Test Method for the Measurement of the Compressive Strength of Unidirectional Carbon Fibre Composites," *Defence Research Agency Technical Report 91031*, May 1991
- [36] Jenq, S.T., Wang, S.B., and Sheu, L.T., "A Model for Predicting the Residual Strength of GFRP Laminates Subject to Ballistic Impact," *Journal of Reinforced Plastics and Composites*, Vol. 11, October 1992, pp. 1127-1141
- [37] Daniel, I.M. and Ishai, O., "Engineering Mechanics of Composite Materials," Oxford University Press, 1994
- [38] Daniel, I.M., Rowlands, R.E. and Whiteside, J.B., "Effects of Material and Stacking Sequence on Behaviour of Composite Plates with Holes," *Experimental Mechanics*, Vol. 14, 1974, pp. 1-9
- [39] Gibson, R.F., "Principles of Composite Material Mechanics," McGraw-Hill, 1994
- [40] Haeberle, J.G. and Matthews, F.L., "Compression Test Methods for Composites and Theories Predicting the Compressive Strength of Unidirectional CFRP," *Applied Solid Mechanics* - 3, 1989, pp. 66-79
- [41] Soutis, C., Curtis, P.T. and Fleck, N.A., "Compressive Failure of Notched Carbon Fibre Composites," *Proceedings of the Royal Society*, 440, 1993, pp. 241-256
- [42] Prichard, J.C. and Hogg, P.J., "The Role Of Impact Damage in Post-Impact Compression Testing," *Composites*, Vol. 21, No. 6, November 1990, pp. 503-511
- [43] Ishikawa, T., Sugimoto, S., Matsushima, M. and Hayashi, Y., "Some Experimental Findings in Compression-After-Impact (CAI) Tests of CF/PEEK (APC-2) and Conventional CF/Epoxy Flat Plates," *Composites Science and Technology*, 55, 1995, pp. 349-363

- [44] Kachanov, L.M., "Separation Failure of Composite Materials," *Mekhanika Polimerov*, No. 5, 1976, pp. 918-922
- [45] Simites, G.J., Sallam, S. and Yin, W.L., "Effect of Delamination of Axially Loaded Homogeneous Laminated Plates," *AIAA Journal*, Vol. 23, No. 9, 1985, pp. 1437-1444
- [46] Yin, W.L., Sallam, S. and Simites, G.J., "Ultimate Axial Load Capacity of a Delaminated Beam-Plate," *AIAA Journal*, Vol. 24, No.1, 1986, pp. 123-129
- [47] Kyoung, W. and Kim, C., "Delamination Buckling and Growth of Composite Laminated Plates with Transverse Shear Deformation," *Journal of Composite Materials*, Vol. 29, No. 15, 1995, pp. 2047-2068
- [48] Nilsson, K.-F., Thesken, J.C., Sindelar, P., Giannakopoulos, A.E. and Storåkers, B., "A Theoretical and Experimental Investigation of Buckling Induced Delamination Growth," *Journal of Mechanics and Physics of Solids*, 41, 1992, pp. 749-782
- [49] Symons, D.D., "Impact Damage Tolerance of CFRP," UTC Report No. 87, Rolls Royce University Technology Centre for Solid Mechanics, Department of Engineering Science, University of Oxford, August 1996
- [50] Chen, Wu and Yeh, "A Parametric Study of Residual Strength and Stiffness for Impact Damaged Composites," *Composite Structures*, 25, 1993, pp. 267
- [51] Matthews, F.L., Kinloch, A.J., Gilchrist, M.D. and Pournaras, A.V., "Growth and Assessment of Fatigue Damage in Continuous Fibre Reinforced Plastics," *Link Fatigue Project Testable Target (a) - Brochure PVJ5-548D*, Imperial College/Rolls-Royce Plc., April 1992
- [52] Bathias, C., "Fracture and Fatigue of High Performance Composite Materials: Mechanisms and Prediction," *Engineering Fracture Mechanics*, Vol. 40, No. 4/5, 1991, pp. 757-783
- [53] Ratwani, M.M. and Kan, H.P., "Compression Fatigue Analysis of Fiber Composites," *Journal of Aircraft*, Vol. 18, No. 6, June 1981, pp. 458-462
- [54] Rosenfeld, M.S. and Gause, L.W., "Compression Fatigue Behaviour of Graphite/Epoxy in the Presence of Stress Raisers," *Fatigue of Fibrous Composite Materials - ASTM STP 723*, American Society for Testing and Materials, 1981, pp. 174-196
- [55] Swanson, S.R., Cairns, D.S., Gyll, M.E. and Johnson, D., "Compression fatigue Response for Carbon Fiber with Conventional and Toughened Matrices with Damage," *Transactions of the ASME - Journal of Engineering Materials and Technology*, Vol. 115, January 1993, pp. 116-121
- [56] Clark, G. and Van Blaricum, T.J., "Load Spectrum Modification Effects on Fatigue of Impact-Damaged Carbon Fibre Composite Coupons," *Composites*, Vol. 18, No. 3, July 1987, pp.243-251

- [57] Beheshty, M.H. & Harris, B., "Post-Impact Fatigue Behaviour of CFRP and the Growth of Low-Velocity Impact Damage During Fatigue" *International Conference on Fatigue of Composites (ICFC 1)*, 3 June 1997, pp. 355 - 362
- [58] Joseph, E., Ferry, L. & Perreux, D., "Fatigue Damage of Composite Pipes: Experimental Analysis and Modelling," *International Conference on Fatigue of Composites (ICFC 1)*, 3 June 1997, pp. 274 - 282
- [59] Harris, B., Gathercole, N., Lee, J.A., Almond, D.P., Adam, T. and Reiter, H., "Fatigue Life Prediction For Fibre-Reinforced Plastics," *ECCM-7 Seventh European Conference on Composite Materials*, Vol. 1, May 1996, pp. 69-74
- [60] Beheshty, M.H. & Harris, B., "The interaction between low-velocity impact and fatigue of carbon- and glass-fibre-reinforced plastics," *ECCM-8 Eighth European Conference on Composite Materials*, Vol. 1, June 1998, pp. 143-150
- [61] Beheshty, M.H. & Harris, B. "A constant-life model of fatigue behaviour for carbon-fibre composites: the effect of impact damage," *Composites Science and Technology*, 58, 1998, pp. 9-18
- [62] Fawaz, Z. and Ellyin, F., "Fatigue Failure Model for Fibre-Reinforced Materials under General Loading Conditions," *Journal of Composite Materials*, Vol. 28, No. 15, 1994, pp. 1432-1451
- [63] Talreja, R., "Fatigue of Composite Materials: Damage Mechanisms and Fatigue Life Diagrams," *Proceedings of the Royal Society, London*, A378, 1981, pp. 461 - 475
- [64] Beaumont, P.W.R., "The Fatigue Damage Mechanics of Composite Laminates," *Damage Mechanics in Composites, AD - Vol. 12, ASME*, December 1987, pp. 53-63
- [65] Brønstedt, P., Lilholt, H. & Andersen, S.I., "Fatigue damage prediction by measurements of the stiffness degradation in polymer matrix composites," *International Conference on Fatigue of Composites (ICFC 1)*, 3 June 1997, pp. 370 - 377
- [66] Kennedy, J.M. & Serdinak, T.E., "Post-Impact Fatigue of Cross-Plyed, Through-The Thickness Reinforced Carbon/Epoxy Composites," *International Conference on Fatigue of Composites (ICFC 1)*, 3 June 1997, pp. 347 - 354
- [67] Orth, F., Hoffmann, L., Zilch-Bremer, H. & G.W. Ehrenstein, "Evaluation of composites under dynamic load," *Composite Structures*, 24, 1993, pp. 265 - 272
- [68] Cherouali, H., Reynaud, P., Rouby, D., "Influence of interface roughness on fibre/matrix cyclic sliding in monofilament model composites with ceramic, polymer and metal matrix," *International Conference on Fatigue of Composites (ICFC 1)*, 3 June 1997, pp. 157 - 164

