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**COMPRESSION-AFTER-IMPACT STRENGTH OF CARBON
FIBRE REINFORCED PLASTIC LAMINATES**

by
Henry Vietinghoff

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

Department of Mechanical and Aerospace Engineering
The Ottawa-Carleton Institute for Mechanical and Aerospace Engineering

Carleton University
Ottawa, Ontario
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ABSTRACT

An experimental investigation was undertaken into the compression-after-impact strength of quasi-isotropic laminates fabricated from T800H/3900-2 and AS4/3501-6 CFRP composite systems. Test specimens were impacted at various energy levels using a drop-weight impactor. Damage was characterized using ultrasonics, X-radiography and fractography. The results confirmed the staircase pattern of delamination and matrix cracks described by other researchers. Test specimens were loaded in compression and the load-strain histories were recorded. For a few specimens, Shadow Moiré techniques were used to monitor damage behaviour. The compressive strength of both material systems appeared to be governed by the instability of the damaged region. However, for AS4/3501-6 specimens delamination growth was observed prior to failure. The results of the study were used to validate a predictive model for compression-after-impact strength.

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I dedicate this thesis to my fiancée, Shawna MacKinnon.

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Damage tolerant composite structures are designed based upon the assumption that BVID exists at the most critical locations within those portions of the structure which are exposed to the possibility of impact. This requirement necessitates analytical methodologies to predict the residual strength and life of structures containing BVID (Baker et al., 1985; Davidson, 1989).

The Institute for Aerospace Research (IAR) on the National Research Council of Canada (NRC) has undertaken the development of an analytical model for the prediction of compression-after-impact (CAI) strength. The verification of this model by evaluation against experimental data forms the basis of this thesis. The work was performed under a collaborative arrangement between Carleton University and IAR.

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1.0 INTRODUCTION

Advanced composite materials such as carbon fibre reinforced plastics are finding increasing use in the aerospace industry. This trend is a result of a number of distinct advantages that composite materials offer over traditional materials such as aluminum alloys. These advantages include higher strength-to-weight and higher stiffness-to-weight ratios, superior corrosion resistance, improved fatigue performance, and an ability to tailor properties with fibre direction.

Composite materials do, however, suffer some serious limitations. One of these limitations is their response to localized impact loading such as that resulting from a dropped tool or runway debris (Cantwell and Morton, 1991). Typically, this type of loading results in the formation of internal delaminations which extend well beyond the immediate contact area made by the impactor. The delaminated plies are readily buckled under compressive load because of the pronounced loss of flexural stiffness (Poon et al., 1990). The portion of overall load supported by the buckled plies decreases as load is increased. As a result, the buckled plies effectively act as a "soft" inclusion within the laminate. Delamination growth may also be precipitated due to the large out-of-plane stresses developed at the delamination boundary. Either response results in a reduced compressive strength, with reductions in strength of up to 40 % of the undamaged value being common (Baker et al., 1985).

An additional and very significant concern regarding impact damage is that it is often not readily detectable by visual inspection. While internal damage may be extensive, external indications may be limited to a small surface indentation. Damage of this type is often referred to as barely visible impact damage (BVID) (Baker et al., 1985).

Current design philosophy for aircraft structures utilizing composite materials is based on damage tolerance. Damage tolerance is concerned with the ability of a structure to tolerate a reasonable level of damage or defects under service conditions (Niu, 1992).

Damage tolerant composite structures are designed based upon the assumption that BVID exists at the most critical locations within those portions of the structure which are exposed to the possibility of impact. This requirement necessitates analytical methodologies to predict the residual strength and life of structures containing BVID (Baker et al., 1985; Davidson, 1989).

The Institute for Aerospace Research (IAR) of the National Research Council of Canada (NRC) has undertaken the development of an analytical model for the prediction of compression-after-impact (CAI) strength. The verification of this model by evaluation against experimental data forms the basis of this thesis. The work was performed under a collaborative arrangement between Carleton University and IAR.

2. STATE OF THE ART IN MODELLING OF COMPRESSION-AFTER-IMPACT STRENGTH

Several models have recently been developed for the prediction of CAI strength of quasi-isotropic laminates. The accuracy of the predictions is expected to be dependent on how well the model reflects the actual damage state (type and distribution of damage) and how well the model reflects the behaviour of the damaged laminate under compressive load. A review of the current knowledge concerning damage characteristics and damage behaviour is presented prior to a review of the models for the prediction of compression-after-impact strength.

2.1 Impact Damage Characterization

2.1.1 Introduction

Establishment of characteristics which typify the damage state has been complicated by the difficulty in determining the type and distribution of the damage. Adding to the scope of the problem has been the number of variables which can affect the damage state.

Impact damage is difficult to characterize as it is largely internal, three-dimensional in nature, multi-layered and consisting of a multiplicity of failure types. Until quite recently, non-destructive inspection techniques were limited to methods, such as X-radiography and through-transmission ultrasonics, which are only able to provide the planar extent of the damage. Through-the-thickness information on the damage state could only be determined by means of destructive techniques such as sectioning and depling. These techniques are labour intensive, demand careful preparation and require destruction of the specimen (Poon et al., 1992). Recently, a more advanced ultrasonic technique, termed Pulse-Echo Time-of-Flight, has become available. This technique allows the depthwise location and spatial geometries of delaminations to be determined.

However, it also has limitations in that the ultrasonic signal cannot penetrate beyond the first layer of damage encountered.

Damage resulting from impact is a function of a number of variables including material properties, material and layup geometries, and boundary and loading conditions. Various combinations of these variables may yield unique damage characteristics (Ilcewicz et al., 1989).

Progress in identifying typical damage characteristics has been made, however, for several common layup types and materials under specific boundary and loading conditions. The results are as follows.

2.1.2 Characteristic Damage State (CDS)

Three types of damage states, characteristic of impact, have been observed. These types are fibre fracture only (rare), matrix damage only, and combinations of fibre fracture and matrix damage (Dost et al., 1991).

By means of fractographic inspection, matrix damage in quasi-isotropic laminates was found to consist of a network of delaminations and inclined transverse matrix cracks (Boll et al., 1986; Gosse and Hause, 1987). This damage extended well beyond the contact area of the impactor, as shown in Figure 1. Delaminations at neighbouring ply interfaces were linked by inclined transverse matrix cracks. A repeating pattern for delaminations was evident on the cross-sections of quasi-isotropic laminates (Poon et al., 1992); for example, the cross-section shown in Figure 1 shows a pattern of delaminations and matrix cracks which repeats every four plies and coincides with the repeating pattern of the stacking sequence. Poon et al. (1991) found that delaminations were generally more extensive towards the back face. Through-transmission ultrasonics revealed a plan view of the damage which was approximately circular in shape. Thus, the damage volume had a conical form.

Figure 1: Cross-Section of Typical Impact Damage $[45/0/-45/90]_{6s}$ (Poon et al., 1990)

Gosse and Hause (1987), of The Boeing Company, determined that delaminations occurred through the thickness of the quasi-isotropic laminates they studied in a very specific pattern and shape. Through the use of sophisticated ultrasonic techniques they were able to image individual delaminations near the impact and back surfaces. They found that individual delaminations were wedge shaped and appeared to subdivide the circular damage area into octants, for stacking sequences where ply orientation increased in increments of 45° . At each ply interface the delaminations occurred in two opposing octants. Neighbouring octants corresponded to neighbouring ply interfaces. Thus, it appeared that a characteristic "staircase" pattern of delaminations was formed whereby delaminations stepped through the thickness of the laminate in a spiral pattern, with each step corresponding to a change in ply orientation. Because the ultrasonic techniques (TOF) used could not detect damage beyond the first layer of damage encountered, the evidence that the delaminations in fact spiralled through the entire thickness was determined from fractography. The repeating pattern of delaminations, described earlier, which was evident on the cross-sections provided confirmation of the staircase pattern. A graphical reconstruction of the damage pattern observed by Gosse and Hause is shown in Figure 2.

Gosse and Hause determined that the angular boundary and angular location of

individual delaminations was defined by matrix cracks which extended radially outward from either side of the point-of-impact. Within each ply, these cracks were oriented parallel to the fibre direction of the ply and corresponded to the observed inclined transverse matrix cracks. Thus, the angular location of a delamination was determined by the fibre direction of the ply above and below the ply interface. Since the matrix cracks not only bounded the delaminations but also appeared to link neighbouring delaminations, Gosse and Hause concluded that the impact damage was continuously linked through the thickness of the laminate. They further concluded that the linkage effectively resulted in a large plane of fracture which spiralled through the thickness and divided the laminate into a set of sublaminae. It is to be noted that the spiral pattern is interrupted at the midplane where a reversal in the direction of ply orientation occurs.

Figure 2: Plan View of Impact Damage Propagation Through
The First Twelve Plies of a Quasi-Isotropic Laminate
 $[-45/0/45/90]_{3s}$ (Gosse and Mori, 1989)

More recent studies, by Dost et al. (1991) and Avery and Grande (1990) of The Boeing Company, have provided further validation of the circular staircase pattern of damage for additional material types and layups.

A study by Lesser and Filippov (1991), similar to that of Gosse and Hause, arrived at slightly different conclusions concerning the damage characteristics. Lesser and Filippov claimed that the delaminations had a "peanut" shape and were oriented such that their major axis lay along the fibre direction of the lower ply of the delamination interface. Quite a number of studies have found a similar result, but only for very particular layups - typically cross-ply layups having considerable grouping of individual plies, for example Liu (1988).

2.1.3 Factors Influencing the Damage State

As stated, the extent and type of damage created by impact is a complex function of a number of parameters. These include intrinsic variables such as material properties, and layup; and extrinsic variables such as impact energy, impact load and boundary conditions.

Matrix toughness and fibre strength are material properties which have been shown to directly affect the type and extent of impact damage (Ilcewicz et al., 1989). Older composite systems have a relatively brittle matrix. As a result, they are highly susceptible to impact damage (Poon et al., 1990). Manufacturers of composite systems have, over the last decade, improved the impact damage resistance by developing toughened resin systems. Poon et al. (1990) demonstrated that these toughened systems can absorb considerably more elastic energy than the non-toughened systems. Dost et al. (1988) determined that the toughened composite systems required much higher impact energies to obtain the same damage size as the brittle composite systems. Further, they found that both the toughened and brittle systems had similar damage characteristics - the circular staircase pattern of matrix damage described in Section 2.1.2. However, a comparison

of damage descriptions from various studies showed that there are some differences in the damage characteristics between brittle and toughened composite systems. For example, Poon (1990) found that for a brittle composite impact face damage consisted of extensive fibre breakage and matrix crushing confined to the contact area of the impactor. Avery and Grande (1990) observed significant fibre breakage outside of the impactor contact area for toughened composites. They stated that the breakage was due to microbuckling of the fibres.

Ply layup or stacking sequence also affects the damage state. Avery and Grande (1990) observed that the orientation of a ply with respect to its neighbour can determine whether or not the creation of a delamination is favourable. They concluded that, in general, a ply interface is favourable to delamination if a change in ply orientation occurs and if this change in orientation is in the same direction as previous changes (i.e. in an increasing sense). If some ply interfaces are not favourable to delamination then larger delaminations tend to occur at those interfaces that are favourable. Comparison of laminates having equal number of favourable interfaces but different bending stiffnesses indicated that larger delaminations occurred with increased stiffness.

Structural parameters such as ply thickness and laminate thickness affect the CDS and the residual compressive strength. For example, if ply thickness is increased while holding laminate thickness constant, fewer interfaces for delamination result. Thus, delaminations will tend to be larger which has a negative effect on buckling resistance. However, this effect is partly offset by the greater buckling resistance of thicker plies (Ilcewicz et al., 1989).

Increasing laminate thickness results in increased bending stiffness and, as described earlier, Avery and Grande (1990) found that this tended to promote larger delaminations. Dost et al. (1991) proposed that very thick laminates would have asymmetric damage with damage concentrated closer to the impacted surface. No experimental study was conducted to substantiate this claim.

The magnitude of the impact energy affects the extent and type of damage created. Inelastic energy, in low energy impacts, is typically absorbed by damage processes of delamination and matrix cracking. At high impact energies (i.e. where significant impactor indentation occurs) these damage processes may be augmented or dominated by other failure mechanisms. For example, Poon et al. (1991) studied the effects of varying impact energy on the impact damage response of quasi-isotropic laminates fabricated from a brittle composite system (AS4/3501-6). Their results indicated that damage area, as seen in plan view, increased relatively linearly with increasing impact energy up to a threshold level, beyond which little or no further growth occurred. At the threshold, external indications of damage such as crushing at the front surface, and matrix splitting and fibre breakage at the rear surface became evident. With even further increase in impact energy a clearly defined punch-out region was developed.

Impact energy is a function of impactor mass and impactor incident velocity. Variations of these two parameters, even if the impact energy is held constant, has been shown to affect the damage state (Cantwell and Morton, 1991; Lesser and Filippov, 1991). The majority of studies on impact damage, found in the literature, concern "low velocity" impact, where the incident velocity is relatively low (<10 m/s) and the mass of the impactor is relatively high (>4 kg). This type of impact is representative of tool drop, and perhaps also runway debris. Testing is typically performed by means of a drop weight system. Studies involving somewhat higher velocity impact (20 to 100 m/s) and low mass projectiles (generally <1 kg) are also documented (Liu, 1987; Springer et al., 1990). This type of impact is representative of runway debris and hailstones, and the higher velocities are typically achieved by means of gas pressure.

2.2 Compression-After-Impact Strength

It has been well established, by means of compression testing, that impact damage results in a reduction in the compressive strength of a composite laminate (Dost et al., 1991). The degree of reduction has been shown to be dependent on damage size, damage

type and material properties (Ilcewicz et al., 1989).

A number of studies have demonstrated that instability or buckling of the damaged region precedes failure (Dost et al., 1989; Poon et al., 1991). Post-buckling responses have been shown to be dependent on material properties and layup. One type of response appears to involve a redistribution of stress from the damaged material to the undamaged material. Another type of response involves stress redistribution and delamination growth.

The stress redistribution model was proposed by Dost et al. (1989), based on experimental studies on quasi-isotropic laminates fabricated from both brittle and toughened composite systems. These studies indicated little or no delamination growth prior to failure under compression. Dost et al. suggested that the buckled plies would have a reduced stiffness relative to the adjacent undamaged material and that, as load was increased, the buckled plies would effectively carry a smaller portion of the total load. The load redistribution and stress concentrations, next to the damaged material, would result in reduced compressive strength.

Delamination growth can result from the large out-of-plane stresses which develop at the delamination boundary following buckling. Delamination growth has been observed in a number of studies (Kassapoglou, 1988; Peck and Springer, 1989) and appears to be more prevalent with brittle matrix composites and perhaps with specific damage geometries. Generally, the observed delamination growth was in a direction perpendicular to the applied load and resulted in a significant reduction in compressive strength.

Experimental techniques for studying the behaviour of impact damaged laminates under compressive load have largely involved the use of strategically placed strain gauges. For example, Peck and Springer (1989) placed gauges adjacent to the damage area, for detection of delamination growth, and in the damage area, to detect buckling. Poon (1991) applied photoelastic techniques and D-sight in an attempt to better understand

damage behaviour. Isochromatic patterns in the photoelastic coating indicated that high strain concentrations existed in the damage region. D-sight, an optical technique for visual enhancement of surface deformation, provided confirmation of the instability of the damaged material.

2.3 Modelling of Compression-After-Impact Strength

Early analytical studies of buckling and post-buckling behaviour involved simplified models of impact damage. The studies typically focused on a single, near surface delamination. A variety of analyses were developed for strip (through-width), rectangular, circular and elliptical delamination shapes.

For example, Chai and Babcock (1984) developed an analytical model for an elliptical delamination which separates a thin orthotropic layer from a thick isotropic plate. The material axis of the orthotropic layer coincides with the ellipse axis (specially orthotropic case). The plate is assumed to be large with respect to the delaminated area and to undergo in-plane deformation only. Buckling of the sublaminates is determined by using the Rayleigh-Ritz technique. This method essentially consists of assuming an admissible displacement solution, calculating the total potential energy, and applying the requirements of a stationary potential energy and the Trefftz criterion to yield eigenvalue equations.

The model was used for a parametric study of the effects of various factors, such as the aspect ratio and size of the delamination, on sublaminates stability and post-buckling behaviour. Delamination growth is determined by comparing the strain energy release rate against the fracture energy of the material and assuming self-similar delamination growth (i.e. growth occurs equally along the entire delamination front). The analysis showed that, depending on the dimensions of the initial delamination and the fracture toughness, delamination growth can occur stably, unstably or be a combination of both.

Shivakumar and Whitcomb (1984) also developed an analytical model for an elliptical sublaminates. However, it differs from Chai and Babcock's in that the sublaminates can have anisotropic material properties and an arbitrary orientation relative to the loading axis of the laminate.

The model assumes a sublaminates composed of several laminae (plies), each of which can have a unique fibre angle. Sublaminates properties are formulated based upon laminate plate theory, considering that the plate undergoes in-plane deformation only. Forces acting on the sublaminates are obtained by transforming the laminate forces into the axis system of the sublaminates (See Figure 3). Only uniaxial compressive forces are considered as acting on the laminate.

Figure 3: Illustration of the Inplane Loads Acting on the Plate and the Sublaminates (Shivakumar and Whitcomb, 1984)

In-plane dimensions of the sublaminates are assumed to be large relative to its thickness and, thus, thin plate linear buckling is assumed to be valid. A clamped edge condition is used, and a three term Rayleigh-Ritz solution is employed to determine buckling strain. The effects of higher modes of deformation are neglected.

The predictions from the analytical model were compared to a finite element analysis for various sublaminates configurations and properties. Good agreement was

found between the two methods except with highly anisotropic sublaminae where the analytical model fails to account for bending-stretching coupling effects due to the unsymmetric layup of the sublaminate.

Kassapoglou (1988) demonstrated an alternative method to the Rayleigh-Ritz technique for determination of buckling load. His method consists of solving the fully coupled partial differential equations for anisotropic elasticity using infinite series. The proposed solution accounts for bending-stretching coupling and treats the sublaminae, on each side of the elliptical delamination, as generally anisotropic. A strength of materials approach is used to predict delamination growth.

Kassapoglou compared buckling predictions from his model against experimental data. Test specimens were made from sandwich panels with graphite composite face-sheets. Elliptical teflon inserts were implanted during layup to simulate delaminations. Good agreement between theory and experimental data was found. In all cases the delamination grew following buckling, in a direction perpendicular to the loading axis.

Finite element methods have been used to study buckling and post-buckling behaviour of delaminations (Whitcomb, 1988), but are an expensive and time-consuming method.

The development of more damage resistant resin systems over the last decade has led to an interest, by commercial aircraft manufacturers, in utilizing composite materials in primary and highly loaded secondary structures. The need to meet damage tolerance requirements and optimize damage resistance has generated a requirement for predictive methods for residual strength. Several models for actual impact-damaged laminates have been proposed. These models build upon the preceding work, in particular the analysis by Shivakumar and Whitcomb (1984).

For example, Dost, Ilcewicz and Gosse (1988), of The Boeing Company,

developed a model (hereafter referred to as the Boeing model) to predict the residual compressive strength of an impact damaged quasi-isotropic laminate. The model is based upon the CDS which they identified for matrix dominated damage. This CDS, which was described earlier, consists of a spiral pattern of interconnected delaminations and matrix cracks. Dost et al. (1988) assumed that, because individual delaminations are interconnected, they actually form larger circular delaminations. These effectively subdivide the damaged region of the laminate into a series of circular interconnected sublaminates. The model idealizes these sublaminates as being flat and of equal diameter and thickness. Sublaminare buckling load is calculated using the analysis by Shivakumar and Whitcomb. However, because the sublaminates are unsymmetric the reduced bending stiffness method (Ashton, 1969) is employed to account for bending-extensional coupling. Total buckling load is set equal to the buckling load for an individual sublaminare multiplied by the number of sublaminates. The analysis accounts for differences in the stacking sequence of each sublaminare, which could affect the buckling strain, by calculating the buckling loads for all possible sequences and averaging them.

Following buckling, the sublaminates are assumed to support a fixed load regardless of changes in the laminate load (elastic-perfectly plastic modulus). Transverse and shear moduli are modified by the ratio of sublaminare to laminate modulus. The model assumes that sublaminare buckling causes a load redistribution in the laminate. The damaged region is simulated as a reduced stiffness inclusion in an infinite width plate to allow the stresses to be calculated using the theory of anisotropic plates. Finite width correction factors, derived from a finite element analysis, are then applied to obtain the actual stresses. Failure is predicted by the model when undamaged laminate strength is reached based upon a maximum strain criteria.

Out-of-plane stresses were not considered in the Boeing model as the materials and laminates under study appeared to be affected primarily by inplane stress redistribution.

Numerical results from the model were compared with data obtained from experimental investigation and good agreement was found between predictions and test results.

Davidson (1989) developed a model to predict residual compressive strength of a laminate with multiple delaminations. Although details are limited, it appears that the buckling analysis is similar to that of the Boeing model. The method of Shivakumar and Whitcomb is used to determine buckling strain and the reduced bending stiffness approach is employed to account for sublaminates non-symmetry. Buckled plies are assumed to support their individual buckling load until catastrophic failure occurs. The Davidson model differs from the Boeing model in its assumptions concerning the damage state and in the approach used to determine laminate failure.

The damage state is considered to be composed of multiple delaminations which locally divide the laminate into elliptical sublaminae having arbitrary orientation. The sublaminae are not necessarily interconnected and as a result external sublaminae may buckle individually.

The model compares the values of sublaminates buckling load, compression failure load and global buckling load. If either of the latter two are attained the failure is considered catastrophic and the load corresponds to the ultimate. If sublaminates buckling occurs the delamination is assumed to propagate unstably and load bearing capability of the buckled region is eliminated. The analysis is then repeated for the reduced laminate. Although not described, presumably the model assumes delamination growth can only occur at external delaminations. Davidson justifies this approach based on experimental studies involving brittle resin systems. For more ductile systems, Davidson stated that the load carrying capacity of the buckled region would need to be considered and the post buckling analysis developed by Chai and Babcock (1984) could be used.

Predictions from the model were compared with data from an experimental investigation, which used simulated elliptic delaminations, and a good agreement was found.

Xiong and Poon, of the Institute for Aerospace Research at the National Research Council, have recently developed a model for prediction of compression-after-impact strength (1992, 1993). The model is similar to the Boeing model but with a few important differences.

The shape of the damage is assumed to be elliptical rather than circular. The user inputs the dimensions and orientation of the ellipse, and the layup of the sublaminae. For this study, predictions were based on buckling of the largest sublaminate, which is located at the back side. This sublaminate is assumed to buckle independently of the other sublaminae and thus buckling of it occurs prior to buckling of the other sublaminae. Following buckling of the largest sublaminate, the entire damage area is modelled as a soft inclusion.

The buckling analysis accounts for the possibility of an initial out-of-plane imperfection due to permanent deformation caused by impact. A six term shape function is used in the buckling analysis. This function includes three linear terms omitted by the Boeing model. Xiong and Poon state that including these terms allows the model to deal with cases where the structure of the sublaminate (i.e. the fibres) is not symmetric with respect to the axes of the sublaminate. Failure is predicted using a characteristic dimension approach (stress value is determined at a finite distance $\{d_0\}$ from the damage area) in conjunction with an appropriate failure criterion (Xiong et al., 1993). The user can choose between three failure criteria, point stress, maximum stress and the Tsai-Wu.

The model has been verified against the results of other analyses but experimental verification is lacking. The verification will form the basis of this thesis.

3. THE PROBLEM OF VERIFYING A MODEL FOR PREDICTION OF COMPRESSION-AFTER-IMPACT STRENGTH

The primary objective of this research project was to verify, by experimental means, a model for the prediction of compression-after-impact strength, developed by IAR (Xiong and Poon, 1992). The model was to be modified or refined, as necessary, to improve upon the accuracy of the predictions. A further objective was to improve on the understanding of impact damage resistance of composite laminates.

As discussed in the review of residual strength prediction models, a comprehensive evaluation against experimental data is currently lacking for the IAR model. Such an evaluation would include assessment of the impact damage and of its structural significance, experimental investigation of the behaviour of a damaged laminate under compressive load, and comparison of predictions against experimental results.

As described in section 2.1.2, significant progress has been made in characterizing the damage due to low velocity impact in quasi-isotropic layups and in modelling their residual compressive strength. Gosse et al. determined the existence of a characteristic damage state, consisting of a spiral pattern of delaminations and matrix cracks. Based on this damage state and experimental observations, Dost et al. proposed a damage behaviour model (Boeing Model). The IAR model incorporates many of the fundamental assumptions of the Boeing model. However, further investigation into the accuracy and/or limitations of these assumptions is warranted for the following reasons.

- The universality of the CDS described by Gosse and Hause (1987) has yet to be determined. As discussed, parameters such as material properties, layup, loading conditions and boundary conditions all influence the damage state. The current study utilizes two material systems and several layups which were not included in the studies by Dost et al. (1989, 1991) and Gosse and Hause (1987).

- The damage behaviour model put forth by Dost et al. (1989) is based on the premise that no or very limited delamination growth will occur prior to catastrophic failure. Several studies, including those by Kasappogolou (1988) and Davidson (1989), observed significant delamination growth.

- Past studies of damage behaviour under compressive load have utilized strain gauges, visual observation, photoelastic techniques and D-sight. While these studies have yielded valuable information further insight could be gained by using Moiré interferometry techniques. These techniques can provide both qualitative and quantitative information on surface deformation.

- The proportion of fibre and matrix fracture is expected to increase with increased impact energy. Dost et al. (1989) claimed that this type of damage had a negligible effect on residual compressive strength, based on the good agreement between predictions which neglect this damage and experimental results. However, fibre and matrix damage do degrade sublaminates stiffness and must thus have some effect on residual strength.

- The Boeing model assumes circular sublaminate with constant diameter through the thickness. Studies by Poon et al. (1991) and others have shown that the overall damage shape is more elliptical than circular and can increase significantly in size from front to back. Damage shape, orientation and size affect buckling resistance.

Following the verification against experimental data the IAR model will form part of the Composites Design and Analysis System, COMDAS (Bellinger et al. 1993), which is being developed at the Institute for Aerospace Research. Such a system can be used by design engineers to verify compliance with damage tolerance requirements and develop preliminary design details, and by maintenance engineers to assess damage criticality and to develop repair strategies.

The work proposed for this study is expected to improve the understanding of impact damage resistance in composite laminates. Better understanding may lead to the development of more damage resistant materials and structures.

4. EXPERIMENTAL INVESTIGATION OF IMPACT DAMAGE AND COMPRESSION-AFTER-IMPACT STRENGTH

4.1 Introduction

The experimental verification of the IAR model involved a detailed characterization of the damage due to impact, assessment of the structural significance of this damage, verification and/or modification of assumptions within the model concerning the damage state, compression testing, and comparison of predictions and experimental results.

Details concerning the techniques used and results achieved are described in the following sections.

4.2 Test Specimens

4.2.1 Materials and Layup

Two composite material systems, Toray T800H/3900-2 and Hercules AS4/3501-6, were selected for the study. T800H/3900-2 is representative of the most recent generation of toughened composites and has been selected for use on several new airframes (Warwick, 1992). AS4/3501-6 is an older, brittle composite system and was used on the Canadian Forces CF-18 fighter aircraft (Hoa, 1993).

The AS4/3501-6 system was chosen as a baseline against which to assess the performance, in terms of damage resistance and CAI strength, of the T800H/3900-2 system. As well, it would confirm or deny the model validity for a brittle resin system as past studies have indicated that delamination growth is more prevalent in brittle resin systems. Neither the IAR nor the Boeing model consider the possibility of delamination growth.

Material properties for both systems are listed in Table 1, based upon characterization performed at IAR (Gaudert et al., 1993; Poon, 1991).

Material	0° Tensile Modulus (E_{T1}) GPa	90° Tensile Modulus (E_{T2}) GPa	Shear Modulus (G_{12}) GPa	Poisson's Ratio (ν_{12})
Toray T800H/3900-2	152.4	9.205	4.275	0.35
Hercules AS4/3501-6	140.0	8.2	6.2	0.30

(a) Elastic Moduli

Mat'l	0° Tens. Strength (X_{T1}) MPa	0° Comp. Strength (X_{C1}) MPa	90° Tens. Strength (X_{T2}) MPa	90° Comp. Strength (X_{C2}) MPa	Inplane Shear Strength (T_{12}) MPa
Toray T800H /3900-2	2,089.0	1,482.0	79.29	231.00	132.8
Hercules AS4 /3501-6	2,144.3	824.6	46.20	172.40	110.3

(b) Ultimate Strengths

Table 1: Mechanical and Material Properties

The primary focus of the study was on specimens fabricated using a quasi-isotropic layup of $[45/0/-45/90]_{3S}$. This layup is widely used for material qualification testing and is specified in a commonly accepted standard for compression testing of composites, Boeing Specification BSS 7260 (The Boeing Company, 1988). This layup will be referred to as the standard layup throughout this thesis. A limited study into the effects of other layup types on impact damage resistance was also carried out. The layups and the purpose for studying them are listed in Table 2. All layups are quasi-isotropic

and midplane symmetric.

Material	Layup	Effect Studied
T800H/3900-2 & AS4/3501-6	$[45/0/-45/90]_{3n}$	Standard Layup
T800H/3900-2	$[45/0/-45/90]_{4n}$	Thickness
T800H/3900-2	$[45/0/-45/90]_{2n}$	Thickness
T800H/3900-2	$[45_y/0_y/-45_y/90_y]_n$	Grouping of Plies
T800H/3900-2	$[60/30/0/-30/-60/90]_{2n}$	Fibre Orientation
T800H/3900-2	$[75/60/45/30/15/0/-15/-30/-45/-60/-75/90]_n$	Fibre Orientation

Table 2: Test Layups

Due to limitations on material availability the two material systems had different ply thicknesses. The Toray system used was grade 190 with a nominal ply thickness of 0.19 mm (0.0075 in) after curing while the Hercules system had a nominal thickness of 0.14 mm (0.0054 in).

4.2.2 Fabrication

Test specimens were made to meet the Boeing Specification BSS 7260 which specifies coupons measuring 10.2 cm (4 in.) by 15.2 cm (6 in.) in size. Ply orientations (per BSS 7260) are shown in Figure 4.

In accordance with standard practice at SML, the specimens were cut from larger fabricated panels. The individual plies were cut out from a roll of prepreg material with the aid of templates and were laid up by hand to form panels. BSS 7260 calls for +/-1

degree fibre alignment in the completed panels. Therefore, the individual plies were laid up using an alignment jig consisting of two straight edges at right angle to each other. As the plies were laid down they were butted up against the straight edges, thus eliminating the inaccuracies that would result when gauging the alignment by eye. Panels were vacuum bagged and cured in an autoclave in accordance with the manufacturer's recommended cure cycle, listed in Appendix A.

Figure 4: Ply Orientation Convention

Panels were fabricated slightly oversize so that approximately 1.3 cm of waste material could be trimmed off the edges, where fibre misalignment and discontinuities might occur during the curing process. The panels were sectioned using a diamond-coated circular saw blade. Fibre alignment within the specimens was maintained by using the straight edges from the layup process as references. Specimens were machined to correct dimensions and tolerances using an overhead, reciprocating grinder.

4.2.3 Inspection

Prior to sectioning into test specimens the panels were inspected for internal defects using ultrasonic C-scan techniques (reflected through-transmission at 5 MHz). The results indicated that the laminates were all of good quality.

The first fabricated specimen was submitted to the metrology facility at NRC for inspection. This inspection confirmed that the specimen met the dimensional and geometrical tolerances specified in BSS 7260. Therefore, it was concluded that the equipment and the techniques used for machining composites at SML were adequate. All

subsequent specimens were inspected with the aid of a machinist's square, a vernier calliper and a micrometer. The coupons were also inspected for correct fibre orientation and ply symmetry. Specimens that failed the criteria were corrected or discarded.

4.3 Impact Testing

4.3.1 Equipment

Specimens were impacted using a Dynatup Model 8200 drop weight system equipped with a GRC 730-I instrumentation package. The impactor, which is dropped from a fixed distance above the specimen, is instrumented with a load cell for measurement of impactor contact load during the impact event. The system is capable of recording 1024 data points (load measurements) during a time interval of 10 milliseconds. Impactor velocity at the point of impact is required for computation of impact energy and is determined by a flag passing through an infrared detector. The detector also triggers data acquisition. Data is stored in computer memory for subsequent analysis by the system software. The drop tower is equipped with a latching mechanism for the impactor to prevent repeated impacts due to rebound.

The specimen to be impacted rests on, and is secured to, an anvil which is perpendicular to the line of action of the impactor. A rectangular cutout in the anvil, smaller than the test specimen, provides an edge support and allows free deflection of the specimen.

The drop weight system was modified to conform to Boeing Specification BSS 7260 for class 1 impact and the modifications are shown in Figure 5. The specification calls for a 15.9 mm (5/8 in) diameter hemispherical impact tip made of hardened steel, an impactor weight of 53.4 N (12 lbf) and an impact base (bolted to the anvil) made of an aluminum - plywood sandwich. The test specimen is centred over a rectangular cutout in the base and is held in place by four rubber-tipped clamps adjusted to exert light

clamping pressure. According to Avery et al. (1990) of The Boeing Company this arrangement simulates a simply supported edge condition.

Dimensions in mm(in)

Figure 5: Boeing Class 1 Impact Fixture

The test matrix shown in Table 3 indicates the number of impact tests performed for each impact energy level, layup and material type.

	Number of Tests							
Impact Energy (J)	7.6	11.4	15.3	22.9	30.5	40.7	50.8	61.6
Material and Layup								
AS4/3501-6 [45/0/-45/90] _{3S}	2*	3*	1*					
T800H/3900-2 [45/0/-45/90] _{3S}		2*	2*+5 =7	2*	5	5	5	5
T800H/3900-2 [45/0/-45/90] _{6S}			3		3	3	3	3
T800H/3900-2 [45/0/-45/90] _{2S}			3		3			
T800H/3900-2 [45 ₃ /0 ₃ /-45 ₃ /90 ₃] _S			3		3	3	3	
T800H/3900-2 [60/30/0/-30/-60/90] _{2S}			3		3	3	3	
T800H/3900-2 [75/60/45/30/15/0/-15/-30/-45 /-60/-75/90] _S			3		3	3	3	

* - Test Set #2

Table 3: Test Matrix

Two sets of impact tests were performed. The first set involved only T800H/3900-2 specimens. Impacting was performed at the five different impact energy levels specified in Boeing Material Specification BMS 8-276 (The Boeing Company, 1990), namely 15.25, 30.50, 40.67, 50.84 and 61.55 Joules (J). However, if impactor penetration was imminent the impact energy was not increased further.

The second set of tests involved specimens from both T800H/3900-2 and AS4/3501-6. Impact energies were selected to cause minimal back surface damage (See Table 3). This approach was required to allow monitoring of damage behaviour using the Shadow Moiré technique during subsequent compression testing (See Section 4.6).

Specimens were impacted with the tool side facing down. Thus, the layup is $[-45/0/45/90]_{3s}$ when viewed from the impact side and $[45/0/-45/90]_{3s}$ when viewed from the back side.

Impact force and energy histories were recorded for future study concerning impact damage resistance. However, some preliminary analysis of the results for the standard layup type was undertaken.

Strain readings were also recorded during impact by means of a single biaxial gauge mounted on the impact side of the specimen. The gauge was located approximately 36 mm from the point of impact (specimen centre) with one element along the fibre direction of the top ply and the other perpendicular to it. The strain data is used to validate a finite element model being developed to determine the damage created by impact or, in other words, the damage resistance of the material (Straznicky et al., 1993).

4.3.2 Test Results - Material: T800H/3900-2

Layup: $[45/0/-45/90]_{3s}$

Plots of load and energy histories during impact are shown in Figures 6 through 10. The results for specimens impacted at a particular energy level are shown superimposed, and the average and standard deviation values of the key impact parameters are tabulated underneath.

The plots show a good agreement between the test results at any one impact energy level, suggesting that the type and extent of damage created should be similar.

	Impact Velocity (m/s)	Impact Energy (joules)	Time of Max. Load (msec)	Total Time (msec)	Maximum Load (kN)	Energy at Max Load (joules)	Energy Absorbed (joules)
Average	2.39	15.52	2.11	4.44	8.72	15.48	5.43
Std. Dev.	0.00	0.04	0.09	0.01	0.04	0.15	0.26

Figure 6: Impact Energy and Load vs Time, 15.25 J

	Impact Velocity (m/s)	Impact Energy (joules)	Time of Max Load (msec)	Total Time (msec)	Maximum Load (kN)	Energy at Max Load (joules)	Energy Absorbed (joules)
Average	3.37	30.78	1.85	4.85	11.66	27.62	16.16
Std. Dev.	0.01	0.13	0.24	0.08	0.62	2.57	0.64

Figure 7: Impact Load and Energy vs Time, 30.5 J

	Impact Velocity (m/sec)	Impact Energy (joules)	Time of Max Load (msec)	Total Time (msec)	Maximum Load (kN)	Energy at Max Load (joules)	Energy Absorbed (joules)
Average	3.88	40.88	1.52	5.30	11.90	29.63	25.94
Std. Dev.	0.00	0.04	0.22	0.04	0.54	4.03	0.73

Figure 8: Impact Energy and Load vs Time, 40.7 J

	Impact Velocity (m/s)	Impact Energy (joules)	Time of Max Load (msec)	Total Time (msec)	Maximum Load (kN)	Energy at Max Load (joules)	Energy Absorbed (joules)
Average	4.36	51.69	1.22	5.75	11.573	28.29	37.5
Std. Dev.	0.008	0.183	0.119	0.115	0.815	3.63	0.95

Figure 9: Impact Load and Energy vs Time, 50.8 J

	Impact Velocity (m/sec)	Impact Energy (joules)	Time of Max Load (msec)	Total Time (msec)	Maximum Load (kN)	Energy at Max Load (joules)	Energy Absorbed (joules)
Average	4.77	61.84	1.09	6.24	12.126	30.67	48.3
Std. Dev.	0.006	0.15	0.155	0.21	0.49	5.39	1.07

Figure 10: Impact Load and Energy vs Time, 61.6 J

Subsequent ultrasonic and surface damage inspection confirmed this conjecture (See Section 4.4.2.). A definite relationship can be seen between plots at different impact energy levels.

At all of the tested energy levels the load initially increases almost linearly with time but small-scale reversals occur, beginning at relatively low loads. The reversals seem to indicate that damage is being generated in the material. An abrupt change in the loading rate can be seen at 11 kN load level, which is approximately the maximum load attained during the impact event. The load change is followed by a series of large load reversals which are most likely caused by progressive failures in individual plies and subsequent load redistributions. The 11 kN load level can be considered a threshold value. If the threshold is not reached then the majority of impact energy is converted into elastic strain energy. If the threshold is reached then the remaining impact energy is dissipated through damage creation. The threshold load level thus corresponds to a threshold energy level of approximately 29 J as seen in Figure 11, which plots initial impact energy, energy absorbed and energy at maximum load, for each energy level.

The specimens which were impacted at 15.25 J (that is, below the energy threshold) did not attain the threshold load value. Based upon the damage characterization, which is discussed later, damage in these specimens is primarily in the form of delaminations. Damage in specimens impacted beyond the energy threshold consists of delaminations accompanied by significant matrix fracture and fibre breakage.

As the impact energy is increased the duration of the series of load reversals increases as shown in Figure 12, which shows an overlay of typical load-time traces from all energy levels. The increased duration corresponds to the larger amount of impact energy absorbed through damage creation.

Figure 11: Comparison of Impact Energy, Energy Absorbed and Energy at Max Load

Figure 12: Comparison of Load vs Time between Impact Energy Levels

4.3.3 Test Results - Material: AS4/3501-6

Layup: $[\pm 45/0/-45/90]_{35}$

Plots of load versus time and energy versus time during impact are shown in Figures 13 and 14, for AS4/3501-6 specimens impacted at 7.6 J and 15.3 J impact energy respectively. The results differ from that of the T800H/3900-2 specimens as follows.

The onset of large scale load reversals occurs at approximately 3 kN load, as compared to 11 kN for T800H/3900-2. However, impact load continues to increase. From the damage characterization study (Section 4.4.2), it is known that the damage in the specimens impacted at 7.6 J is primarily in the form of delaminations and transverse matrix cracks. Thus, the large load reversals on the ascending portion of the load time plot can be equated with damage processes of delamination and matrix cracking.

The results for the specimen impacted at 15.25 J indicate a much higher proportion of impact energy absorbed inelastically as compared to the specimens impacted at 7.6 J, or T800H/3900-2 specimens impacted at 15.25 J. The load-time plot indicates a maximum impact load of 5.7 kN, followed by a series of large scale load reversals. From the damage characterization study, it is known that significant fibre breakage and matrix fracture, in addition to delamination, occurs at this impact energy level. Thus, the load reversals, following the peak load, may be associated with damage processes of fibre breakage and matrix fracture. The 5.7 kN load level can be considered a threshold value similar to the 11 kN load level for T800H/3900-2.

In summary, the results confirm that the toughened matrix system (T800H/3900-2) can absorb a greater amount of elastic energy than the brittle matrix system (AS4/3501-6). However, a measure of the improvement in damage resistance cannot be based directly on the above results due to the difference in laminate thickness between the two systems tested.

	Impact Velocity (m/s)	Impact Energy (joules)	Time of Max. Load (msec)	Total Time (msec)	Maximum Load (kN)	Energy at Max Load (joules)	Energy Absorbed (joules)
Average	1.68	7.63	3.37	6.87	4.58	7.82	1.81
Std. Dev.	0.00	0.04	0.04	0.10	0.04	0.18	0.02

Figure 13: Impact Energy and Load vs Time, 7.6 J

Impact Velocity (m/s)	Impact Energy (joules)	Time of Max. Load (msec)	Total Time (msec)	Maximum Load (kN)	Energy at Max Load (joules)	Energy Absorbed (joules)
2.37	15.30	2.05	7.68	5.65	10.43	9.27

Figure 14: Impact Load and Energy vs Time at 15.3 J

4.4 Damage Characterization

4.4.1 Techniques and Equipment

Damage was characterized using several different techniques. Surface or external damage was inspected visually and, in a few cases, using Shadow Moiré techniques. Internal damage was characterized by means of ultrasonics, X-radiography and fractography. Details of the inspection techniques and equipment are described in the following sections.

4.4.1.1 Visual Inspection

Visual inspection was limited to observations based on the unaided eye. Damage characteristics were noted and photographic records were taken of a representative specimen from each of the impact energy levels at which the standard layup was tested.

4.4.1.2 Shadow Moiré

Shadow Moiré is an optical method for creating contour lines representative of the out-of-plane deformation of a surface - complete details of the technique are given in Section 4.6.2. The method was used on a few T800H/3900-2 specimens from the standard layup.

4.4.1.3 Ultrasonic Inspection

Two different ultrasonic inspection techniques were employed in this study; the reflected through-transmission C-scan method and the pulse-echo C-scan method coupled with Time-Of-Flight (TOF) software. In both techniques a scan is generated by moving a transducer over the area of interest.

The reflected through-transmission method (conventional C-scan) compares the attenuation of an ultrasonic signal which has been transmitted through the specimen, with a reference signal obtained from an undamaged area of the specimen (Poon et al., 1992). Since the ultrasonic signal travels through the material twice, the method is very sensitive to damage. However, it yields only the overall plan view of the damage and so it was used to confirm the planar shape of the damage obtained by other methods, and to provide an accurate measure of overall damage size. The scans were performed using an automated scanning system coupled to a California Data Corporation Model 644 computer for data acquisition. The transducer (Transmitter/Receiver) frequency was 5 MHz at a 102 mm focus.

In the TOF method, the time required for an ultrasonic signal to travel from the transducer to the defect and to reflect back is compared to the time required for the signal to travel to the rear surface of the specimen and back (DuPont NDT Systems, 1989). The difference in time intervals in conjunction with the specimen thickness allows for determination of the distance of the damage from the surface of the specimen. Thus, the TOF method provides not only the planar view of the damage but also its depthwise location. There are some limitations, however. The ultrasonic waves reflect from the first damage area that they reach so that any damage located behind this area cannot be detected. The impact damage in composites is usually multi-layered and thus can not be fully characterized using TOF method. Specimens were scanned from both front and rear surfaces in an attempt to overcome this limitation. An additional limitation of the method is that it is only sensitive to relatively large flaws such as delaminations and not to minute damage such as fibre fracture and matrix cracking.

TOF ultrasonic scans were done using a portable scanning system, an NDT Instruments Portascan PS1-3000. The system consists of a hand operated mini-scanning frame coupled with a Novascope 3000 ultrasonic thickness gauge and a personal computer. The configuration of the scanning frame, as supplied by the manufacturer, was intended for contact scanning, meaning that the transducer directly contacts the surface

of the material being inspected. Contact scanning has limitations in that defects very close to the surface cannot be resolved, due to merging of the transmitted and reflected pulses and amplifier saturation (DuPont NDT Systems, 1989). As well, surface irregularities (typical on impact damaged composites) affect the orientation of the transducer making the results unreliable. Both problems can be overcome or improved upon by immersing the material and transducer in water, which allows the transducer to be separated from the material.

A previous alteration to the system, to allow it to be used in the water immersion mode, included mounting a plastic spacer on the transducer to provide the correct standoff distance for the transducer from the specimen. However, such an arrangement does not address the problem of surface irregularities as the spacer must contact the specimen surface.

Figure 15: Ultrasonic TOF Scanning Setup

This setup was improved upon for this study by incorporating a number of modifications. These are shown in Figure 15 and include a spacer plate and a new transducer holder. The brass spacer plate has been machined parallel and flat within 0.1 mm, with a large hole at the centre for scanning through. The spacer plate is placed on top of and centred over the specimen. Surface irregularities due to impact lie within the hole and thus do not affect the orientation of the spacer plate. The new transducer holder has a flat plastic base plate which slides on top of the spacer plate. With this new setup the transducer is securely mounted, unaffected by surface irregularities and is maintained at a precise standoff distance.

However, the most important parameter affecting the quality of the results that can be achieved from a TOF scan is the transducer. Transducer frequency, damping and size all have a direct effect on resolution, sensitivity and penetrating power. Selection of the optimum combination is influenced by the material characteristics of the specimen. After some experimentation, the results of which are described in a separate report (Vietinghoff and Straznicky, 1993), a transducer of 5 MHz frequency and 5/16 inch diameter was chosen (NDT Instruments Model C14).

Calibration of the TOF system was achieved by calibrating system output against known thicknesses, representative of the maximum and minimum depth measurements expected to be encountered. Calibration specimens were machined from laminates of the same material and layup as the specimens to be inspected.

Interpretation of the TOF C-scans is as follows. Colours are used to represent damage at different depths with each colour bar representing a depth increment. The colour bars and corresponding depth scales are shown underneath each set of scans. System characteristics dictate a trade-off between a depth increment and a total viewed depth. In this case, a depth increment corresponding to the thickness of one ply results in a total viewing depth which is less than the total specimen thickness.

Since damage characterization through the entire thickness was required, scans at two different viewing depths were produced for each specimen. The first scan provides a ply-by-ply representation of the damage, where each colour bar represents one ply interface. However, the viewing depth is limited to just over one half the specimen thickness. The second scan provides damage information through the complete thickness of the specimen; however, each colour bar represents four to six ply interfaces - equivalent to a major delamination (explained in detail later).

Clarity of some of the scans performed from the rear surface is affected by signal loss as indicated by the black and some white regions in the images. This loss can be attributed to the increased matrix damage towards the rear face and the damage protrusion which can result in reflection of the ultrasonic signal away from the transducer.

4.4.1.4 X-Radiography

X-radiography inspections were used to detect matrix cracks - one aspect where the effectiveness of ultrasonic scanning is limited - and to confirm the overall damage shape. An X-ray opaque penetrant was applied to the specimens to enhance the X-ray images. Inspection was limited to a few specimens representative of the range of impact energies tested and of layup [45/0/-45/90]_{3s}.

Penetrant was applied to the specimens either by direct application over the area of interest or by application assisted by vacuum from the opposite side. Direct application was used for the lowest impact energy specimen as no cracks existed on the impact side for vacuum to draw through. Thus, penetrant was applied to the rear surface, where matrix cracking had occurred, and was allowed to soak for several hours. For the remaining specimens vacuum was applied to the impact side while a pool of penetrant was maintained on the rear side. When considerable penetrant had migrated through to the vacuum side the vacuum was turned off and the penetrant was allowed to soak for several more hours. A water wash was used on all specimens prior to X-raying to

remove penetrant remaining on the surface. Exposure time was 0.7 minutes at a 40 kV, 4 mA setting using standard X-ray film (Kodak Industrex MAA-2).

Matrix cracks are quite faint on the images presented, due to a loss of image resolution in the creation of contact prints.

4.4.1.5 Fractography

Fractographic techniques were used to determine internal damage not accessible or detectable by ultrasonic inspection. Specimens were sectioned in the lengthwise direction along a number of parallel and equi-distant cuts spanning the entire damage area. This was progressively done across the width of the specimen to allow each newly exposed edge to be polished, inspected and photographed. Material removal was done by grinding, using a very light material removal rate of 0.13 mm/cut (0.005 inch) to ensure that the existing impact damage would not be affected.

Fluorescent dye-penetrant coupled with ultraviolet light was used to highlight the damage. A double exposure technique was employed to obtain photographic records. The first exposure captured the damage image as indicated by the fluorescent dye-penetrant. The second superimposed the backlighting outline of the specimen onto the first image.

Because the majority of impacted specimens were required for subsequent compression testing, only one specimen from each impact energy level specified in BMS 8-276 was inspected using fractographic techniques. For AS4/3501-6 specimens sectioning was limited to one specimen from the 7.6 J energy level.

Representative specimens from layups other than the standard layup of $[45/0/-45/90]_{3S}$ were sectioned along the centreline and in some instances also at a distance offset from the centreline.

4.4.2 Results

4.4.2.1 Layup: [45/0/-45/90]_{3s}

Material: AS4/3501-6

Surface Damage

Surface damage on all the specimens consisted of an indent on the impact face, at point-of-impact, and a corresponding bump or bulge on the back face. The bump was accompanied by matrix cracks and in some cases fibre fracture.

The depth of the indent was approximately equal to the height of the bump; however, the bump was larger in diameter. Indent depth/bump height generally increased with increased impact energy. Depths and heights were measured using a mechanical dial gauge and values are listed in Appendix B. The indent and bump can be considered as an initial eccentricity which will tend to reduce the buckling stress of the damaged region.

Matrix cracking occurred at the back face at all impact energy levels. These cracks were parallel to the fibre direction of the back ply and were located at the bump, crossing over and extending slightly beyond it.

Fibre fracture occurred in some specimens at all the tested impact energy levels. At the lowest impact energy of 7.6 J one of the specimens had a small amount of fibre breakage at the top of the bulge. At 11.4 J one of two specimens had fibre fracture at the top of the bulge. Indent depth for this specimen was correspondingly 50 % greater than that for specimens at 7.6 J. At the highest impact energy of 15.3 J (photographs shown in Figure 16) considerable fibre fracture occurred at the back face. Fibre fracture was also evident on the impact face, along the edge of the indent. Indent depth at this energy level was about 240 % greater than that at 7.6 J.

Figure 16: Typical Impact Face and Back Face Damage - AS4/3501-6, 15.25 J

Internal Damage

TOF C-scans for one specimen from each impact energy level are shown in Figures 17 and 18, for scans performed from the impact and back side respectively. The scans indicate that the internal damage has an overall shape which is generally elliptical.

TOF C-scans which show ply by ply information indicate two distinct wedge shaped delaminations at each ply interface. This is most evident on scans produced from the impact side. Each wedge shaped delamination generally scribes an angle of about 45° thus subdividing the total damage area into octants, centred about the point-of-impact. Delaminations at each ply interface lie in opposing octants. The angular orientation of the plies at any given ply interface define the angular boundaries, and therefore also the angular orientation of the delamination which forms there. For example, if the upper ply

Figure 17: TOF C-Scans, Impact Side Mat'l: AS4/3501-6, Layup: [45/0/-45/90]₃₅

1 cm

Figure 18: TOF C-Scans, Back Side Mat'l: AS4/3501-6, Layup: [45/0/-45/90]_{ss}

of a given interface is oriented at 45° and the lower at 0° delamination will occur in the swept area between these two angles, where the angles are defined with respect to the point-of-impact. The apex of each octant is slightly offset from the point-of-impact.

The scans show that neighbouring delaminations correspond to neighbouring ply interfaces. As a result, it appears that delaminations step through the thickness of the laminate in a spiral pattern, with each step corresponding to a change in ply orientation. Because the ultrasonic signal reflects from the first damage area encountered this conclusion cannot be validated by ultrasonic techniques alone. However, the TOF C-scans do indicate the peripheries of delaminations at greater depths (damage area increases in size through-the-thickness) which appear to correspond with the damage pattern near the surface. That is, delaminations which extend beyond the outer edge of those closer to the surface are generally at a depth four plies greater, which corresponds with the four ply repeating pattern of the layup.

Fractographs for one specimen, impacted at 7.6 J, are presented in Figure 19 together with the corresponding TOF C-scan, which indicates the location of cross-sections. Based on the presented fractographs and on others not shown the following observations were made. There is a central core of material, approximately 3.2 mm (1/8 inch) in diameter, which is largely undamaged, except for matrix cracking near the back surface. Delaminations at neighbouring ply interfaces are linked by matrix cracks. These cracks are inclined at about 45° , in a direction towards the point-of-impact. Linkage of neighbouring delaminations results in the formation of "major" delaminations. Major delaminations generally repeat themselves at four ply intervals and progressively increase in size towards the back surface - forming a damage volume which is slightly conical. Delaminations at neighbouring ply interfaces generally do not overlap, with some exceptions, thus confirming that the extent of the delamination is limited to that visible in TOF C-scan. The repetition of the staircase pattern of delaminations, evident on the outer surfaces of the laminate from TOF C-scan, is confirmed through the thickness by the similarity between major delaminations. However, a reversal in the spiral pattern

SEC. A-A

SEC. B-B

SEC. C-C

2 mm

Figure 19: Fractographs (Specimen 826-6, 7.63 J)
AS4/3501-6, [45/0/-45/90]_{ss}

occurs at the midplane. By analyzing the location of the matrix cracks at the various cross-sections it was determined that the cracks appear to be continuous and are aligned with the fibre orientations of the plies. X-radiography results, not shown due to poor image quality, provided additional confirmation of this observation. Thus, matrix cracks both link and bound delaminations, explaining the wedge shaped appearance of delaminations. Matrix cracks are tangential to the central core of undamaged material, explaining why the apex of the delaminations is offset from the point of impact. These findings agree closely with the results of the study by Gosse and Hause (1987).

Quite good agreement of TOF C-scan results with actual damage is evident, except where delaminations are only one or two plies from the surface. A depth of three plies appears to be close to the minimum thickness that the transducer is capable of resolving. As a result, for delaminations which are only one or two plies from the surface the system appears to trigger on the echo from the next deeper delamination, four plies back. For impact-side scans this limitation has little effect on the results as delamination at the second ply interface is very minor. At the back surface, however, the octant shown as grey and purple next to the blue octant should in fact be yellow. Back surface scans are also disrupted by signal loss, indicated by the black and white areas, due to back surface damage. Because of the degree of signal loss and measurement error a graphical reconstruction of the major delamination at the back surface was undertaken based upon the fractographic results. The results are shown in Figure 20 and indicate the individual delaminations along various cross-sections through the damage. Only one half of the damage area is shown. The results generally confirm the prior comments concerning the TOF C-scan image. As well, they indicate that a large area of delamination occurs at a depth of one ply from the back surface.

Figure 20: Back Surface Delamination Based on Fractographic Results

Referring back to the impact side TOF C-scan, the closest delamination to the impact surface, of any significance, occurs between plies three and four (blue). This delamination does not occupy the complete octant as do delaminations at following ply interfaces. By combining this delamination with those at the next three ply interfaces (recall that delaminations are connected by means of matrix cracks) a major delamination (i.e. 360° of revolution) is formed. Modified TOF C-scans where the colour bars have been adjusted to represent major delaminations are presented beneath the scans showing the ply-by-ply information. The major delaminations effectively divide the laminate, in the damaged area, into a series of sublaminate. However, the sublaminate are all interconnected due to the spiral pattern of damage. Each sublaminate is connected to the next by three plies along an axis oriented at 45° - for the major delamination arrangement selected.

The outer sublaminate on the impact side has a thickness which ranges from three plies to six plies, or an average thickness of four and one half plies. Interior sublaminate, with the exception of the midplane sublaminate, have a constant thickness of four plies. The midplane sublaminate varies in thickness but has an average thickness

of four plies. The back side sublaminate has a thickness which ranges from one to five plies or an average thickness of three plies. The overall shape of the back sublaminate has a more pronounced elliptical shape than the impact side sublaminate.

Effect of Impact Energy

AS4/3501-6 specimens were only impacted over a small range of impact energies and the staircase pattern of delaminations was evident in all of these. TOF C-scans of different energy levels are presented side by side to demonstrate the increase in damage size with increase in impact energy. The increase in damage size is relatively linear over the energy levels tested, as shown in Figure 21; however, it is not in direct proportion to the increase in impact energy. This result can be attributed to the increasing significance of fibre breakage and matrix fracture in absorbing inelastic energy at the higher energy levels.

Figure 21: Damage Area vs Impact Energy

4.4.2.2 Layup: [45/0/-45/90]_{3s}

Material: T800H/3900-2

Surface Damage

The impact face and back face of a typical impact-damaged specimen from each of several different impact energy levels are shown in Figures 23 and 24.

Similar to the AS4/3501-6 specimens, surface damage on all the specimens was distinguished by indentation on the impact side and a corresponding outward deformation on the back side. Figure 22 shows Moiré images of the indent and bump for a specimen impacted at 15.25 J. Out-of-plane displacement in this case is about 0.15 mm (.05 mm/fringe). With increase in impact energy an increase in indent depth and bump height was noted for all specimens. Depth and height values for several specimens are listed in Appendix B.

Impact Face

Back Face

Figure 22: Shadow Moiré Images of Deformation due to Impact
(Specimen 667-3A, 15.25 J)

Figure 23: Typical Impact Face and Back Face Damage
T800H/3900-2, [45/0/-45/90]_{1s}

(a) Impact Face - 50.8 J

(b) Impact Face - 61.5 J

(a) Back Face - 50.8 J

(c) Back Face - 61.5 J

Figure 24: Typical Impact Face and Back Face Damage
TR00H/3900-2, [450/-45/90]_{ss}

For specimens impacted at the lowest energies of 11.4 and 15.3 J external damage consisted solely of indentation/deformation and matrix cracks. The cracks crossed over and extended beyond the bump, aligned with the fibre direction of the outer ply (See Figure 22a). At an impact energy of 22.8 J and higher, fibre fracture was evident at both the impact and back faces. On the impact side this was in the form of a single crack which extended outward on both sides of the indent. This crack, seen in Figures 22b through 23b, is perpendicular to the fibre direction of the front ply and its edges form a ridge, indicative of micro-buckling of the fibres. Subsequent fractographic and X-radiographic inspection revealed that the resulting fibre fracture was confined to the outer ply. On the back side, fibre fracture was concentrated at the bump with a greater amount of fibre breakage occurring with each increase in impact energy. In addition, indent depth and size increased and cracks began to form both within the indent and around it. At the highest impact energy levels a number of matrix cracks had formed which intersected and were perpendicular to the micro-buckle cracks.

Back face damage at increasing impact energy was characterized by rear ply splitting and delamination, fibre fracture, and outward deformation.

Internal Damage

TOF C-scans, X-radiographs and conventional C-scans for one specimen from each impact energy level are shown in Figures 25 to 30. For the higher impact energy levels of 40.7, 50.8 and 61.6 J no back side TOF C-scans are presented as extensive damage at the back surface resulted in poor quality images. Conventional C-scans were included as they show the relationship of the damage area relative to the entire specimen.

The scans and X-radiographs for impact energy levels which result in minimal surface damage indicate that the damage has an overall shape which can be approximated by an ellipse. With increased impact energy the damage shape becomes more irregular

Figure 25: TOF C-Scans, Impact Side Mat'l: T800H/3900-2, Layup: [45/0/-45/90]_{3s}

Figure 26: TOF C-Scans, Back Side Mat'l: T800H/3900-2, Layup: [45/0/-45/90]_{3s}

Figure 27: TOF C-Scans, Impact Side Mat'l: T800H/3900-2, Layup: [45/0/-45/90]₃₅

X-Radiographs: Full Scale
C-Scans: 0.6 Scale

Specimen
Orientation
Marker

15.25 J (Specimen 666-1A)

30.5 J (Specimen 668-2A)

Figure 28: X-Radiographs With Corresponding Conventional C-Scans
Mat'l: T800H/3900-2, Layup: [45/0/-45/90]_{3s}

X-Radiographs: Full Scale
C-Scans: 0.6 Scale

40.7 J (Specimen 669-3A)

50.8 J (Specimen 663-4A)

Figure 29: X-Radiographs With Corresponding Conventional C-Scans
Mat'l: T800H/3900-2, Layup [45/0/-45/90]_{3S}

X-Radiographs: Full Scale
C-Scans: 0.6 Scale

61.6 J (Specimen 663-5A)

Figure 30: X-Radiographs With Corresponding Conventional C-Scans
MatTM: T800H/3900-2, Layup: [45/0/-45/90]_{3S}

as damage, corresponding to rear ply splitting, fibre fracture and local delamination, extends outward from the central core of damage. X-radiographs indicate the core area particularly well as it appears as a darkened region. The central core area is generally elliptical in shape although less so as impact energies approach that resulting in impactor penetration.

The circular staircase pattern of delaminations, described for AS4/3501-6 specimens, is also evident in the TOF C-scans for T800H/3900-2 specimens, particularly at the lower impact energies (See Figures 25 to 26). However, the shape and location of individual delaminations is not as well defined. That is, individual delaminations may extend somewhat into neighbouring octants and have an angular location slightly offset from that described for the AS4/3501-6 specimens. As well, a greater variability was seen in the TOF C-scan results at a given impact energy level for specimens fabricated from T800H/3900-2 as opposed to AS4/3501-6.

Fractographic results for specimens at the lower impact energies, shown in Figures 31 to 33, indicate that delaminations at neighbouring ply interfaces often overlap slightly - i.e. the matrix cracks do not necessarily bound the delaminations. Major delaminations, while displaying the staircase pattern, show greater variation in appearance through the thickness than for AS4/3501-6 specimens. However, the four ply repeating pattern between major delaminations is retained. Towards the back surface the aspect ratio of individual delaminations to major delaminations may be skewed with respect to that observed at the front or impact side. Backside scans of AS4/3501-6 specimens also show this characteristic to some extent.

Because T800H/3900-2 plies are thicker than those of AS4/3501-6 the resolution problem encountered with the latter was not as significant for T800H/3900-2 scans. However, based on the fractographic results some measurement error did result. As well, signal loss occurred due to back surface damage. For these reasons a graphical reconstruction was undertaken, based on fractographic results, for a specimen impacted

C-scan

Sec. A-A

Sec. B-B

2 mm

Figure 31: Fractographs (Specimen 663-1A, 15.25 J)

Representative Damage
Area Indicating X-section
Locations

SEC. A-A

SEC. B-B

SEC. C-C

1
2 mm

Figure 32: Fractographs (Trial Specimen, 13.55 J)
T800H/3900-2

SEC. A-A

SEC. B-B

SEC. C-C

2 mm

Figure 33: Fractographs (Specimen 668-2A, 30.5 J)
T800H/3900-2

at 15.25 J. The results, which are shown in Figure 34, indicate the extent of individual delaminations which make up the major delamination closest to the back surface. Again, delaminations are only shown for one half of the specimen. The results indicate that the closest delamination to the back surface occurs at two plies depth as opposed to one ply depth for AS4/3501-6.

Figure 34: Back Surface Delaminations based on Fractographic Results

For specimens impacted at 15.25 J the closest delamination to the impact surface of any consequence occurs between plies three and four (blue). However, indications of delamination are evident between plies one and two (white) and between plies two and three (yellow). The impact side sublaminates have an average thickness of four and one half plies. The back side sublaminates have an average thickness of three and one half plies, as compared to three plies for the AS4/3501-6 specimens. Interior sublaminates have a constant thickness of four plies, except the midplane sublaminates which have an average thickness of four plies.

Effect of Impact Energy

T800H/3900-2 specimens were impacted over a range of energies, from that resulting in minimal surface damage to that resulting in partial penetration of the impactor. The effects of impact energy on the damage state are summarized as follows.

a) Staircase Pattern of Damage:

The circular staircase pattern of delaminations and matrix cracks is readily identifiable up to the 40.7 J energy level. Beyond 40.7 J the staircase pattern is recognizable only at the peripheries of the total damage area, and may involve considerable overlap of delaminations at neighbouring ply interfaces (See Figures 35, 36 and 37).

b) Fibre and Matrix Fracture:

Fibre breakage and matrix fracture (referring to a high concentration of matrix cracks) is evident at an impact energy of 22.8 J and higher. This type of damage is concentrated near the centre of the total damage area, and at the lower impact energies primarily at the back side of the specimen. At the higher energy levels the fracture zone extends from front to back. Fibre breakage generally occurs at the very centre of the damage area. The microbuckling of fibres evident at the impact face appears to affect only the outer ply.

c) Damage Symmetry:

As described earlier, the damage tends to increase in size towards the back face, resulting in a damage profile through-the-thickness which is asymmetric. At the highest impact energies of 40.7, 51.4 and 61.6 J the asymmetry may be quite pronounced.

d) Damage Size:

The effect of impact energy on overall damage size is shown in Figure 38. Damage size is separated into total and core damage areas. Core damage area refers to

the central damage area which consists of delaminations through the thickness of the laminate. This area may have an elliptical shape. Total damage area refers to core area plus the damage area which extends beyond it due to damage at the back surface. The results indicate that core damage area increases with increased impact energy but that damage growth is a decreasing function of impact energy. Beyond the third impact energy level of 40.67 J, growth of the core damage area was quite reduced. Total damage area, on the other hand, appears to increase with increased impact energy throughout the entire range of impact energies tested.

C-Scan

Figure 35: Fractographs (Specimen 668-3A, 40.67 J)
T800H/3900-2

C-Scan

SEC. A-A

SEC. B-B

SEC. C-C

2 mm

Figure 36: Fractographs (Specimen 668-4A, 51.4)
T800H/3900-2

C-Scan

SEC. A-A

SEC. B-B

SEC. C-C

2 mm

Figure 37: Fractographs (663-6A, 61.6 J)
T800H/3900-2

Figure 38: Damage Area vs Impact Energy

4.4.2.3 Results - Other Layups

Material: T800H/3900-2

Damage assessment of the other layups was limited to TOF C-scan inspection and fractographic inspection of a representative sample(s). This sample was generally chosen from the low end of the range of impact energies, where the least matrix and fibre fracture had occurred. Damage area was determined for all specimens, based on conventional C-scan.

$[45/0/-45/90]_{6s}$

As expected, the results for the $[45/0/-45/90]_{6s}$ layup indicate that increasing the laminate thickness improves the impact damage resistance. At the lowest impact energy of 15.25 J there was no indication of delamination in either of the two test coupons. At the next higher level of impact energy, 30.5 J, TOF C-scan indicated a small, circular damage area (See Figure 39). Actual indication of delamination was confined to a small central area within the larger damage area. Fractographic results, shown in Figure 40, revealed minor delaminations near the impact face as well as significant matrix cracking and minor delamination in the back half of the specimen.

At impact energy levels of 40.7, 50.8 and 61.6 J the TOF C-scans indicated a significant increase in the size of delaminations. TOF C-scans for a specimen impacted at 50.8 J are shown in Figure 39. The pattern of delaminations, indicated by the ply by ply scan, is the staircase pattern observed in the standard layup. However, there is a considerable difference in the size of individual delaminations near the back face relative to delaminations throughout the rest of the laminate. This is most evident on the cross-section shown in Figure 40. The larger delaminations also follow the staircase pattern of damage and combine to form a significantly larger major delamination. The net result is an outer sublaminar at the back face having an average thickness of 6 plies next to a

Back Side (50.8 J)

Back Side (50.8 J)

1 cm

Impact Side (50.8 J)

Impact Side (50.8 J)

48 Piles

Impact Side (30.5 J)

Impact Side (30.5 J)

MAJOR DELAMINATIONS

PLY BY PLY DELAMINATIONS

Figure 39: TOF C-Scans, Impact Side Mat'l: T800H/3900-2, Layup: [45/0/-45/90]_{ks}

Figure 40: Fractographs - $[45/0/-45/90]_{6S}$
T800H/3900-2

series of significantly smaller sublaminates of an average thickness of four plies.

Conventional C-scans indicated that the overall shape of the damage was generally circular at all the energy levels tested. A plot of damage size against impact energy is shown in Figure 41. The results indicate that damage size increased progressively with each increase in impact energy.

$[45/0/-45/90]_{2s}$

The $[45/0/-45/90]_{2s}$ layup was impacted at 15.25 and 30.5 J. Testing was halted at 30.5 J as impactor penetration was imminent. TOF C-scans and fractographs for one specimen, impacted at 15.25 J, are shown in Figures 43 and 44, respectively. The results indicate a staircase pattern of delamination and matrix cracks similar to the standard layup. This damage is accompanied by considerable matrix and fibre fracture in the back half of the specimen, concentrated primarily near the centre of the total damage area. A plot of damage size against impact energy is shown in Figure 42. The results indicate that a doubling in impact energy resulted in a doubling in damage area, despite a significant increase in fibre fracture and matrix cracking.

$[45_3/0_3/-45_3/90_3]_s$

C-scans of specimens having the $[45_3/0_3/-45_3/90_3]_s$ layup indicated a large propeller shaped damage area oriented along the fibre direction of the outer ply.

TOF C-scans and fractographs for one specimen, impacted at 15.25 J, are shown in Figures 45 and 46, respectively. The results show that delaminations occur primarily at interfaces where a change in ply orientation takes place. This results in a three ply repeating pattern between delaminations, except at the midplane where six plies separate the delaminations. Delaminations appear to be oriented along the fibre direction of the lower ply of the interface. The impact side scan indicates that delaminations increase,

Figure 41: Damage Area vs Impact Energy

Figure 42: Damage Area vs Impact Energy

Figure 43: TOF C-Scans, Impact Side Mat'l: T800H/3900-2, Layup: [45/0/-45/90]_{2s}

Figure 44: Fractographs - $[45/0/-45/90]_{25}$
T800H/3900-2

Figure 45: TOF C-Scans, Impact Side Mat'l: T800H/3900-2, Layup: [45/0/-45/90]₃s

TOF C-Scan
Specimen 684-1, 15.25 J

SEC. A-A

SEC. B-B

Figure 46: Fractographs - $\{45,0/-45,90\}_s$
T800H/3900-2

rather significantly, in size through the depth up to the midplane. Scans from the back face indicate large wedge shaped delaminations which form a circular staircase pattern of damage. Matrix cracks are evident in the fractograph shown in Figure 46 and presumably link delaminations. However, this could not be confirmed based upon the limited fractography undertaken.

A plot of damage size against impact energy is shown in Figure 47. The results indicate that damage size increased progressively with increase in impact energy.

[60/30/0/-30/-60/-90]_{2c}

C-scans of specimens having the [60/30/0/-30/-60/-90]_{2c} layup indicated an overall damage shape which was generally elliptical for all the impact energies tested.

TOF C-scans and fractographs for a specimen impacted at 15.25 J, are shown in Figures 49 and 50, respectively. They indicate that delamination and matrix damage follows a circular staircase pattern with delaminations occurring at 30° intervals. With each step through the thickness, delaminations, on the impact side, extend outward a little further. As a result, major delaminations show a greater variation in size and shape than was observed for the standard layup. Based upon limited results an average sublaminar thickness of six plies appears to be valid.

A plot of damage size against impact energy is shown in Figure 48. The results indicate that the rate of increase in damage size drops off after 30.5 J.

[75/60/45/30/15/0/-15/-30/-45/-60/-75/90]_{2c}

C-scans of specimens having this layup showed an overall damage shape which more closely resembled a figure eight than an ellipse.

2 of/de 2

PM-1 3½"x4" PHOTOGRAPHIC MICROCOPY TARGET
NBS 1010a ANSI/ISO #2 EQUIVALENT

PRECISIONSM RESOLUTION TARGETS

Figure 47: Damage Area vs Impact Energy

Figure 48: Damage Area vs Impact Energy

Figure 49: TOF C-Scans, Impact Side Mat'l: T800H/3900-2, Layup: [60/30/0/-30/-60/90]_{2s}

Figure 50: Fractographs - $[60/30/0/-30/-60/90]_2$
TR00H/3900-2

TOF C-scans and fractographs for a specimen impacted at 15.25 J, are shown in Figures 51 and 52, respectively. Clarity of the scans is affected by considerable signal loss. Distinct, well developed delaminations are not evident until about the sixth or seventh ply interface. A staircase pattern is evident. Impact side scans show delaminations up to the midplane forming a major delamination which has the shape of a long elongated ellipse oriented along an axis at 75°. From the midplane to the backside delaminations combine to form a major delamination which is larger than the one on the impact side and which is oriented at about 45°.

A plot of damage size against impact energy is shown in Figure 53. The results indicate a doubling in damage size from 15.25 to 30.5 J, followed by a much reduced rate of increase in damage size from 30.5 to 50.8 J.

4.4.2.4 Damage Size Comparison

A plot of damage area against impact energy for all the layups is shown in Figure 54. The average value of the damage areas, for a given impact energy level, were used. Damage areas for the standard layup are based on the core area.

The results indicate that material properties have a significant effect with respect to damage size. Although test specimens from the two material types tested cannot be directly compared due to differences in ply thickness, the results nevertheless indicate the significantly higher damage resistance of the toughened composite system. For example, at 11.4 J impact energy the AS4/3501-6 specimens had an average damage area equal to 5.89 sq.cm. while the T800H/3900-2 specimens had an average damage area of 1.77 sq.cm., or 70 % less.

The large damage sizes of the $[45_3/0_3/-45_3/90_3]_s$ layup, relative to the other layups, confirms that if fewer interfaces favourable to delamination are provided larger delaminations will occur at them.

Figure 51: TOF C-Scans, Impact Side Mat'l: T800H/3900-2, Layup: [75/60/45/30/15/0/-15/-30/-45/-60/-75/90]_{2s}

Figure 52: Fractographs - [75/60/45/30/15/0/-15/-30/-45/-60/-75/90]_s
T800H/3900-2

Figure 53: Damage Area vs Impact Energy

Figure 54: Damage Area of all Layups vs Impact Energy

T800H/3900-2 layups of $[45/0/-45/90]_{3S}$, $[60/30/0/-3/-60/90]_{2S}$ and $[75/60/45/30/15/0/-15/-30/-45/-60/-75/90]_S$ show large increases in damage size between 15.25 and 30.5 J. Beyond 30.5 J, increases in damage size are relatively small. This result can be attributed to the increasing significance of energy absorption by fibre fracture and matrix cracking beyond the 30.5 J impact energy level.

Damage sizes for the $[45/0/-45/90]_{2S}$ layup, at 15.25 and 30.5 J, are similar to the standard layup, despite a thickness lower by 33 %. A comparison of specimens from the two different layups revealed a much higher amount of fibre fracture, matrix cracking and indentation in the thinner layups at these energy levels.

4.5 Implications of the Damage Characteristics on Modelling of Compression-After-Impact Strength

4.5.1 Standard Layup - $[45/0/-45/90]_{3S}$

Based upon the results of the damage characterization, the following conclusions were drawn concerning the implications of the damage characteristics on the residual compressive strength and its modelling.

- The linkage of individual delaminations by means of matrix cracks results in the formation of larger planes of delamination (i.e. major delaminations) which divide the laminate into a series of sublaminates. The buckling strength of these sublaminates is expected to be a key factor affecting the residual compressive strength.
- Due to the spiral pattern of damage, the sublaminates are all linked together and thus are constrained to buckle in unison. At impact energy levels resulting in minimal surface damage there is an additional linkage between sublaminates through the central core of undamaged material. At impact energies which result in or approach impactor penetration the spiral pattern of damage begins to break down and it is

conceivable that sublaminae could buckle individually.

- The sublaminae vary in size, planar shape and orientation through the thickness. Size tends to increase towards the back face. This is particularly pronounced in many of the T800H/3900-2 specimens impacted at the high end of the impact energy range. Basing the sublaminate buckling analysis on the dimensions of the backside sublaminate appears to be a reasonable and conservative approach. Approximating the damage shape as an ellipse rather than as a circle more closely matches the actual damage shape.

- The sublaminae vary in thickness. In general, interior sublaminae, with the exception of the midplane sublaminate, have a constant thickness of four plies; however, stacking sequence between octants varies. The outer and midplane sublaminae have variable thickness. In general, the impact side sublaminate has an average thickness of four and one half plies and the midplane sublaminate has an average thickness of four plies. The backside sublaminate has an average thickness of three and one half plies for T800H/3900-2 specimens and three plies for AS4/3501-6 specimens. At high impact energy levels (i.e. approaching impactor penetration) the aforementioned values do vary. The collective buckling stress of the sublaminae is expected to be the average of the individual buckling stresses multiplied by the number of sublaminae.

- The layup of the sublaminae is unsymmetric. This will lower the stress at which buckling occurs.

- Buckling stress depends on stacking sequence, but stacking sequence for sublaminae changes from octant to octant. Thus, a simplifying assumption must be adopted to determine the buckling stress. Predictions for the current study were generated (IAR Model) with the assumption that the stacking sequence of all sublaminae corresponds to the layup of the laminate. This contrasts with the Boeing

model, which averages the buckling stress of the four different stacking sequences found in an interior sublaminate. Predictions using both methods are compared with experimental results in Section 5.

- The indentation and bulge which result from impact can be thought of as an initial eccentricity which will tend to reduce the buckling stress of the sublaminate.
- Matrix and fibre fracture is expected to degrade the material properties of the sublaminate, as well as to cause stress concentrations. Matrix and fibre fracture damage was evident at 30.5 J impact energy for T800H/3900-2 specimens and became more significant as impact energy was increased. However, this type of damage was generally concentrated near the centre and backside of the total damage area. Fractographs of specimens at all impact energies showed delaminations extending beyond the matrix and fibre fracture zone.
- Overlap of delaminations at neighbouring ply interfaces, evident in specimens subjected to the higher impact energies of 40.7, 50.8 and 61.6 J, is expected to further reduce the stiffness of the sublaminate.
- Contact forces generated between sublaminate during buckling should increase the collective buckling stiffness of the sublaminate.

4.5.2 Other Layups

Based upon the results of a limited damage characterization study, it was concluded that many of the implications discussed for the standard layup will apply also to the other layups. This includes the effects on buckling stress from indentation/outward deformation, sublaminate non-symmetry, linkage between major delaminations, etc. Implications specific to each layup type are as follows.

- Residual compressive strength of the $[45/0/-45/90]_{6s}$ layup will be affected by the asymmetric nature of the damage through the thickness. For specimens impacted at 40.7 to 61.6 J impact energy, the damage could be modelled as a series of ten circular sublaminates having an average thickness of four plies interconnected with a significantly larger circular sublaminate near the back surface having an average thickness of six plies. The collective buckling stress would be a weighted summation of the individual buckling stresses. For impact energies below 40.7 J sublaminate stability may not be significant; however, such a conclusion must be based upon a detailed ultrasonic inspection of the damage from both impact and back surfaces.

- Impact damage in the $[45/0/-45/90]_{2s}$ layup is similar to that found in the standard layup except that matrix and fibre fracture initiate at lower impact energies. Modelling strategies used for the standard layup should be appropriate for this layup as well. However, due to the extent of fibre and matrix fracture, it may be appropriate to calculate sublaminate buckling load based on degraded material properties.

- The behaviour under compressive load of an impact damaged laminate having the $[45_3/0_3/-45_3/90_3]_5$ layup is expected to be quite different from that of the standard layup. TOF C-scans indicated a large plane of delamination (light blue) near the backside, at a depth of three plies from the surface. Buckling of the resultant sublaminate could promote global instability prior to buckling of thicker sublaminates. Experimental investigation is required to determine the behaviour of this layup under compressive load.

- Modelling strategies used for the standard layup should be appropriate for the $[60/30/0/-30/-60/90]_{2s}$ layup, with the following modifications. Sublaminates will have an average thickness of six plies. Due to the significant increase in the size of major delaminations through the thickness, a buckling stress based on an average of the impact and backside sublaminate dimensions may be more accurate.

- Based on the damage state for the $[75/60/45/30/15/0/-15/-30/-45/-60/-75/90]_s$ layup, it is expected that buckling of delaminations close to the surface will lead to global instability prior to further local buckling. Experimental investigation is required to determine the behaviour of this layup under compressive load.

4.6 Compression-After-Impact Testing

4.6.1 Introduction

The residual compressive strength of impact-damaged specimens was determined by loading them to failure in compression. The results were used for verification of the IAR prediction model. Two separate sets of tests were conducted.

The first concerned T800H/3900-2 specimens with the standard layup of $[45/0/-45/90]_{3s}$. Three specimens from each of the five standard impact energy levels (specified in BMS 8-276) were loaded to failure. Data acquisition was limited to monitoring of far-field strains and applied load.

The second set of tests concerned specimens from both T800H/3900-2 and AS4/3501-6 material. Testing was also limited to the $[45/0/-45/90]_{3s}$ layup. The objectives of the second set of tests were to collect additional test data for verification of the IAR model, to monitor the out-of-plane deformations of the damaged material and to survey strains on, near to and far from the damage site, with increase in compressive load

4.6.2 Equipment and Test Methods

4.6.2.1 Equipment and Test Methods - Test Set #1

A support fixture as specified in BSS 7260 and shown in Figure 55 was used for compression testing. The fixture provides a simple support to the side edges of the specimen to prevent global buckling, and a clamped support to the top and bottom edges to suppress end "brooming" (an edge delamination). Two pairs of back-to-back strain gauges were mounted on the specimen to measure far-field strain and to detect the presence of global buckling. Compressive load was applied under stroke control at a displacement rate of 1.27 mm/min using an MTS 880 load frame and data acquisition system. The load frame had a 445 KN (100 KIP) load capacity and was hydraulically driven. A gimbaled lower platen was used to minimize induced bending moments due to misalignment of the load frame and/or out-of-squareness of the specimen.

Figure 55: Compression Support Fixture

4.6.2.2 Equipment and Test Methods - Test Set #2

Equipment and test procedures for the second set of tests were similar to the first with the following exceptions. Only one pair of far-field strain gauges was used in order to reduce setup time and to conserve the majority of a limited number of data acquisition channels for the monitoring of local strains. Because the MTS 880 system was not available compression testing was conducted on an MTS 980 KN (220 KIP) load frame. This load frame was not equipped with a gimballed platen and, as a result, shims were used to minimize unwanted bending strains. Shimming was deemed acceptable when all axial strains were within about 10 % of each other at a preload equal to 10 to 20 % of expected failure load. A Sciometric Instruments System 200 data acquisition system was used for load and strain data collection.

Back-to-back strain gauges were typically installed in, adjacent and slightly offset from the damaged area (based on TOF C-scan results). However, no gauges were placed directly on the backside damage area when Moiré methods were employed. Approximate gauge locations are indicated on the diagrams accompanying each set of strain results. A small gauge (biaxial, 2mm square) was selected for use in the damage area, in part to minimize any effect of the stiffness of the strain gauge on the response of the sublaminates. Extra care was taken in application of this and other gauges near the damage area to ensure that spreading of the adhesive beyond the gauge contact area was minimized.

Moiré interferometry techniques were used to observe and quantify out-of-plane deformations. The two Moiré techniques commonly used to observe out-of-plane deformations are Shadow Moiré and Projection Moiré. Both are based on a similar principle. If light is projected through a grating (a set of parallel equally spaced lines on a transparent medium) onto an irregular surface the light reflecting off the surface will interfere with the grating in a manner which results in the creation of fringes - where the fringes represent surface topography in a manner analogous to contour lines on a map.

Two parameters control the degree of resolution or surface definition that can be achieved. These are the angle of the light source with respect to the surface being viewed and the spacing between the lines (line density) on the grating. As the angle is decreased or the line density is increased the resolution improves but at the expense of fringe clarity (Kotanen and Komorowski, 1991). To meet the objectives of this project a high degree of resolution was desirable.

The Shadow Moiré method uses a grating which is held very close to the surface of the specimen. At failure or under dynamic loading the required proximity can become problematic due to interference between the specimen and grating. In Projection Moiré two separate gratings are used. Light is projected through one grating onto the surface using an instrument similar to a photographic slide projector. The second grating is located in a separate receiving/viewing station. By using very high resolution (high line density) gratings the magnification effect of projection is countered. The advantage of this technique is that the projector and viewing station can be positioned some distance away from the surface under examination (Schwarz, 1988)(Pirodda, 1982). Thus, surface deformation under compressive load poses no threat to the grating. For this reason, as well as several others, the Projection Moiré technique appeared to be better suited to the requirements of the current study.

However, after considerable experimentation with a Projection Moiré system, recently acquired by IAR, it was determined that the method is extremely sensitive to alignment and that it is not suitable for the current work.

As a result, it was necessary to use the Shadow Moiré technique. As mentioned, a number of problems existed in the application of the technique to the current work. The following summarizes the problems and the solutions used to overcome them.

- a. Because initial deformation and expected buckling deformation are small, a very high line density grating is required to achieve sufficient resolution (i.e. a minimum of

two fringes for the initial deformation). The Moiré image shown in Figure 22 was achieved using a 40 line pairs/mm grating in direct contact with the top of the specimen and an illumination angle of 45°. Moving the grating away from the surface as little as 0.25 mm (0.01 inch) resulted in a loss of image. Even with a grating of 20 line pairs/mm a loss of some fringes occurred at this distance. The proximity problem was compounded by the need for extra allowance for out-of-plane deformation as the grating was not expendable. After considerable experimentation, it was determined that a setup using a grating of 10 line pairs/mm and a very large illumination angle would meet the absolute minimum resolution requirements.

b. The compression support fixture is quite deep and thus a large illumination angle was not possible. This was overcome by mounting a mirror on the inside wall of the fixture and reflecting the light off it.

c. Due to the small size of the specimen the existing glass grating could not be used. A grating on a mylar film backing was cut to an appropriate size.

d. As the grating was flexible it required support. A plate of glass was obtained which had a very high level of flatness (approaching 0.01 mm / 0.0004 inch over 75 mm by 100 mm) and of a quality and thickness (1.2 mm) which would result in minimal loss of light intensity. A small hole was ground through the glass near one edge for attachment of a vacuum line. By sealing the edges of the grating with electrical tape and drawing a vacuum the grating attained the equivalent flatness of the glass.

e. Due to the recessed location of the specimen in the compression support fixture the existing supports for gratings could not be used. This posed an additional problem in that the grating must be parallel to the surface of the specimen. These problems were overcome by mounting a shim of precise thickness on the bottom of either end of the grating. The shim thickness was selected to provide the required clearance space between the grating and the top of the deformed surface when the shims were contacting the flat portion of surface of the specimen. By slightly modifying the compression support fixture

a method of supporting the grating was devised. This method applied a light clamping pressure to the grating, overtop of the shim at either end. It was thought that since the clamping pressure was light, and away from the damage area, it should not have any significant effect on the behaviour of the specimen under compressive load. On a few of the latter tests a thin film of lubricant was applied to the surface of the shims.

f. Because the illumination and viewing angles could not be precisely determined a method of calibration for the setup was required. Ideally, a calibration wedge or cone would be mounted on the specimen and left there throughout the duration of the test to provide a calibration reference for each photographic record. Due to a lack of clearance space this was not possible. The following technique was employed instead. A calibration disk was machined such that its two end faces were parallel. On one face a conical bevel was machined, at a very shallow angle of 1.5° , but a flat centre section was left (i.e. the bevelled face resembled a truncated cone). The disk was mounted on a glass backing identical to the grating, including spacer shims. Prior to commencing a test the calibration device was placed against the surface of the specimen, bridging the damage area. Then, the grating was placed on top of it and indexed against the flat section on the calibration disk. By recording the number of fringes, the overall distance they spanned and making use of the known angle of the calibration disk a fringe displacement value was determined.

4.6.3 Test Results - T800H/3900-2

The results of the first set of compression tests together with equivalent data from second set are shown in Table 4. Average and standard deviation values are given for the results for each impact energy level tested. Failure stress is based upon the cross-sectional area of the specimen prior to loading. A reasonable repeatability has been achieved.

	Specimen Number	Impact * Energy (joules)	Ultimate Stress (MPa)	Compressive Modulus (GPa)
	827-1a	11.4	412	53.8
	827-2a	11.4	415	52.4
Average			413	53.1
Std. Dev.			2.3	0.99
	827-3a	15.3	382	51.8
	827-4a	15.3	356	52.9
	668-1a	15.3	368	52.2
	666-1a	15.3	429	51.4
	664-1a	15.3	399	50.7
Average			387	51.8
Std. Dev.			28.7	0.81
	827-5a	22.9	318	52.4
	827-6a	22.9	340	N/A
Average			329	
Std. Dev.			15.0	
	663-2a	30.5	312	52.8
	664-2a	30.5	325	52.9
	666-2a	30.5	327	52.8
Average			322	52.8
Std. Dev.			8.4	0.09
	666-3a	40.7	272	53.0
	663-3a	40.7	275	53.2
	667-2a	40.7	283	53.3
Average			277	53.2
Std. Dev.			5.8	0.17
	666-4a	50.8	239	54.7
	667-4a	50.8	277	53.4
	664-4a	50.8	262	54.0
Average			259	54.0
Std. Dev.			19.4	0.66
	668-5a	61.6	219	53.8
	666-5a	61.6	219	54.4
	664-5a	61.6	210	54.6
Average			216	54.3
Std. Dev.			5.5	0.42

* Nominal impact energy shown

Table 4: Compression Test Results - T800H/3900-2

Plots of strength ratio versus impact energy and strength ratio versus damage area are presented in Figures 56 and 57. Strength ratio is defined as the ratio of the compressive strength of the damaged laminate to that of an undamaged laminate. The results indicate that an increase in either impact energy or damage area results in a reduction in compression-after-impact strength. Both plots show relatively linear behaviour throughout the range of test data.

Stress-strain data from the two specimens impacted at 11.44 J is shown in Figures 58 and 59. Strain values show good agreement with each other and linear response up to a stress of approximately 300 MPa. However, data from the vertical gauge on the back side of the damage (7), for both specimens, indicates a slower rate of strain increase. This response may have been due to the out-of-plane deformation of the damaged region, as a result of the initial eccentricity.

Beyond 300 MPa the strains diverge. Strain data for one half of the gauges show an increasing rate while data for the other half show a declining strain rate. Gauges which show a declining strain rate are located on one side at the middle of the specimen and on the other side at the top of the specimen. The gauges showing increasing strain rate are in the opposite locations. This response suggests global buckling of an end clamped specimen and was not expected since anti-buckling guides were used. The strain results indicate that one specimen buckled towards the impact side and the other to the back side.

A limited in-house buckling analysis using finite element methods was conducted with IAR to determine if buckling could occur. It was determined that if the laminate compressive and shear moduli were reduced by approximately 10 % from theoretical values and a damage area equivalent in size to the measured area was incorporated in the model a buckling load similar to the experimental value was obtained. The moduli of the damaged area were assumed to be 50 % of the undamaged moduli. Compressive moduli

Figure 56: Strength Ratio vs Impact Energy - T800H/3900-2

Figure 57: Strength Ratio vs Damage Area - T800H/3900-2

Figure 58: Compression Test Results

Figure 59: Compression Test Results

calculated from the results of the tests showed that experimental values for the laminate were in fact about 10 % lower than theoretical values. Repeating the analysis without the damage area gave a higher buckling load, approaching the value for inplane compressive failure. This study confirmed that the damage tends to reduce the global buckling load of the laminate.

Failure of the two specimens impacted at 11.44 J occurred approximately 30 % below the expected load. This result can be explained by the additional strain due to global buckling. It is likely that the damage area acted as the failure origin.

Strain histories for two specimens (827-3A and 827-4A), impacted at the next higher energy level of 15.25 J, are presented in Figures 60 and 62. Because Shadow Moiré techniques were used, strain gauges were located only on the impact side of the damage area.

Strain values for specimen 827-3A show a close agreement, with linear response, up to the point of divergence (approximately 330 MPa). The strain response after this point is similar to that of the two specimens impacted at 11.4 J (discussed above), indicating that global buckling occurred. Moiré images, presented in Figure 61, provide additional indication of global buckling. At loads close to failure fringes span the entire Moiré grating in a manner consistent with global buckling of the specimen.

Strain results and Moiré images indicate that sublaminar buckling was relatively minor. This result agrees with the predictions which give a buckling stress higher than the actual failure stress.

Strain data for the second specimen (827-4A) is presented in Figure 62. For this specimen, smaller strain gauges were used than for 827-3A (i.e. 2mm x 2mm as opposed to 5mm x 2mm), at the locations adjacent to the damage area. The smaller gauges would reduce strain averaging and would be more sensitive to local strains. The Moiré setup

Figure 60: Compression Test Results

0.0 MPa

115 MPa (12,000 lbf)

193 MPa (20,000 lbf)

273 MPa (28,300 lbf)

308 MPa (31,980 lbf)

327 MPa (34,000 lbf)

366 MPa (38,000 lbf)

377 MPa (39,200 lbf)

Figure 61: Moiré Images Specimen 827-3, 15.25 J

Figure 62 (1 of 2): Compression Test Results

Figure 62 (2 of 2): Compression Test Results

was also adjusted so as to achieve higher resolution (.006 mm/fringe as opposed to .01 mm/fringe for 827-3A).

Strain results show fair to good agreement with each other and indicate a linear response up to approximately 290 MPa (30,000 lbf). Beyond this value, strains on the impact and back side diverge. Strain data for gauge number three, located on the impact side of the damage, indicates that compressive strain increased more rapidly with load in this location compared to other locations. This suggests that out-of-plane deformation of the sublaminates was occurring, in a direction towards the back side. Moiré images, shown in Figure 63, confirm this result. Strain data for gauges two and five, which are adjacent to the damage and on the same side, indicate that beyond 290 MPa (30,000 lbf) global buckling occurred with deflection towards the impact side - in an opposite direction to that of the sublaminates. A result confirmed by the Moiré images as they become progressively more faint with increased load.

A distinct disturbance in strain response is evident at approximately 347 MPa (36,000 lbf). Moiré images show a crack indicating fibre fracture across the impact damage area. Failure of the specimen occurred shortly after formation of the crack.

Strain histories for specimens impacted at 15.25 J and having only far-field gauges are presented in Appendix C. Some scatter in failure strength is evident (See also Table 4). Strain data show fair to good agreement with each other and indicate a linear response up to a certain load value. This value varies considerably between the three specimens. Beyond this value strain data from impact and back sides diverges, indicating global buckling. The specimen having the lowest failure strength (668-1A) also had the earliest indication of strain divergence.

The strain histories for two specimens impacted at the next higher energy level of 22.9 J are shown in Figure 64 and 65. The results for specimen 827-5A, shown in Figure 64, indicate a generally linear response up to failure, with the exception of gauges nine

0.0 MPa

193 MPa (20,000 lbf)

231 MPa (24,000 lbf)

270 MPa (28,000 lbf)

308 MPa (32,000 lbf)

337 MPa (35,000 lbf)

347 MPa (36,000 lbf)

Fibre Fracture

348 MPa (36,100 lbf)

Figure 63: Moiré Images Specimen 827-4, 15.25 J

Figure 64: Compression Test Results

Figure 65: Compression Test Results

and ten. The results for these gauges, located on the impact side of the damage area, are consistent with an out-of-plane displacement of the damaged area. Strain data from gauges adjacent to the damage show no significant indication of load transfer or stress concentration as a result of buckling of the damaged area. However, it is possible that some averaging of strains may have occurred. The small jump in strain data at about 250 MPa (26,000 lbf) is due to a halt in specimen loading to allow removal of the Moiré grating. The strain results for gauge five were not consistent with that of other gauges. No explanation for this behaviour was determined. In conclusion, the effects of global buckling on final failure appear to have been negligible.

The second specimen (827-6A) had strain gauges on both the impact and back side of the damage area. Strain results are presented in Figure 65. Data from far field gauges (4 & 5) indicates a linear response up to specimen failure (340 MPa). Results from other vertical gauges indicate a linear response up to a stress of approximately 240 MPa. At this value, gauges on the impact and back side of the damage area indicate a distinct change in response. The results for the gauges on the back side of the damage (11 & 12) are consistent with sublaminar buckling - with buckling towards the back face. The results for the gauges on the impact side of the damage (9 & 10) initially indicate a partial unloading of stress but as compressive load increases further the response becomes consistent with sublaminar buckling towards the back face.

In conclusion, final failure for specimen 827-6A was not influenced by global buckling. Failure appears to have been a result of the instability of the sublaminae in the damaged area, and perhaps also as a result of stress concentrations due to fibre fracture and matrix cracks.

Strain data for specimens impacted at 30.5, 40.7, 50.8 and 61.6 J is limited to far-field strain and the results are presented in Appendix C. Strain data for all specimens impacted at 30.5 and 40.67 J show evidence of global buckling close to failure; however, it is more significant in some specimens than in others. Strain data for specimens

impacted at 51.44 and 62.55 J generally indicate a linear response up to failure but one or two specimens also show strain divergence indicative of global buckling.

In summary, the test results for T800H/3900-2 specimens having layup [45/0/-45/90]_{3s} indicated the following.

- Global stability effects became important at small damage sizes.
- Plies in the damaged region buckled towards the back side in the direction of the initial eccentricity, and appeared to buckle in unison (further verification is required however).
- Global buckling, if it occurred, tended to result in deflection towards the back side.
- No significant stress concentrations were observed next to the buckled area; however, damage sizes tested were relatively small and strain averaging may have occurred.
- The shape of the buckled region, as evidenced by the Moiré images, corresponded with that of buckling of major delaminations as opposed to buckling of individual delaminations.
- Out-of-plane displacement of the buckle was relatively minor for the two specimens monitored, in the order of 0.08 to 0.1 mm.

4.6.4 Test Results - AS4/3501-6

The results of compression testing on specimens fabricated from AS4/3501-6 are shown in Table 5. Too few specimens were tested to draw any conclusions concerning repeatability. Plots of strength ratio versus impact energy and strength ratio versus damage area are shown in Figures 66 and 67. The results show that residual strength decreased as impact energy was increased from 7.6 to 11.4 J but no further decrease resulted with increase in impact energy up to 15.25 J.

Specimen Number	Impact * Energy (joules)	Ultimate Stress (MPa)	Compressive Modulus (GPa)
826-3	7.6	207	49.2
826-4	7.6	220	51.6
826-1	11.4	157	N/A
826-2	11.4	157	51.9
826-5	15.3	169	52.3

* Nominal Impact Energy

Table 5: Test Results

Strain results for two specimens (826-4A and 826-3A) impacted at the lowest impact energy of 7.6 J are shown in Figures 68 and 70, respectively. Far field strain data (5 & 6) for specimen 826-4A indicates linear response up to failure, with no indication of global buckling. Strain data from gauges located on the impact side of the damage (2 & 3) indicate significant sublaminar buckling, towards the back face. Moiré images, presented in Figure 69, confirm that significant sublaminar buckling occurs. Each fringe corresponds to 0.17 mm (.007 inch) of out-of-plane deflection. Based on the strain results for gauges two and three (impact side) and the Moiré images (back side) it appears that the sublaminates buckled in unison.

Figure 66: Strength Ratio vs Impact Energy - AS4/3501-6

Figure 67: Strength Ratio vs Damage Area - AS4/3501-6

Figure 68: Compression Test Results

Figure 69: Moiré Images Specimen 826-4, 7.6 J

Moiré images indicate that delamination growth occurred. This growth was in a direction perpendicular to the load and was primarily towards one edge. Delamination growth increased progressively with load application in what appeared to be a stable manner. Loading was not halted to confirm this conclusion, however. Strain data indicated that final failure was accompanied by a sudden outward buckling along the entire midlength on both sides.

Moiré images indicate a shape and location for the initial buckle which is different than expected. Strain gauges, which are faintly visible, were located to either side of and centred with respect to the circular impact damage area. While fringes indicating a central circular deformation are evident, a much more prominent elliptical buckled region occurs at the top side of it. The elliptical buckle essentially spans two octants of the original damage area. Based on the previous damage characterization, it is known that delamination will occur at a depth of two plies from the surface for the right hand octant and could occur at a depth of one ply from the surface for the left hand octant. Because one edge of the buckle was bounded by a matrix crack some of the constraint on buckling was removed, allowing the plies in these quadrants to buckle before the buckling load of the entire damaged area was attained. Subsequent delamination growth at the back face emanates from the elliptical buckle rather than the central buckle.

Strain data for the second specimen, 826-3A, is presented in Figure 70. Far-field strain data (7,9 & 10) indicate linear response up to failure. The vertical gauge on the impact side of the damage (4) indicates sublaminar buckling towards the impact side. Moiré images, shown in Figure 71, indicate localized buckling towards the back side and delamination growth. Similar to specimen 826-4A, buckling of individual delaminations occurred prior to buckling of the entire damaged area. In this case, buckling took place in two areas, which were across from one another about the centre of impact. These areas are consistent with the locations of individual delaminations at one and two plies depth from the surface, as given by the damage characterization study. Final failure was due to similar causes as for specimen 826-4A.

Figure 70: Compression Test Results

0 MPa (0 lbf)

106 MPa (8,000 lbf)

159 MPa (12,000 lbf)

206 MPa (15,500 lbf)

Figure 71: Moiré Images
Specimen 826-3, 7.6 J

Stress-strain data for one of two specimens (826-2A) impacted at the next higher impact energy of 11.4 J is presented in Figure 72. Strain-time data is presented in Figure 73. For this specimen, strain gauges were mounted on both the back and impact side of the damage. Far-field strain data (7 & 8) indicates a linear response up to failure (Note that the waver in the strain data curve is due to an unintentional waver in load application). The results for the vertical gauges (4 & 11) indicate sublaminate buckling in a direction towards the impact face. The results from gauge eleven may also have been influenced by the fibre and matrix damage near the gauge. At 135 MPa strain data from gauge ten, which is on the back side adjacent to the damage area, shows a sudden change from compressive to tensile strain indicating a "snap" outward buckling, in this region. At the failure load (157.4 MPa), gauges on or adjacent to the back side damage, gauges eleven and thirteen respectively, show a similar response. This result is only evident on the Strain vs Time plot. Shortly thereafter, gauges further out (9 & 14) also register a similar response, which indicates that the buckled delamination grew outwards towards both edges of the specimen at failure.

Strain data for the one specimen impacted at 15.25 J (826-5A) is presented in Figure 74. Far-field strain data (4 & 5) indicates a linear response up to failure. Strain gauges on the damage area were located on the impact side only. These gauges (2 & 8) indicate sublaminate buckling, towards the back side, from first application of load. Delamination growth on the backside is evident early in the loading cycle as backside gauges adjacent to the impact damage area (6 & 7) indicate non-linear behaviour indicative of it. Gauge seven, in particular, indicates significant buckling suggesting that the delamination grew towards the side it was located on. Impact side gauges adjacent to the damage (1 & 3) show an increasing rate of strain increase, possibly due to load transfer from the buckled plies onto the undamaged area. At failure, strain gauges indicate outward buckling on both sides of the specimen (only evident on strain-time plot, not shown).

Figure 72: Compression Test Results

Figure 73: Compression Test Results

Figure 74: Compression Test Results

In summary, the test results for AS4/3501-6 specimens having layup [45/0/-45/90]_{3s} indicate the following:

- Individual delaminations close to the back surface may buckle prior to buckling of the sublaminae created by major delaminations.
- Delamination growth emanates from the buckled individual delaminations in a direction perpendicular to the load.
- No global buckling occurred in the specimens tested; however, for smaller damage sizes it may become important.
- Sublaminar buckling may occur either in a direction towards the impact side or in a direction towards the back side. Thus, the initial eccentricity does not appear to be as significant for this material system.

5.0 PREDICTIONS vs EXPERIMENTAL RESULTS

Predictions for buckling and failure strength ratios together with experimental failure strength ratios are presented in Tables 6 and 7, for specimens fabricated from T800H/3900-2 and AS4/3501-6 respectively. No experimental values for buckling are listed as no distinct buckling point was evident.

Predictions for the first set of tests (Xiong and Poon, 1993) are identified by 600 series identification numbers while the second set of tests are identified by 800 series numbers. TOF C-scans on which the second set were based were performed using an improved setup which gave more accurate results.

PC version 1.0 of the IAR prediction model was used to generate the second set of predictions. The point stress failure criterion was utilized. The characteristic dimension (d_0), for the T800H/3900-2 specimens, was determined using the formulation given by Xiong et al. (1993) and was in the order of $d_0 = 0.00035$. For AS4/3501-6 specimens, a characteristic dimension of $d_0 = 0.0$ was used, as previous work at IAR had shown that this value gave the best agreement between predictions and experimental results for this material system.

For T800H/3900-2 specimens impacted at 22.8 J and higher predictions generally show very good agreement with experimental results. Most predictions were within 10 % or better of the experimental value.

Predictions for T800H/3900-2 specimens impacted at 15.25 J show either very good or poor agreement with experimental results. Global buckling was evident at this energy level but the degree of buckling varied from specimen to specimen. At the lowest impact energy level of 11.4 J strain results clearly indicated significant global buckling and experimental strength values are correspondingly about 30 % below predictions.

Spec No.	Impact Energy (J)	Predicted Buckling Stress $=\sigma_B$ (GPa)	$M_t = (\sigma_B/\sigma_u)^*$	Predicted Strength Ratio (P)	Experim. Strength Ratio (E)	% Diff $=(P-E)/E*100$
827-1	11.4	-0.6559	1.0	1.0	0.7563	32.2
827-2	11.4	-0.5436	0.9988	0.9993	0.7624	31.1
827-3	15.25	-0.4554	0.8368	0.9027	0.7009	28.8
827-4	15.25	-0.3255	0.5981	0.7405	0.6543	13.2
664-1	15.25	-0.3255	0.5980	0.7295	0.7407	-1.5
666-1	15.25	-0.3669	0.6742	0.7863	0.7937	-0.9
668-1	15.25	-0.3449	0.6338	0.7459	0.6803	9.6
827-5	22.8	-0.2159	0.3968	0.6045	0.5852	3.3
827-6	22.8	-0.2477	0.4552	0.6427	0.6241	3.0
663-2	30.5	-0.1996	0.3668	0.5443	0.5814	-6.4
664-2	30.5	-0.2221	0.4082	0.5888	0.6061	-2.9
666-2	30.5	-0.2013	0.3698	0.5428	0.6061	-10.4
663-3	40.67	-0.1592	0.2925	0.5166	0.5102	1.3
666-3	40.67	-0.2051	0.3768	0.6198	0.5051	22.7
667-2	40.67	-0.1660	0.3049	0.5177	0.5236	-1.1
664-4	50.84	-0.1154	0.2121	0.4829	0.4854	-0.5
666-4	50.84	-0.1036	0.1903	0.4480	0.4444	0.8
667-4	50.84	-0.1170	0.2151	0.4613	0.5128	-8.6
664-5	61.55	-0.0927	0.1704	0.4398	0.3891	13.0
666-5	61.55	-0.0967	0.1776	0.4347	0.4065	6.9
668-5	61.55	-0.0788	0.1448	0.4062	0.4065	-0.1

* - σ_u refers to the undamaged failure stress

Table 6: Predictions vs Experimental - T800H/3900-2, [45/0/-45/90]₃₅

The close agreement between predictions and experimental values at the highest impact energies of 50.8 and 61.6 J is surprising as fractographic results indicated

extensive matrix and fibre fracture. The stiffness reduction or stress concentration resulting from this type of damage was not accounted for in the predictions, although the model does allow the user to define sublamine properties independent of the laminate properties. A possible explanation is that the predictions were based on buckling of the backside sublamine. The damage characterization study determined that at these energy levels the damage size increased, often quite significantly, towards the back side. If sublaminae at these energy levels buckle in unison (still to be confirmed) then using the dimensions of the backside sublamine would result in a very conservative estimate of the collective buckling stress. The conservative approach may have accounted for any reductions in buckling stress due to matrix and fibre damage.

Due to the small sample size for the second set of tests, no conclusions were drawn concerning the effect of the improved TOF C-scans on the agreement between predictions and experimental results.

Spec. No.	Impact Energy (J)	Predicted Buckling Stress $=\sigma_B$ (GPa)	$M_t = (\sigma_B/\sigma_u)^*$	Predicted Strength Ratio (P)	Experim. Strength Ratio (E)	% Diff $=(P-E)/E \times 100$
826-3	7.6	-0.0866	0.1462	0.4014	0.3485	15.2
826-4	7.6	-0.0944	0.1592	0.4163	0.3714	12.1
826-1	11.4	-0.0723	0.1220	0.3868	0.2644	46.3
826-2	11.4	-0.0772	0.1302	0.3934	0.2655	48.2
826-5	15.25	-0.0580	0.0978	0.3595	0.2851	26.1

* - σ_u refers to the undamaged failure stress

Table 7: Predictions vs Experimental - AS4/3501-6, [45/0/-45/90]_{3s}

Predictions of failure strength for AS4/3501-6 specimens have fair to poor agreement with experimental values. Factors which may have contributed to the lack of better agreement are that the effect of delamination growth was not accounted for and damage size may have been underestimated. As well, sample size was very small and the results may not represent the typical or average case.

Delamination growth is a failure mode that the IAR model does not account for. Based upon the Moiré images, delamination growth was relatively minor and thus could be considered as having only a small effect on the residual compressive strength. Further testing is recommended, however, to determine whether failure could have been precipitated, in part, by a sudden rapid growth of the buckled delamination. Testing could be done using projection Moiré and perhaps a high speed camera.

Predictions for T800H/3900-2 specimens were based on damage dimensions from TOF C-scan performed from the back side. However, for AS4/3501-6 specimens the fractographic investigation revealed that the backside TOF C-scans were only partly correct (See Section 4.4.2, pg 51). That is, the C-scan transducer was unable to sense the existing delamination at one and two plies depth and, instead, indicated the next delamination at five and six plies depth, respectively. These delaminations were shown to be smaller than those at one and two plies depth. Because no conventional C-scan existed for these specimens (due to C-scan equipment upgrade during the course of the second test set), damage dimensions were based on the delaminations at three and four plies depth, using the backside scan. This method provided an accurate measure of the minor diameter of the damage, based on a comparison against fractographic results, and a reasonable estimate of the major diameter (The impact side scan provided a further check). Damage size in future tests, particularly for this material system, should be confirmed by conventional C-scan.

Dost et al. (1989) performed a similar test program with a composite system, IM6/3501-6, which uses the same matrix. They found no indication of delamination

growth and good correlation with predictions, based upon a sublaminates stability model, over a large range of damage sizes. One explanation for the discrepancy between the present work and that by Dost et al. is that the composite system they used had a ply thickness 37 % greater than that of the current study. The greater bending stiffness resulting from this increased thickness translates into a reduction in the out-of-plane deflection of the buckled sublaminates at a given load, as compared to the deflection of a thinner sublaminates. As a consequence, significantly higher compressive loads are required, for the thicker sublaminates, to achieve out-of-plane stresses at the crack front of sufficient magnitude to cause delamination growth. Thus, specimen failure due to stress redistribution may occur prior to any significant delamination growth.

Table 8 shows a comparison between predictions of buckling and failure strength with and without accounting for the initial deformation (i.e. indent/bulge). For the two T800H/3900-2 specimens listed accounting for the initial deformation resulted in a drop in the prediction of buckling strength of about 6 to 8 % and a drop in the prediction of failure strength of about 3 to 4 %. Accounting for the initial deformation improved the agreement between predicted and experimental values.

	Predictions						Experimental
	Buckling Strength Ratio			Failure Strength Ratio (FSR)			FSR
	With Indent	Without Indent	% Difference	With Indent	Without Indent	% Difference	
827-4	0.3255	0.3468	6.5	0.7405	0.7664	3.5	0.6543
827-5	0.2159	0.2322	7.5	0.6045	0.6246	3.3	0.5852

Table 8: Predictions With and Without Accounting for the Initial Deformation, together with Experimental Values

Table 9 shows a comparison between predictions of buckling strength with and without accounting for variations in sublaminates layup. The first set of predictions is based on a stacking sequence corresponding to the layup of the laminate (i.e. [45/0/-

45/90]), and approximates the layup of the backside sublaminate. This layup was used to generate all of the predictions in Table 6 and 7, and will be referred to as the simplified assumption in the following discussion. The second set of predictions is based on the Boeing model which assumes all sublaminate have a stacking sequence corresponding to an interior sublaminate. Buckling strength is determined by averaging the buckling strength of the four variations in stacking sequence of an interior sublaminate. For the one example shown (Specimen 827-5), the Boeing method resulted in a buckling strength prediction 18.6 % higher and a failure strength prediction (averaged) 11.7 % higher than the predictions based on the simplified layup (Note: All values were generated using the IAR model). The failure strength prediction based on the simplified layup had good agreement with the experimental value. Using the Boeing assumption would have resulted in a significant difference in predicted and experimental failure strengths, yet the Boeing assumption seems logical. Further analysis concerning the effect of both the variation in stacking sequence of a sublaminate on buckling strength and the variation in stacking sequences between exterior, midplane and interior sublaminate on buckling strength may be worthwhile.

Sublaminate Layup	Buckling Stress (MPa)		Failure Stress (MPa)		
	Boeing Assumption	Simplified Assumption	Boeing Assumption	Simplified Assumption	Experimental
[45/0/-45/90]	216	216	328	328	
[0/-45/90/45]	248		350		
[-45/90/45/0]	308		391		
[90/45/0/-45]	252		353		
Average	256	216	356	328	318

Table 9: Effect of Stacking Sequence on Predictions

6.0 CONCLUSIONS

6.1 Conclusions

1. An experimental verification of the model developed by IAR for prediction of compression-after-impact strength has been achieved for the Toray T800H/3900-2 material system, based on coupons of layup $[45/0/-45/90]_{3s}$ and impact and compression testing as per Boeing Specification BSS 7260. Good agreement was generally observed between predicted and experimental failure strengths for impact energies from 22.9 to 61.5 J. At impact energies of 11.4 and 15.25 J results were obscured by the effects of global buckling, which was to be avoided by the use of the compression support fixture.

2. Based on experimental observation and comparison of predictions with experimental results, the following assumptions in the model appear to be correct for the T800H/3900-2 layups tested.

- Residual compressive strength appears to be governed by the stability of the damaged area.
- Stability of the damaged region is dependent on the sublaminates created by the major delaminations rather than the sublaminates created by individual delaminations.
- The initial eccentricity resulting from impact caused sublaminate out-of-plane deformation to occur early in the loading cycle and resulted in reduced compressive strength.
- The buckling load predicted by the model generally corresponded to a relatively significant out-of-plane deformation of the damaged area.

3. For the Hercules AS4/3501-6 material system the model somewhat overpredicted the residual compressive strength. Further investigation into the discrepancy is required. Experimental observations during compression testing indicated the following.

- Individual delaminations buckled prior to buckling of the sublaminates created by major delaminations.
- Buckling of individual delaminations led to delamination growth; however, growth was relatively minor up to the failure load.

4. Modelling strategies for the prediction of compression-after-impact strength will be dependent on layup. For example, an increase in laminate thickness may promote asymmetric damage and necessitate a buckling analysis based on several different sizes and thicknesses of sublaminates.

5. The specimen design given by BMS 8-276 for compression testing is not of sufficient thickness to prevent global buckling of specimens with small damage sizes. A change of thickness from 24 to 32 plies is recommended.

6. The staircase pattern of delaminations and matrix cracks described by Gosse and Hause (1987) was shown to be generally correct for the material systems tested.

7. The toughened composite system, T800H/3900-2, demonstrated significantly higher damage resistance as compared to the brittle system, AS4/3501-6.

6.2 Future Research

Extensions of the present work which appear worthwhile are as follows.

1. Repeat CAI testing for T800H/3900-2 specimens at impact energies ranging from 11.4 J to 40. J using a thicker layup, such as $[45/0/-45/90]_{4s}$, so as to eliminate or reduce the effects of global buckling. This layup modification is required to verify the model against small damage sizes.
2. Repeat CAI testing for AS4/3501-6 specimens basing the predictions on damage dimensions determined by reflected through-transmission C-scan.
3. Perform compression-after-impact testing of T800H/3900-2 specimens (standard layup) impacted at higher energies (i.e 30.5 J and higher) using Shadow Moiré and strain survey techniques to observe damage behaviour. Shadow Moiré could be performed from the impact side.
4. Poon (1993) suggested that Moiré methods could be used to quantify the out-of-plane deformation of the buckled sublaminates and thus verify and perhaps refine the shape function used in the analysis. An attempt was made to do this work; however, further refinement of the Moiré equipment and test techniques used in this study is required to accomplish this objective. The refinements should include the development of a support fixture for the Moiré grating which is independent of the specimen and the development of a more efficient means of calibration.
5. Perform compression testing of the other layup types proposed in the study.
6. Perform a detailed analysis of the effect of sublaminate layup on buckling strain (i.e. account for the effect of various sublaminate configurations and stacking sequences).

6.3 Summary of Contributions

1. Provided a limited experimental verification of a CAI strength prediction model developed by IAR.
2. Provided further confirmation of the circular staircase pattern of delaminations and matrix cracks, described by Gosse and Hause, for the Toray T800H/3900-2 and Hercules AS4/3501-6 composite systems with layup of $[45/0/-45/90]_{3S}$.
3. Produced an "atlas" of characteristic damage states for different layups of the T800H/3900-2 material system.
4. Produced a database of compression-after-impact strengths for T800H/3900-2 and AS4/3501-6 material systems, for a range of impact energies.

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APPENDIX A Cure Cycles

- 1/ Apply 22 inches Hg minimum vacuum to bag.
- 2/ Pressurize autoclave to 85 psig. Vent vacuum bag to atmosphere when pressure reaches 20 ± 10 psig. Keep vented until end of cure. Pressure under vacuum bag shall be 0 ± 5 psig throughout remainder of cure cycle.
- 3/ Apply heat after full pressurization of autoclave.
- 4/ Heat-up rate shall be 1 to 5 F/min.
- 5/ Hold for 120 ± 60 minutes at 355 ± 10 F and 85 ± 15 psig. Temperature is based on lagging thermocouple. Cure begins when last thermocouple reaches the minimum cure temperature. All thermocouples must remain in cure temperature range.
- 6/ Cool under pressure until the part temperature reaches 140 F or below. Natural pressure drop (19 psig max.) in autoclave due to cool down is allowed. Cool-down rate shall be 5 F/minute maximum.
- 7/ When part temperature reaches below 140 F, release pressure and remove part.

Figure A.1: Cure Cycle -Toray T800H/3900-2

1. PLACE VACUUM BAGGED LAYUP IN AUTOCLAVE AND CLOSE CLAVE.
2. APPLY MINIMUM VACUUM OF 25 IN OF Hg.
3. APPLY 85 ± 5 PSIG.
4. AT A RATE OF 3° TO 5° /MIN, RAISE THE LAMINATE TEMPERATURE TO $240^\circ\text{F} \pm 10^\circ\text{F}$, WHILE HOLDING $85 \pm$ PSIG AUTOCLAVE PRESSURE AND 20-29 INCHES OF Hg VACUUM.
5. HOLD AT $240 \pm 10^\circ\text{F}$, 85 ± 5 PSIG AND 20-29 IN. OF Hg FOR 60 TO 70 MINUTES.
6. RAISE PRESSURE TO 100 ± 5 PSIG AND VENT THE VACUUM BAG TO ATMOSPHERE PRESSURE.
7. RAISE TEMPERATURE AT A RATE OF 3 TO 5° /MIN TO $350 \pm 10^\circ\text{F}$. HOLD FOR 120 ± 10 MINUTES UNDER 100 ± 5 PSIG AUTOCLAVE PRESSURE.
8. AT A RATE OF $5^\circ \pm 1^\circ$ /MIN LOWER LAMINATE TEMPERATURE TO 200°F . RELEASE AUTOCLAVE PRESSURE.
9. REMOVE LAYUP FROM AUTOCLAVE AND UNBAG.

Figure A.2: Cure Cycle-Hercules AS4/3501-6

APPENDIX B - Indent Depths and Bump Height Values

T800H/3900-2			
Specimen	Impact Energy (J)	Indent Depth (mm)	Bump Height (mm)
827-1A	11.4	0.09	0.11
827-2A	11.4	0.11	0.10
827-3A	15.3	0.15	0.17
827-4A	15.3	0.18	0.13
667-3A	15.3	0.16	N/A
827-5A	22.8	0.18	0.20
827-6A	22.8	0.20	0.53
667-1A	30.5	0.20	N/A
664-3A	40.7	0.20	N/A
663-4A	50.8	0.89	N/A
663-6A	61.6	1.14	N/A

Table B.1: Indent Depth and Bump Height
- T800H/3900-2

NOTE: All depths and heights were measured using a mechanical depth gauge.

AS4/3501-6			
Specimen	Impact Energy (J)	Indent Depth (mm)	Bump Height (mm)
826-3	7.6	0.10	0.14
826-4	7.6	0.10	0.14
826-6	7.6	0.11	0.13
826-1	11.4	0.10	0.13
826-2	11.4	0.15	0.64
826-5	15.3	0.34	N/A

Table B.2: Indent Depths and Bump Heights

NOTE: All depths and heights were measured using a mechanical depth gauge.

APPENDIX C

Stress-Strain Plots - Test Set #1

Figure C.1: Compression Test Results

Figure C.2: Compression Test Results

Figure C.3: Compression Test Results

Figure C.4: Compression Test Results

Figure C.5: Compression Test Results

Figure C.6: Compression Test Results

Figure C.7: Compression Test Results

Figure C.8: Compression Test Results

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