Damage Resistance and Tolerance of 3D Woven Composites

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DAMAGE RESISTANCE AND TOLERANCE OF 3D WOVEN COMPOSITES

By

Justin T. McDermott

B.S. University of Maine, 2017

A THESIS

Submitted in Partial Fulfillment of the Requirements for the Degree of Master of Science (in Mechanical Engineering)

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Composite materials have been adopted into primary aircraft structures by virtue of their great strength-to-weight and stiffness-to-weight ratios, fatigue insensitivity, and corrosion resistance. These characteristics are leveraged by aircraft designers to deliver improved fuel efficiency and reduced scheduled maintenance burdens for their customers. These benefits have been impressively realized in the Boeing 787 and Airbus A350 XWB, with airframes utilizing about 50% composites by weight. TEMPERING these successes, however, are the inherent vulnerabilities of carbon-fiber reinforced composites. When compared to conventional metallic structure, composite laminates are more sensitive to stress concentrations at mechanical fastenings and damage due to low-velocity impact. In the case of low-velocity impact, the delamination failure mode presents a unique potential for critical strength and stiffness reduction with little visible indication that damage has occurred. This “Barely Visible Impact Damage” (BVID) is a critical design case and is typically addressed in the aircraft design process by reducing material allowables to account for undetected damage, as well as defining specific maintenance and inspection plans to be carried out by operators during service. The next generation of composite materials are being developed which effectively eliminate the delamination failure mode using
three-dimensional pre-formed fiber architectures. So-called 3D woven composites offer enhanced through-the-thickness performance, and exhibit improved damage containment when exposed to out-of-plane impact loading. This study follows on to work by Warren [1–4] and London [5], who characterized the response of similar 3D woven composites to other critical design cases such as open hole compression, single and double shear bearing, and fatigue of bolted connections. This study adds critical information to the knowledgebase on 3D woven composites by comparing the impact damage resistance and tolerance performance of a 3D woven composite with that of an industry-typical multi-directional laminate of comparable thickness, stiffness, and strength. Both materials were first characterized using standard ASTM tension, compression, V-notched rail shear, fiber-volume fraction by acid digestion, and mode-1 fracture toughness tests. Damage resistance and tolerance were evaluated using standard drop-weight impact and quasi-static compression-after-impact tests under a range of BVID-type conditions from 5 to 130 Joules. The AITM 1-0010 test standard was used with minor adaptations. Post-impact non-destructive inspection (NDI) methods included damage area measurement using ultrasonic C-scan, damage characterization using micro-computed tomography (μCT), and manual indentation measurements for comparison with industry data.
DEDICATION

— To Emma —
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CHAPTER 1
INTRODUCTION

1.1 Background

Carbon fiber reinforced polymers (CFRP) have become prevalent in aircraft structures over traditional aerospace materials like aluminum, titanium, and steel because of their superior strength-to-weight and stiffness-to-weight ratios, in addition to fatigue and corrosion insensitivity. These benefits have been realized in industry with many applications over a lengthy integration process. CFRP structural components were initially introduced in the mid 1970’s with a limited run of Boeing 737 flight spoilers, followed shortly by empennage components on various Boeing, Douglas, and Lockheed aircraft as part of the NASA Aircraft Energy Efficiency (ACEE) program [16]. Various Airbus models have utilized CFRP primary structures from 1985 onward [17], and the current state of the art in composite airframe design results in about 50% of airframe mass attributable to CFRP composite materials [18, 19].

Along with the clear-cut advantages, CFRP also have crucial differences in behavior when compared with metallic materials. CFRP exhibit more linear strain to failure, greater static notch sensitivity, weaker transverse properties, higher variability of mechanical properties, and higher sensitivity to hygrothermal environment. In-plane delamination, as opposed to through-thickness crack propagation, has been the primary damage growth mechanism [20]. These significant differences in behaviour compared to the traditional materials have necessitated entirely new certification methodology, especially with regards to design for damage tolerance and maintenance planning. Component-level tests and in-service findings from the numerous examples of composite structures already discussed have performed excellently, particularly in regard to environmental deterioration and fatigue damage [21-25], leaving accidental damage
as the primary consideration for damage tolerance design and maintenance planning on thick-gauge composite primary structure [16]. Recent industry workgroup presentations express that the primary damage threats to aircraft are related to ground handling, which are reported at a much higher rate than operation-induced impact damage such as runway stones, hail, or bird strikes [17].

Conventional laminates present an inherent vulnerability to out of plane loading such as accidental impact because of their susceptibility to delamination. In the case of low-velocity impact, this failure mode presents the potential for critical strength and stiffness reduction without commensurate visual indication of damage [16, 26]. Such damage is frequently referred to as barely visible impact damage (BVID), and can become a critical parameter to address when defining material allowable strains.

A new class of CFRP materials have been introduced called 3D woven composites which include through-thickness reinforcement. Using three-dimensionally woven pre-formed fiber architectures, this class of CFRP offers many benefits over laminates such as reduced manufacturing cost and the production of near-net-shape preforms. Most relevantly, they offer enhanced through-thickness performance and can control or eliminate the delamination failure mode [1, 5, 27]. 3D woven composites have already found applications in aircraft structures for applications where improved damage tolerance is necessary. The CFM International “Leading Edge Aviation Propulsion” (LEAP) high-bypass turbofan engines utilize 3D woven CFRP fan blades and fan casings. The Boeing 787 utilizes a 3D woven main landing gear brace of approximately 2 meters length.

As aircraft OEMs ramp up focus on reducing operating costs for their customers, fuel burn and maintenance costs are the primary targets. Both areas can be significantly influenced by airframe design [18]. Light-weighting for fuel efficiency, lengthening inspection intervals to reduce scheduled maintenance burden, and using more durable components to reduce unscheduled maintenance burdens are pertinent interests
in airframe design, however such measures must be defended with sound engineering judgment to ensure safety during service.

Regulatory agencies such as the United States Federal Aviation Administration (FAA) have written regulations with the ultimate goal of preserving aviation safety by regulating damage tolerance requirements. CFR Title 14 Part 25 contains a variety of regulations pertaining to transport category aircraft, providing standards to which manufacturers must adhere. 14-CFR 25.571 “Damage - tolerance and fatigue evaluation of structure” (a) is central to this subject and provides a succinct summary of the goal of these regulations:

> “An evaluation of the strength, detail design, and fabrication must show that catastrophic failure due to fatigue, corrosion, manufacturing defects, or accidental damage, will be avoided throughout the operational life of the airplane.”

Naturally, this regulation is equally applicable to metallic and composite structures, and many approaches to providing catastrophe-free air transport are possible within the bounds of the regulations. In practice, the type certification process is a cooperative and interactive process between original equipment manufacturers (OEMs) and the FAA. FAA documents AC 25-571-1D [28] and AC 20-107B [7] provide guidance and suitable technical approaches for complying with the regulation for metallic and composite structures, respectively. The European Union Aviation Safety Agency (EASA) has published nearly identical documents with minor grammatical modifications [29]. The Composite Materials Handbook 17 (CMH-17) volume 3 [16] is a much more thorough reference for damage tolerance design methodology.

Structural damage tolerance improvements like the adoption of more damage tolerant materials have potential to improve maintenance economy by increasing
allowable damage limits (ADLs) and lengthening inspection intervals (routine maintenance), which is discussed in more depth in Chapter 4. Damage resistance improvements have the potential to improve maintenance economy by decreasing the frequency of non-routine maintenance i.e. structural component repair or replacement, as a result of lower probability of detectable damage. Non-routine maintenance frequently doubles or even triples the total labor hours expended during a maintenance check [30]. Damage resistance evaluation is discussed in Chapter 3.

1.2 Purpose of research

This thesis provides experimental knowledgebase to address the following broad research questions regarding the superior impact damage resistance and tolerance of 3D woven composites when compared to 2D laminated composites:

1. What effect does the increased damage resistance of 3D woven composites have on the life cycle cost of an aircraft structure by reducing the frequency of component repair as a result of a lower probability of detectable damage occurrence?

2. Can the increased damage tolerance in 3D woven composites reduce life cycle cost by lengthening the inspection interval?

The experimental work compares the damage resistance and damage tolerance performance of a 3D woven composite to that of a conventional multi-directional laminate of comparable thickness, stiffness, and strength. In-plane mechanical properties were also evaluated. Material details can be found in section 1.3.

Damage resistance was evaluated using standard low-velocity drop-weight impact testing under BVID-type conditions. Resulting damage was assessed using energy absorption analysis, damage area measurement with ultrasonic through-transmission C-scan imaging, failure mode analysis using micro-computed tomography (µCT), and simple dent depth measurements. Comparisons were made between the responses of the two material systems by presenting the energy absorption, dent depth and C-scan
damage-area metrics as functions of incident impact energy. This approach yielded a comprehensive understanding of the damage containment properties of the 3D woven composite. Additional insights into damage mechanisms were obtained using µCT scan technology to identify and analyze the resulting damage patterns.

Damage tolerance was evaluated using standard quasi-static compression-after-impact (CAI) tests at three discreet damage levels which have been determined to correspond well with in-service impact threats to aircraft structures.

1.3 Description of materials evaluated

Both materials evaluated were manufactured by Albany Engineered Composites, Inc. (AEC) using the resin transfer molding technique. Constituent materials for both composites were Hexcel® IM7 carbon fibers and Cycom® PR520 toughened epoxy matrix. The 3D woven composite was a ply-to-ply architecture with a warp tow spacing of 2.54 mm. The design warp/weft reinforcement ratio — 60% warp, 40% weft — resulted in a weft/pick tow spacing of 4.14 mm and nominal thickness of 4.42 mm. 24k tows were used throughout. No straight “stuffer” tows were included in this architecture. All warp and weft tows also act as through-thickness binders. The unit cell, which is the smallest volume that can represent the entire periodic weave architecture, measured 15.2 mm along the weft direction × 24.8 mm along the warp direction. A schematic illustrating the weave architecture can be seen in Figure 1.1. Warp tows are shown in blue and weft tows are shown in red.

The 2D baseline material was a 24-layer multidirectional symmetric non-crimp fabric (NCF) laminate. The laminate schedule [-45/+45/0/90/0/+45/-45/0/0/0/90/+45]s was chosen by AEC as a typical layup (44/44/11) used by aerospace OEMs for aircraft applications. The approximate reinforcement distribution was 41.6% in the 0°, 41.6% in the ±45° and 16.6% in the 90° directions. Similar reinforcement distributions have been used in fiber-dominated layups for fighter aircraft [11, 31]. Nominal thickness
was 4.42 mm. Individual NCF laminae were comprised of unidirectional 12k carbon
tows weighing 194 g/m². A polyester stitch thread was used running perpendicular
to the tows with a spacing of 2 mm and a total weight of 9.15 g/m².

1.4 Damage tolerance methodology

This section will review industry and regulatory treatment of damage resistance
and damage tolerance to explain the selection of experimental parameters and to
provide a framework for interpreting results.

Damage detectability is a key parameter in commercial aircraft damage tolerance
methodology since it is directly related to post-impact strength requirements. Figure
1.2 appears in the CMH-17, the FAA AC20-107B, the European equivalent (AMC
20-29), and in many industry work-group presentations. It summarizes the post-impact
strength requirements as well as the categories used to classify damage, which are
closely related to detectability. Although the magnitude of the strength requirements
vary based on application, the chart provides a useful visualization based on "Limit"
and "Ultimate" load levels which are applicable to all primary structure. Limit loads
refer to a set of criteria given in 14-CFR 25.571 (b), which describe the conditions an
aircraft must be capable of withstanding, dependent on flight regime. For example,
in the situation where a transport-category aircraft is being maneuvered into a dive,
the minimum limit symmetric maneuvering load factor which the FAA will allow is -1.0, i.e., all transport category aircraft must be safely capable of withstanding a force applied downward on the wings, at least equal to the weight of the aircraft, unless special arrangements are made. This requirement is useful for defining limit loads on airframe components, for instance, the necessary compressive capability of the bottom wing skin. For landing gear and related structure, this -1.0 load factor requirement is likely not critical, and an analogous requirement (14-CFR 25.473) describing the condition during a hard landing would be used to define limit loads instead. For pressure bulkheads, other worst-case situations are defined. As an aggregate, loads defined by these "worst case scenarios" are referred to as Limit Loads (LL) or Design Limit Loads (DLL), and should normally be avoided in aviation operations. "Ultimate load" (UL) or "Design Ultimate Load" (DUL) simply refers to a safety factor 1.5× limit load.

In general, the goal is to operate commercial aircraft with DUL capability. In the case of accidental damage, this safety factor is allowed to be temporarily eroded, for intervals and to levels commensurate with the detectability of the damage. In the case of very obvious damage like that caused by bird-strike or uncontained engine failure, the affected structure needs only to maintain “get-home” loads (less than limit loads.) On the other hand, undetectable damage must maintain DUL capability for the lifetime of the structure. Damage categories 1-5 describe the extent to which residual strength is allowed to degrade depending on damage detectability. Details on damage categories 1-5 are given in section 1.4.1.
As can be surmised from Figure 1.2, Allowable Damage Limit (ADL) and Critical Damage Threshold (CDT) are the damage severities for which strength requirements change. Damages up to and including the ADL are required to maintain DUL capability. Damage between the ADL and CDT must support greater than limit loads, and so-called "critical damage" must support limit loads. The ADL and CDT are not set by the regulator, rather they are location-dependent criteria set by the OEM early in the design process based on material-level durability and damage tolerance characteristics determined at the lower levels (coupon or element level) of the building-block test pyramid (Figure 1.3.) Later in the design process, ADLs are substantiated with full-scale component or sub-component tests which validate analytical strain or displacement predictions with the inclusion of critical allowable damage.
Although indentation depth is not a reliable predictor of residual strength after impact [9, 32], it is the typically-used metric to describe impact damage severity, since it correlates well with probability of detection during visual inspections. In general, OEMs are at liberty to define any inspection procedures that can be demonstrated effective. For instance, in structure that would be inspected using ultrasonic C-scan, the ADL might be set in terms of damage area. For non-impact type damage (e.g. lighting strike) ADL may be defined using other metrics. Industry practice demonstrates that visual inspections are the preferred method. Thus, in practice, the ADL typically describes the maximum impact dent depth, length, and width, that may be cosmetically, as opposed to structurally, repaired. The most common philosophy used for composite fatigue design is the no-growth approach, coupled with the load enhancement factor (LEF) approach introduced by Northrop report NADC-87042-60 [20]. To substantiate the ADL, designers must demonstrate that structure containing damage up to the ADL will sustain ultimate flight loads (1.5\times\text{limit load}) after being subjected to 1-3 service lifetimes [10, 33] of representative fatigue testing with LEF. LEF are statistically-defined enhanced loads used to achieve reliability given the high fatigue life scatter in composites.
1.4.1 Damage categories

The damage categories 1-5 organize the maintenance treatment that must be afforded to damages of different severity. The CMH-17 [16] provides good, comprehensive definitions, which are repeated here. These are consistent with those found in AC20-107B, but include some additional helpful dialogue. AC20-107B includes a statement that other categories of damage besides these described may be used by applicants (for type certification) in agreement with the regulatory authorities, should they help outline a specific path to fatigue and damage tolerance substantiation. This statement points to the cooperative nature of the damage tolerance certification process and relieves the necessity of aligning examples found in industry presentations to the precise definitions given here.
Category 1

Category 1 includes damage that may go undetected by the scheduled or directed field inspection methods, as well as manufacturing anomalies below the rejectable level as defined in the controlling manufacturing specification. Examples include barely-visible impact damage, minor environmental degradation, scratches, and gouges, as well as allowed disbonds and porosity. Structure with these types of damage or defect must retain Ultimate Load capability through the life of the aircraft. Substantiation must therefore address structure with these damage and defect types at critical locations. Detectability thresholds must be established (see Volume 3, Section 12.4.6). A reliable service life must be demonstrated along with retention of Ultimate Load capability throughout this life. This sets limits on the number of cycles and/or time the structure can remain in service without further substantiation. Critical environments must also be addressed.

Category 2

This category includes damages that can be reliably detected by the defined inspection program (i.e., inspection methods and intervals). Detailed inspections that are required by the inspection program when damage is found by a scheduled technique are also considered to be part of the inspection program. Typical damages in this category include visible impact damage (ranging from small to large), deep gouges or scratches, initially undetected anomalies associated with manufacturing process breakdowns, detectable delamination or disbonding, and major local heat or environmental degradation. Structure with this type of damage must retain Limit Load capability until the damage is found and repaired. Time
frames for reliable detection of such damages must be determined and demonstrated, and retention of Limit Load capability during these time frames must also be demonstrated.

**Category 3**

Category 3 damages are those that can be reliably detected within a few flights of the occurrence of damage by operations or ramp maintenance personnel with no special skills in composite inspection. They may be initially detected either visually in a pre-flight walk-around inspection, or due to a loss of form, fit, or function. In either case, additional inspections are warranted to identify the full extent of damage to the part and surrounding structure. Examples of Category 3 damage include large visible impact damage, and damages that create other obvious signals (e.g., fuel leaks, system malfunctions, failure to pressurize, or cabin noise). Structure with this type of damage must retain a specified load capability until the damage is found and repaired. This load level depends on the time needed to reliably detect the damage, and is therefore dependent on the damage detectability and location. The specific load requirements should be negotiated with the regulatory authority, and are generally less than or equal to Limit Load. The reliable and rapid detection of such damages must be demonstrated.

The primary difference between Category 2 and 3 damages is the maximum time to detection; Category 2 damage may exist for one or more inspection intervals (typically thousands of flights), while Category 3 damage will only exist for a few flights. During structural substantiation, therefore, a much larger number of repeated load cycles must be applied to structure with Category 2 damage prior to demonstrating static residual strength capability than for structure with Category 3 damage.
Category 4

This category includes discrete source damage that is known to the flight crew, allowing flight maneuvers to be limited until landing. Examples include damage caused by rotor burst, bird strike, severe lightning, landing gear tire burst, and severe in-flight hail. Structure with this level of damage must retain “Get Home” capability for the duration of the flight. Since the structure will be repaired prior to additional service, repeated load demonstrations are limited to completion of that single flight. The specific events and static loading levels that must be demonstrated are defined in the regulations and the associated guidance material, respectively, but are generally lower than Limit Load levels (e.g., 70% of Limit for flight maneuver loads).

Category 5

Category 5 encompasses severe damage that is created by anomalous ground or flight events that were not anticipated, and therefore not considered during the aircraft’s design. Structure with this type of damage must not fly until the damage is assessed and repaired, as appropriate. Typical examples of events causing Category 5 damage are severe service vehicle collisions, anomalous flight overload conditions, abnormally hard landings, and loss of aircraft parts in flight, including possible subsequent high energy blunt impact with adjacent structure. These scenarios are not substantiated during the certification process.

Special considerations are therefore required to ensure that such damages do not threaten the safety of the aircraft. Procedures and training must be defined to ensure that any anomalous events are reported by operations personnel for immediate assessment. Directed inspections are required
to determine the extent of damage. These should not be limited to the area immediately adjacent to the actual event location (e.g., impact site), since damage may be caused in remote areas (e.g., load reaction points). Engineering personnel familiar with the load paths and response of the structure must therefore be involved in the assessments. Substantiation of structural capability may be required to address either unrepaired damage or repairs that are beyond those covered during the certification process. Additional discussions of treatment of Category 5 damage are contained in Volume 3, Section 12.3.3.

Like the ADL and CDT, damage categories for impact are given definitions based on measurable criteria. A primary outcome for this research was to understand the impact energy required, for each material, to create BVID (Category 1), VID (Category 2) and obvious damage (Category 3). Some research effort was dedicated to obtaining these industry damage criteria, as well as the range of impact energies which would be expected in operation. This research is presented here.

Wichita State University NIAR holds periodic workshops on composite damage tolerance, fatigue, and maintenance. The workshops have provided a platform for representatives from airlines, manufacturers, certifying agencies, and academia to present their perspectives on damage tolerance and maintenance methodology. The workshops also serve to provide pertinent updates for the CMH-17.

The 2006 workshop on Composite Damage Tolerance & Maintenance was held on July 19-21 in Chicago, IL. Presentations given by Allen Fawcett and Gary Oakes from Boeing, and Chantal Fualdes and Roland Thevenin from Airbus addressed topics including damage detectability criteria, impact threats, and structural substantiation methods.

Fualdes presented Figure 1.4 which summarizes load carrying requirements over the domain of detectability and impact energy, and highlights the thresholds that
need to be set by the manufacturer for a given application; namely the damage metric, BVID definition, Large VID definition, and impact threat.

Figure 1.4: Airbus depiction of damage tolerance domain [8]

Fualdes [8] explains that the BVID definition is the dent depth corresponding to 90% probability of detection (POD) with 95% confidence. BVID definition is also dependent on visual inspection type. There are two visual inspection types in common use for both Airbus and Boeing aircraft. General Visual Inspection (GVI) and Detailed Visual Inspection (DVI or DET).

Airbus selected their visibility thresholds based on probability-of-detection experiments that mimicked the respective inspection conditions. The GVI experiment consisted of a total of 240 inspections. The DET experiment consisted of a total of 902 inspections. Inspection parameters are compared in Table 1.1
Table 1.1: Inspection parameters from Airbus probability of detection experiments

<table>
<thead>
<tr>
<th>Inspection Type</th>
<th>GVI</th>
<th>DVI</th>
</tr>
</thead>
<tbody>
<tr>
<td>Inspection Distance (cm)</td>
<td>100</td>
<td>50</td>
</tr>
<tr>
<td>Panel Size ($m^2$)</td>
<td>9.6</td>
<td>multiple, 0.01 - 0.8</td>
</tr>
<tr>
<td>Lighting Condition</td>
<td>ambient</td>
<td>ambient &amp; grazing</td>
</tr>
<tr>
<td>Impactor Diameter (mm)</td>
<td>multiple, 6-120</td>
<td>6 &amp; 16</td>
</tr>
<tr>
<td>Duration (s)</td>
<td>30</td>
<td>unlimited</td>
</tr>
<tr>
<td>Paint</td>
<td>white</td>
<td>varied</td>
</tr>
</tbody>
</table>

The results were statistically processed using a search for maximum plausibility type approach, yielding a log-normal cumulative distribution function which could be probed for the dent depth corresponding to 90% POD.

To validate the obtained detectability threshold for GVI inspection, Airbus compared with findings from an in-service survey of about 1000 damage records (500,000 flight hours, 73 aircraft over a 3-year interval) from a European airline. It was found that 85% of damages detected during GVI were below the Airbus detectability threshold, indicating the Airbus BVID threshold is conservative. Fualdes was reluctant to share the threshold values in this presentation, so the x-axis values are redacted (Figure 1.5a.) In September 2015 at the Composite Transport Workshop on Damage Tolerance and Maintenance in Dorval, Fualdes [9] presented Figure 1.5a once more, this time including the axis values. From that plot the BVID threshold used by Airbus for GVI-type inspections was read as 1.3 mm. The results of the DVI inspection experiment are also presented (Figure 1.5b.) The dent depth corresponding to 90% POD can be read here as 0.3 mm.
Fawcett and Oakes \cite{10} presented a consistent definition for BVID with respect to the general visual inspection:

“Small damages which may not be found during heavy maintenance general visual inspections using typical lighting conditions from a distance of 5 feet –typical dent depth ~0.01 to 0.02 inches (OML) – Dent depth relaxation must be accounted for.”

Explicit details regarding the probability of detection studies are not presented, but typical dent depths for the BVID category are given as 0.01 - 0.02 in (0.25 - 0.5 mm)

Large, or “obvious” visible impact damage is associated to walk-around inspection, and will remain in service for no more than a few flights. This threshold is significant because it defines where load carrying capacity drops to limit load. To define this threshold, Fualdes’ presentation states simply that Category 3 damage is considered on a case-by-case basis. Typically, the damage involves penetration. Example LVID for a sandwich structure are shown in Figure 1.6

Fawcett and Oakes presented some samples of large visible damage (Figures 1.7 and 1.8) along with their required load capacity (Limit load) and durations (one missed inspection interval). By the AC20-107B definition, Category 3 damage is

![Diagram](image.png)

(a) Airbus GVI threshold & Airline findings  
(b) Airbus DVI experiment results

Figure 1.5: Airbus visibility thresholds \cite{8, 9}
required to carry limit load, but is not associated with a specific inspection interval. Because of the mention of missed inspection interval, it is unclear whether Boeing classifies these examples as Category 2, 3, or if other arrangements have been made. Regardless, the presented damage likely represents the AC-20 107B Category 3 damage.

Figure 1.6: Example large (Cat 3) damage from Airbus presentation [8]

Referring to Figure 1.4, two more pertinent criteria must be established to describe the relevant domain for the damage resistance and tolerance experiments: ‘Realistic’ impact energy threat ($p = 10^{-5}$ /flight hour) and ‘Remote’ impact energy threat ($p = 10^{-9}$ /flight hour.)

Airbus impact damage threat assessments are based on an extensive collection of in-service data, and validated by bibliographic studies DTO/FAA/AR-96/111 and AR-95/17 [11, 34]. The in-service data originate from four surveys. One 18,740,000 flight hour survey includes all Airbus types and focuses on impact damage to wings, one 1,140,000 flight hour survey covers the A320 family and focuses on impact damage to the fuselage. A similar 500,000 flight hour survey covers the A320 family and presents impact damage data over the whole aircraft. An additional survey tallied 10,330,000 flight hours, but no specification was presented. [8]

The realistic and remote damage threat levels were defined based on aircraft zone. In service data showed that the most dominant impact threat was ground handling.
incidents, occurring at the belt line near passenger doors, below the beltline near luggage bay doors, and on the upper root of the horizontal tail plane. These zones are flagged as “high threat” or “medium threat” zones and carry a higher risk of high energy strikes. Areas outside these zones are classified as “Typical”. A summary of the energy level cutoffs for selected zones is given in Table 1.2.

“Advanced Certification Methodology for Composite Structures” (FAA report FAA/AR-96-111) [11] details the development of a certification methodology that permits certification of bonded and co-cured composite structures, as well as addressing the threat of low-velocity impact.
Based on bibliographic studies, Kan et al. [11] developed a set of three impact threat distributions modeled on a two-parameter Weibull model in terms of impact energy. These were called “High” “Medium” and “Low.” The distributions are shown in Figure 1.9.
Table 1.2: Airbus impact threat parameters from in-service surveys

<table>
<thead>
<tr>
<th>Zone</th>
<th>Realistic threat (J)</th>
<th>Remote threat (J)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Typical</td>
<td>35</td>
<td>90</td>
</tr>
<tr>
<td>HTP root/rear fuselage skin</td>
<td>140</td>
<td>–</td>
</tr>
<tr>
<td>Doorway zones</td>
<td>132.5</td>
<td>238.5</td>
</tr>
<tr>
<td>Wing [9]</td>
<td>60</td>
<td>–</td>
</tr>
</tbody>
</table>

Figure 1.9: Impact threat distributions from Northrop [11]

The distributions were checked against additional in-service survey data taken from fighter aircraft. Under a Northrop/MCAir collaborative research program, MCAir conducted a field survey of low-velocity impact damage to quantify impact threat to composite aircraft structures. In this survey, 1644 damages were considered from a collection of four in-service aircraft types (F-4, F-111, A-10, and F-18). The data was collected in the form of dent depth and presented as an exceedance curve.
In order to apply this information to composite structures, an additional study was needed which correlated metallic dent depth to incident energy. This testing was carried out by the same partners on an F-15 wing skin. This correlation was used to transform the data from the field survey into an energy level exceedance curve.

The energy-based exceedance data was converted into a probability distribution and compared with the three threats described above. The resulting fit used modal energy $X_m = 1$ ft-lb and $X_p (p =.0005) = 35$ ft-lb. The fitted distribution compared with the three model threats is shown in Figure 1.10.

![Figure 1.10: Fighter survey threat comparison with bibliographic threat distributions - Northrop [11]](image)

Finally, Kan discusses the BVID criteria established in [35]. This criterion defines visible impact damage as damage with 0.05 inch or deeper dent for thin laminates and damage produced by 100 ft-lb (136 J) impact for thick laminates. Kan states that this criterion is more consistent with in-service findings, and that despite the criteria being generated for F-18 wing skins based on navy test data, comparison with the data generated by Northrop and McAir’s field survey indicates that the criterion is applicable for other composite materials as well.
Fawcett and Oakes [10] presented a table of impact threat criteria, using different terminology (Figure 1.11) It is likely that the Boeing term “General acreage” corresponds roughly with the Airbus “Typical Zones”, and that “Repeat Impact Threat areas” correspond with Airbus “High threat zones.” Aircraft from the two companies are operated side by side and face similar damage threats. Although these terms are likely interchangeable, the Boeing presentation does not seem to address the “Remote threat” parameter. This would be the energy level at which BVID would be expected only to support limit load capability. Based on the requirements indicated in Figure 1.11, Boeing requires ultimate load capability be carried out to energy levels that Airbus and other sources would call “extremely improbable.”

<table>
<thead>
<tr>
<th>Threat</th>
<th>Criteria</th>
<th>Requirement</th>
<th>Notes</th>
</tr>
</thead>
<tbody>
<tr>
<td>Small Tool Drop</td>
<td>48 in-lbs normal to surface</td>
<td>No visible damage</td>
<td>1” diameter-hemispherical impacter</td>
</tr>
<tr>
<td></td>
<td></td>
<td>No non-visible damage growth for 3 DSOs</td>
<td></td>
</tr>
<tr>
<td></td>
<td></td>
<td>Accounted for in Ultimate Design Allowables</td>
<td></td>
</tr>
<tr>
<td>Large Tool Drop</td>
<td>1200 in-lbs or a defined dent depth cut-off</td>
<td>Barely visible damage which may not be found during HMV</td>
<td>1” diameter-hemispherical impacter</td>
</tr>
<tr>
<td>(BVID general acreage</td>
<td>(considering relaxation) based on level</td>
<td>No damage growth for 3 DSOs with LEF</td>
<td></td>
</tr>
<tr>
<td>(FAR 29.309, AC-20-107A)</td>
<td>of visibility as related to the inspection</td>
<td>Capable of Ultimate strength</td>
<td></td>
</tr>
<tr>
<td></td>
<td>method.</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Large Tool Drop</td>
<td>1200 in-lbs or a defined dent depth cut-off</td>
<td>Barely visible damage which may not be found during HMV</td>
<td>1” diameter-hemispherical impacter</td>
</tr>
<tr>
<td>(BVID) repeat impact</td>
<td>(considering relaxation) based on level</td>
<td>No damage growth for 3 DSOs with LEF</td>
<td></td>
</tr>
<tr>
<td>threat areas (FAR</td>
<td>of visibility as related to the inspection</td>
<td>Capable of Ultimate strength</td>
<td></td>
</tr>
<tr>
<td>29.309, AC-20-107A)</td>
<td>method.</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Visible Impact Damage</td>
<td>No energy cut-off</td>
<td>Visible Damage with a high probability to be found during HMV</td>
<td>1” to 6” diameter-hemispherical impacter</td>
</tr>
<tr>
<td>(VID) (Damage Tolerance</td>
<td></td>
<td>No damage growth for 2 times the planned inspection interval with LEF</td>
<td></td>
</tr>
<tr>
<td>FAR 25-571d)</td>
<td></td>
<td>Capable of residual Limit strength</td>
<td></td>
</tr>
</tbody>
</table>

Figure 1.11: Boeing impact threats as presented in Fawcett, [10]
1.5 Conclusions and recommendations

This chapter has reviewed the industry and regulatory treatment of damage resistance and damage tolerance to explain the selection of experimental parameters and to provide a framework for understanding results. Airbus POD testing data confirms historical BVID depth recommendations. Boeing uses less conservative BVID dent depths, but is more conservative in carrying ULL requirements to higher energy levels. Substantial Airbus in-service survey data confirms historical damage threat assessments given by [11, 34]. Typical AC20-107B Category 3 damage is beyond the scope of the current study. While a 16-mm complete perforation might be classified as Category 3 (obvious), categorization would vary depending on application as the damage might or might not be obvious in the scope of a pre-flight walk-around, depending on location. Additionally, a total perforation is understood to be a repair scenario in nearly all cases, so evaluating post-impact strength of this type of damage is not of primary concern given the research questions posed in section 1.2. BVID thresholds were found to vary widely by manufacturer, so other criteria besides damage category will be used to choose the three discrete energy levels at which to perform CAI testing.

Realistic damage threats vary by manufacturer and aircraft zone. Current Boeing criteria and older reports [35] indicate that 136 Joules is an appropriate cutoff energy for Category 1 damage to thick parts. Current Airbus criteria seems to be in closer agreement with [11], indicating that 35 Joules is an appropriate realistic energy cutoff for typical zones.

1.6 Contributions of thesis

The primary contribution of this thesis lies in the experimental comparison of two state-of-the-art aerospace materials, especially their damage resistance and tolerance characteristics. The unique aspect of this research is the quality of equivalence
between the two composite architectures; the two composites share constituent materials, thickness, fiber-volume-fraction, and a high degree of similarity of in-plane strengths and stiffnesses. This is primarily a study on the effect of fiber architecture on damage resistance and tolerance performance.

A secondary novel contribution of this work is the demonstration and detailed explanation of the “Berry” experimental compliance method for the experimental characterization of the mode-I fracture toughness of a conventional laminate using double cantilevered beam experiments.

1.7 Thesis outline

Chapter one contained general project goals, parameters, and a literature review section discussing commercial aircraft industry treatment of damage tolerance.

Chapter two details the experimental determination of in-plane material properties and mode-I fracture toughness, as well as providing review of relevant literature to enhance the reader’s understanding of the results.

Chapter three details the experimental determination of material damage resistance properties, as well as providing review of relevant literature to enhance the reader’s understanding of the results.

Chapter four details the experimental determination of material damage tolerance properties, as well as providing review of relevant literature to enhance the reader’s understanding of the results.

Chapter five summarizes all the results and conclusions from Chapters 1-4.
CHAPTER 2
EXPERIMENTAL CHARACTERIZATION OF MECHANICAL PROPERTIES

2.1 Introduction

The purpose of this chapter is to report and discuss the mechanical properties of a three-dimensional (3D) woven ply-to-ply carbon-epoxy aerospace composite and an industry-typical multi-directional NCF laminate of similar constituent materials, in-plane reinforcement distribution, and thickness. Both materials have been evaluated using standard tensile, compressive, in-plane shear, and mode-1 fracture toughness test methods. Fiber-volume fraction was also measured by acid digestion. Since composites based on 3D woven preforms have been characterized in numerous past studies, this introduction section will serve to briefly overview these previous findings. A thorough review of recent advancements in mechanical characterization of 3D woven composites, especially with emphasis on in-plane properties, is given by Saleh and Soutis in [36].

3D woven reinforcements are known to provide a trade off of improved through-thickness performance and damage resistance at the expense of in-plane properties as compared with conventional laminates. Tensile and in-plane shear properties tend to be reduced significantly, while compression properties may be reduced to a lesser extent or even improved [37].

3D woven preforms are typically classified as orthogonal, ply-to-ply, or angled interlock. All three architectures share the characteristic of reduced in-plane reinforcement percent to make room for through-thickness binders; by definition of the weaving process, some in-plane tows are re-purposed as through-thickness binders. The resulting composite is naturally subjected to the rule-of-mixtures strength and
stiffness reduction when compared with a non-woven composite. Additional compromises of in-plane stiffness and strength result from the tow crimp [38], and in-plane strengths are compromised as a result of inelastic tow straightening and tow-matrix interaction [1, 39]. With the exception of recent experimental work in multiaxial weaving [40], woven preforms lack in-plane bias reinforcement, resulting in a scissoring-type shear deformation mechanism and drastically reduced in-plane shear and bias-direction tensile properties [1, 5, 40, 41].

Tensile behaviour of 3D woven composites are presented by Brandt et al. [37], Callus et al. [39], Bogdonavich et al. [41], Warren et al. [1, 2], and London et al. [5].

Brandt et al. [37] reviewed the mechanical performance of various 3D woven composites by comparison of in-plane stiffness and strength, damage tolerance, energy absorption capability, and fracture mechanical properties. In part one, through-the-thickness (TTT) orthogonal woven GFRP composites were tested in tension, compression, and interlaminar shear (ILS) with varying z-direction fiber share between 0 and 10 percent. Increasing z-fiber share was found to significantly reduce tensile strength below the rule-of-mixtures prediction. In the case of compression loading, small decreases in warp-direction strength were offset by significant increases in weft-direction strength. Interlaminar shear strength was not significantly effected by varying z-fiber share.

In part two — a separate experiment — the effects of weave topology were investigated by comparing the performance of TTT orthogonal, ply-to-ply interlock, and angled TTT woven CFRP composites with the same epoxy matrix and equivalent (approx. 6%) z-fiber share. Results were presented for tensile, compressive, interlaminar shear, energy absorption during through-penetration impact tests, compression-after-impact strength, and double-cantilever-beam (DCB) peel tests. The ply-to-ply interlock composite appeared to be generally superior, having the highest tensile strength and modulus in both the warp and weft loading directions
when compared to the other two woven architectures and the 2D baseline. Compressive strength of the ply-to-ply composite was significantly greater than that of the orthogonal weave, but only slightly better than that of the TTT angle interlock. Compression after impact strength was significantly higher than the other two weave types. Energy absorption was significantly higher than the orthogonal weave, however no value was reported for the angled TTT weave. The interlaminar shear strength was naturally significantly lower than that of the angled TTT weave.

3D woven composites of similar architecture have been recently investigated experimentally and numerically by London et al. [5] and Warren et al. [1, 2]. London [5] investigated the bearing-bypass interaction and fatigue of bolted joints using a 3D woven ply-to-ply carbon composite using the same topology and constituents as the material used in this study, with the exception of the warp/weft reinforcement ratio, which has been adjusted from 50/50 to 60/40 for the present study. Warren [1] experimentally and numerically investigated the progressive damage behaviour of bolted connections using several 3D woven architectures.

Mode-I fracture of 3D woven carbon-epoxy composites using the DCB test has been investigated by Fishpool et al. [6]. Three fiber architectures, namely orthogonal (3% and 6% z-direction reinforcement), ply-to-ply (6% z-direction reinforcement, standard and toughened matrix), and TTT angle interlock (6% z-direction reinforcement). Nominal fiber volume fraction for all materials was 55%, and the standard matrix was MVR444 epoxy. The toughened matrix was proprietary.

Preliminary tests of orthogonal specimens resulted in failure at the root of one of the cantilever arms, so the reinforced DCB (RDCB) test was employed for this architecture. The RDCB test employs adhesively bonded reinforcement tabs on the specimen faces to increase the stiffness and prevent failure through bending. This technique has been used in previous studies [42–44] and is still currently used
[45, 46] to evaluate interlaminar toughness of composites with through-thickness reinforcement. The technique was validated by Fishpool on a 2D unidirectional CFRP and on the ply-to-ply reinforced 3D woven composite. These validation tests showed good agreement between the results from DCB and RDCB tests.

For reasons of commercial confidentiality, results were only presented normalized as a percentage of the standard layer-to-layer weave initiation value. This is sufficient to compare the performance of the different woven architectures but not for providing a useful comparison to traditional laminate performance.

Initiation value was found to be dominated by matrix toughness, and not significantly affected by the weave architecture for typical cases where the crack does not initiate directly against a bridging fiber. Propagation values, on the other hand, were significantly affected by weave architecture since crack propagation is governed by the extent of fiber bridging. Experimental results for Mode-I testing are presented in Table 2.1.

Table 2.1: Mode-I fracture results for 3D woven architectures from Fishpool et al. [6]

<table>
<thead>
<tr>
<th>Weave type</th>
<th>Initiation (SEM)</th>
<th>Propagation (SEM)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Standard L2L 6%</td>
<td>99 (5)</td>
<td>1061 (45)</td>
</tr>
<tr>
<td>Toughened L2L 6%</td>
<td>146 (14)</td>
<td>1202 (151)</td>
</tr>
<tr>
<td>Orthogonal 3%</td>
<td>91 (9)</td>
<td>840 (51)</td>
</tr>
<tr>
<td>Orthogonal 6%</td>
<td>111 (8)</td>
<td>1569 (89)</td>
</tr>
<tr>
<td>Angle interlock 6%</td>
<td>159 (14)</td>
<td>704 (26)</td>
</tr>
</tbody>
</table>

Dransfield [43] details the derivation of an analytical expression for $G_{IC}$ of an RDCB specimen including the effects of shear deformation and the inclusion of the metallic reinforcement tab.

Tamus [44] used RDCB specimens but found that debonding of the metallic tabs from the composite specimen was a problem. To alleviate this, an edge-slotted specimen was used to decrease overall loads. In these specimens a 1 mm thickness slot was cut down the midplane of the specimen on either side to reduce the cross-sectional area at the fracture plane. The notch was then sharpened with a thin blade.
2.2 Experimental methods

2.2.1 Specimen preparation

Specimens for the tensile, compressive, shear, and fiber-volume fraction tests were cut from the material panels using a Flow Mach 3 abrasive water jet cutting at 379 MPa. Specimens for Mode-1 fracture toughness testing were cut using a Bosch TC10 wet tile saw to avoid potential upset of the crack front due to water pressure entering the engineered pre-crack. In both cases the specimen dimensions were monitored during machining to ensure adherence with the dimensional tolerances furnished in the relevant test standard. After machining, specimens were stored for a minimum of 24 hours at 21°C, 50% RH before final dimensions were measured and recorded for data reduction.

2.2.2 Digital image correlation

Warren [1, 2] utilized digital image correlation (DIC) to characterize 3D woven composites to avoid potential difficulties from the placement of strain gauges on the relatively large unit-cell of the evaluated materials. Following this example, GOM ARAMIS DIC software has been used for all strain measurements unless otherwise noted.

3D Digital Image Correlation (DIC) is a non-contact method for measuring surface deformations and strains. The system utilizes a stereoscopic, dual-camera arrangement which is calibrated to allow absolute position measurements to be made from the recorded images. Before testing, a random speckle pattern must be applied to the specimen face. Photo-pairs (called stages) are taken during testing. For the present study, stages were collected at 2 Hz. After the test has been completed, DIC software divides the computation region (the painted specimen face) of each image into a grid of square facets. The facets are recognizable by the unique paint pattern in that region. Facet size can be adjusted to suit the density of the paint pattern, and facet
overlap can be adjusted to provide the desired resolution. Displacements of each facet are computed and tracked over the image series, using the first stage as a reference for zero displacement. The surface strain field is computed as the gradient of measured displacements. In the present study, facet size used was 15 pixels for the small-scale material characterization tests, and 30 pixels for the larger CAI tests to account for sparser paint pattern. Facet step was maintained at 13 pixels for both cases. For all testing, the threshold for allowable point-to-point random strain noise was held to ±200µε.

Reported strains were obtained by averaging facet strains over a rectangular measurement region. For tensile and compressive testing, the strain field is uniform, so measurement region needed only to be selected such that defects caused by fixture reflection, free edge effects, or other interferences were avoided. In the case of v-notched shear testing (Section 2.2.5), strain decreases as a function of distance from the V-notch. In this experiment, larger measurement regions yield lower average strain values. A sensitivity study was undertaken to address this issue and is discussed in Section 2.2.5. For consistency between specimens, measurement regions were located using two user-defined points based on easily recognized fixture geometry, and drawn automatically by macro using a consistent width and height. Measurement region sizes for each test are reported in Table 2.2

<table>
<thead>
<tr>
<th>Test</th>
<th>Width (in)</th>
<th>Height (in)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Tension ASTM D3039</td>
<td>0.660</td>
<td>3.344</td>
</tr>
<tr>
<td>Compression ASTM D6641</td>
<td>0.780</td>
<td>0.755</td>
</tr>
<tr>
<td>Shear ASTM D7078</td>
<td>0.28</td>
<td>1.127</td>
</tr>
</tbody>
</table>

2.2.3 Quasi-static tensile testing

Tensile tests were carried out in accordance with ASTM D3039 [47] using a 250kN Instron servo-hydraulic testing machine. Seven to eight specimens were tested for each material at each orientation, 0°, 45°, and 90°. The tests were performed under
position control with a constant actuator speed of 1.27 mm/min. Specimen load and actuator position data were recorded continuously at 100 Hz by the controller PC and at each strain stage (2Hz) by the ARAMIS DIC system. Surface strains were computed during post-processing of DIC data (see Section 2.2.2 for additional detail on DIC data collection.)

Young’s modulus was computed as per ASTM D3039 recommendation between 1000 and 3000 microstrain. A summary of tensile properties is presented in Section 2.3.

### 2.2.4 Quasi-static compressive testing

Compressive tests were carried out in accordance with ASTM D6641 [48] using a 100kN Instron servo-hydraulic testing machine. Seven to eight specimens were tested for each material at each orientation, 0°, 45°, and 90°. Specimen width was adjusted from the standard 12.7 mm to 25.4 mm following the recommendation by London [5] that the specimen width exceed the material unit cell width (15.24 mm) to decrease scatter. Tests were performed under position control with a constant actuator speed of 1.27 mm/min. Specimen load and actuator position data were recorded continuously at 100 Hz by the controller PC and at each strain stage (2Hz) by the ARAMIS DIC system. Surface strains were computed during post-processing of DIC data.

### 2.2.5 Quasi-static shear testing

In-plane shear testing was carried out in accordance with ASTM D7078 [49] using a 100kN Instron servo-hydraulic testing machine. Eight to nine specimens were tested for each material. Since live strain measurement was not feasible using DIC, a constant actuator speed of 1.27 mm/min was used, as per the standard recommendation. Figure 2.1 illustrates the decrease in strain magnitude as a function of distance from the specimen centerline.
As noted in 2.2.2, rectangular measurement regions were tested with widths ranging from 0.100 to 0.500 inches. Figure 2.2 shows the significance of this dimension over the course of a typical 2D NCF shear test. Because the strain range for 3D woven tests was much greater, the dimension carries an even greater significance. Ultimately, the region width 0.28 inches was chosen for consistency with previous work [5].
2.2.6 Fiber volume fraction by acid digestion

Constituent content of both materials was measured in accordance with ASTM D3171-15 Method 1, procedure B; matrix digestion using sulfuric acid and hydrogen peroxide. 3 specimens were tested from each material panel. Two specimens from opposing corner regions and one from the central region to ensure consistent matrix distribution throughout the panel and all mechanical test specimens.

2.2.7 Mode-1 fracture toughness testing

Mode-I interlaminar fracture toughness was characterized using ASTM D5528 [12] standard double cantilever beam (DCB) procedures. A 24-ply unidirectional NCF composite was prepared using the same composition as the multidirectional laminate used in the other characterization tests (sections Sections 2.2.3–2.2.5). A Teflon insert with a thickness of 0.0005 inches was included in the layup at the midplane on one end of the panel. Specimens were cut from the panel such that the Teflon insert
provided a square-front engineered pre-crack of a controlled length as shown in the ASTM D5528 specimen geometry sketch, Figure 2.3.

![Figure 2.3: ASTM D5528 sketch of DCB specimen geometry [12]](image-url)

Attempts were also made to characterize the 3D-woven composite using the double cantilever beam test, although this work was exploratory in nature since this application is well outside the scope of the ASTM test standard. The 3D woven composite for this test was prepared with a woven bifurcation to provide the engineered pre-crack and initiation point for the natural crack. The bifurcation was maintained during the RTM molding and curing process by the inclusion of a 0.0005-inch thickness Teflon insert. Since a uniform path for crack propagation does not exist, it was suspected that DCB specimens would fail at the root of the engineered precrack; the crack propagating away from the midplane and severing one of the “arm” portions, as discussed in [6]. To combat this phenomenon, a path for crack propagation was created along the specimen midplane using a slot of 0.015-inch thickness, reducing the specimen width along the desired gage area. Figure 2.4 is copied from [13], where the inspiration for this approach was derived, and illustrates the principle of the edge-slot modification to a DCB specimen.
The apparatus for slotting the specimen was a standard vertical milling machine with a custom specimen holder. The saw blade used was 3 inches in diameter, 0.015 inches thick, and consisted of a high concentration of fine-grit diamond abrasive. The arbor head was enlarged by the use of 2” washers to help in supporting the fragile blade. Cuts were made with a constant flood of cutting fluid to aid in cooling and debris removal. The apparatus for slitting is shown in Figure 2.5. Machine speed was set to 500 RPM and feed rate was approximately 5 inches per minute. Preliminary DCB trials used edge slots of 0.00, 0.05, 0.1, 0.15, 0.20, and 0.25 inches depth. It was found that even the deepest edge slot, which corresponded to a 50% reduction in gauge width did not sufficiently weaken the fracture plane in relation to the specimen arms, and arm failure by flexure occurred. The test specimen dimensions for each material are given in Table 2.3

Table 2.3: Mode 1 fracture DCB test specimen dimensions

<table>
<thead>
<tr>
<th></th>
<th>3D 60/40 ply to ply</th>
<th>NCF 44/44/11</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>24K/24K</td>
<td>12K</td>
</tr>
<tr>
<td>Length ‘L’ (mm)</td>
<td>230</td>
<td>230</td>
</tr>
<tr>
<td>Initial pre-crack length ‘a₀’ (mm)</td>
<td>50</td>
<td>50</td>
</tr>
<tr>
<td>Specimen width ‘W’ (MM)</td>
<td>25.4</td>
<td>25.4</td>
</tr>
<tr>
<td>Fracture Width ‘b’ (mm)</td>
<td>12.7-25.4</td>
<td>25.4</td>
</tr>
<tr>
<td>Thickness ‘h’ (mm)</td>
<td>4.42</td>
<td>4.42</td>
</tr>
</tbody>
</table>
The specimens and piano hinges (see Figure 2.3) were roughened with 220-grit sandpaper and cleaned with isopropyl alcohol prior to attachment of the hinges with Ashland pliogrip 7779. Hinges were adhered to multiple specimens at once using a 3D printed alignment jig shown in Figure 2.6. Top and bottom hinges could be applied simultaneously on up to 12 specimens. Shim material of an appropriate thickness was inserted to hold the hinges open slightly, thus providing a level surface on which to apply clamping pressure. The specimens were staggered in pairs at each 10 mm pre-crack length for the purpose of creating an empirical compliance to crack-length calibration curve between 50 and 100 mm. Specimens were subsequently separated
from one another by cutting the hinge in between using a vertical bandsaw, or in later trials, a high-pressure abrasive waterjet.

Figure 2.6: 3D printed jig for aligning and gapping specimens during hinge bonding

Specimens were mounted to a servo-hydraulic universal testing frame by the hinges with a 250 N capacity load cell, a custom adapter, and one hydraulic grip, as shown in Figure 2.7. Specimens were then loaded in tension using position control at 5.08 \( \frac{\text{mm}}{\text{min}} \) in accordance with the test standard. Specimen load and actuator displacement data were recorded at 100Hz as the crack propagated along the midplane of the specimen, in other words, as the specimen was peeled apart.
2.2.8 Mode-1 fracture toughness data reduction

Energy release rate, $G_I$, is defined as the energy released per unit area of an extending crack. This quantity can be measured even when the crack cannot grow under a given load case. In such a case, $G_I$ would simply be interpreted as the energy that would be released per unit area of crack extension, should the crack extend. Critical strain energy release rate ($G_{IC}$) or crack resistance ($R$) is the required energy per unit area for a crack to grow. When $G_I$ increases to $G_{IC}$, the crack becomes critical and will extend [50]. Energy release rate is mathematically derived using
elementary mechanics in [51] starting from the energy balance shown in Equation 2.1, where $G$ is energy release rate, $\Delta A$ is the incremental change in crack area, $\Delta U$ is the change in strain energy, and $\Delta W_{\text{EXT}}$ is the change in external work performed on the body.

$$G\Delta A + \Delta U = \Delta W_{\text{EXT}} \quad (2.1)$$

The resulting expression, the Irwin-Kies formula, is given in equation 2.2. This formulation is often cited in literature [51–57], and even used as a validation for new analysis methods [58]. This equation is referred to in this work as the experimental compliance method (ECM) formulation

$$G_I = \lim_{\Delta a \to 0} - \frac{1}{b} \frac{\Delta \Pi}{\Delta a} = \frac{P^2}{2b} \frac{dC}{da} \quad (2.2)$$

where $\Pi$ is the stored potential energy, $a$ is the crack length, $b$ is the width, $P$ is the force, and $C$ is compliance.

The ECM formulation contains no assumptions about specimen geometry or crack type, so it is generally applicable. For the case of a DCB specimen, additional assumptions can be made regarding the form of $\frac{dC}{da}$. For example, the modified beam theory (MBT) method, as given in ASTM D5528 [12] uses the following beam theory assumptions:

A double cantilevered beam, loaded at the tips has a total deflection ($v$):

$$v = \frac{2Pa^3}{3EI} \quad (2.3)$$

where $E$ is the flexural modulus and $I$ is the moment of inertia:

$$I = \frac{bh^3}{12} \quad (2.4)$$

where $b$ is the specimen width and $h$ is the height of only one arm. For any general load scenario, compliance is defined as the inverse of stiffness ($k$):
\[ C = k^{-1} = \frac{v}{P} \] (2.5)

Substituting Equation 2.3 into Equation 2.5:

\[ C = \frac{2a^3}{3EI} \] (2.6)

Substituting Equation 2.4 into Equation 2.6:

\[ C = \frac{8a^3}{Ebh^3} \] (2.7)

Differentiating with respect to cracklength \((a)\):

\[ \frac{dC}{da} = \frac{24a^2}{Ebh^3} \] (2.8)

Substituting Equation 2.8 into the ECM formulation Equation 2.2:

\[ G_I = \frac{P^2}{2b} \frac{dC}{da} = \frac{P^2}{2b} \frac{24a^2}{Ebh^3} = \frac{12a^2P^2}{Eb^2h^3} \] (2.9)

Factoring out the expression for DCB displacement \((v, \text{Equation 2.3})\):

\[ G_I = \frac{12a^2P^2}{Eb^2h^3} = \frac{3P}{2ab}v \] (2.10)

Equation 2.10 is the MBT formula given in ASTM D5528. The advantage of such methods, which use an idealized model for \(\frac{dC}{da}\), are that good approximate results can be had from as little as one physical test. The disadvantage of such methods are that the model may not be quite correct; the beam-roots in a real DCB specimen are not perfectly built-in, as is assumed in the MBT formulation. This fact necessitates an additional empirical correction step [12]. Another disadvantage is that the cracklength \(a\) appears in the formula, indicating that crack length must be physically monitored and recorded during the test. In-situ crack-length measurements can be difficult and
unreliable for the reasons that crack fronts may not remain linear, or may progress asymmetrically.

The Berry method, introduced by [13] is one technique for implementing the ECM. In general, the ECM involves measuring the quantity \( \frac{dC}{da} \) experimentally, as opposed to analytically. The Berry method specifically uses a power relation (Equation 2.11) to curve-fit the experimental compliance to crack-length data. This technique has been effectively utilized by many studies [57, 59, 60], and been shown to produce moderately lower scatter than methods utilizing in-situ crack measurement [61].

\[
C = qa^r
\]  \hspace{1cm} (2.11)

where \( C \) is compliance, \( a \) is the crack length, and \( q \) and \( r \) are least-squares fitting parameters.

The current study utilized a compliance experiment of 12 specimens per material; two each at 50, 60, 70, 80, 90, and 100 mm engineered precrack lengths. The compliance of each specimen was tested without propagating the crack, guaranteeing a straight and normal crack front. The compliance of each specimen was plotted against its crack length, and the 12 specimens together were curve-fit using the power form in Equation 2.11. For the 2D unidirectional composite, the resulting fitting parameters and coefficient of determination are given in Equations (2.12)–(2.14). The resulting closed-form expression for specimen compliance as a function of crack length is given in Equation 2.15.

\[
q = 6.344 \times 10^{-7} \hspace{1cm} (2.12)
\]

\[
r = 2.772 \hspace{1cm} (2.13)
\]

\[
R^2 = 0.9994 \hspace{1cm} (2.14)
\]

\[
C = 6.344 \times 10^{-7}a^{2.772} \hspace{1cm} (2.15)
\]
After the compliance experiment was completed, six \( G_{IC} \) specimens were tested with nominal pre-crack length \( a_0 = 50 \text{mm} \) as per the ASTM standard. Tests were run until total crack extension was greater than \( a = 100 \text{mm} \).

To compute \( G_{IC} \), Equation 2.15 was differentiated:

\[
\frac{dC}{da} = 1.759 \times 10^{-6} a^{1.772} \tag{2.16}
\]

and substituted into Equation 2.2:

\[
G_I = \frac{P^2}{2b}a^{1.772} \tag{2.17}
\]

Equation 2.15 was used in alternate form (equation 2.18) to identify the equivalent crack-length \( a \) based on the measured compliance for every data point recorded during the test.

\[
a = \left( \frac{C}{6.344 \times 10^{-7}} \right)^{\frac{1}{1.772}} \tag{2.18}
\]

The resulting relation \( G_{IC}(a) \), or R-curve, for all specimens was plotted on a single axes.

\[\textbf{2.3 Results and discussion}\]

\[\textbf{2.3.1 Summary of properties}\]

Both materials were evaluated with standard acid digestion [62], tensile [47], compressive [48], v-notch shear [49], and mode-1 fracture toughness [12] experiments. The average fiber-volume fraction (ASTM D3171-15 Method 1 Proc. B) for the 3D ply to ply (24k/24K) material was 58.4\% with standard deviation 1.06\%, and for the NCF (12K) material was 57.2\% with standard deviation 0.56\%. Experimental tensile, compressive, and shear properties (ultimate strengths and moduli with associated coefficients of variation) for each material and orientation are presented immediately below in Tables 2.4–2.7.
Table 2.4: Summary of tensile properties

<table>
<thead>
<tr>
<th>Material</th>
<th>Orientation</th>
<th>Ultimate Strength MPa (COV)</th>
<th>Young’s Modulus GPa (COV)</th>
</tr>
</thead>
<tbody>
<tr>
<td>3D 60/40 ply to ply 24K/24K</td>
<td>Warp</td>
<td>1120 (8.00%)</td>
<td>85.8 (4.51%)</td>
</tr>
<tr>
<td></td>
<td>Bias</td>
<td>176 (2.92%)</td>
<td>17.5 (5.52%)</td>
</tr>
<tr>
<td></td>
<td>Weft</td>
<td>393 (7.54%)</td>
<td>43.5 (3.69%)</td>
</tr>
<tr>
<td>NCF 44/44/11 12K</td>
<td>0°</td>
<td>983 (1.96%)</td>
<td>72.2 (3.76%)</td>
</tr>
<tr>
<td></td>
<td>45°</td>
<td>709 (5.21%)</td>
<td>51.8 (1.87%)</td>
</tr>
<tr>
<td></td>
<td>90°</td>
<td>566 (6.37%)</td>
<td>40.7 (2.76%)</td>
</tr>
</tbody>
</table>

Table 2.5: Summary of compressive properties

<table>
<thead>
<tr>
<th>Material</th>
<th>Orientation</th>
<th>Ultimate Strength MPa (COV)</th>
<th>Young’s Modulus GPa (COV)</th>
</tr>
</thead>
<tbody>
<tr>
<td>3D 60/40 ply to ply 24K/24K</td>
<td>Warp</td>
<td>399 (10.2%)</td>
<td>75.5 (5.04%)</td>
</tr>
<tr>
<td></td>
<td>Bias</td>
<td>181 (2.14%)</td>
<td>16.7 (4.22%)</td>
</tr>
<tr>
<td></td>
<td>Weft</td>
<td>239 (6.87%)</td>
<td>41.9 (6.38%)</td>
</tr>
<tr>
<td>NCF 44/44/11 12K</td>
<td>0°</td>
<td>328 (9.13%)</td>
<td>64.0 (2.03%)</td>
</tr>
<tr>
<td></td>
<td>45°</td>
<td>308 (10.2%)</td>
<td>46.9 (7.19%)</td>
</tr>
<tr>
<td></td>
<td>90°</td>
<td>293 (7.79%)</td>
<td>38.8 (4.09%)</td>
</tr>
</tbody>
</table>

Table 2.6: Summary of in-plane shear properties

<table>
<thead>
<tr>
<th>Material</th>
<th>Orientation</th>
<th>Ultimate Strength MPa (COV)</th>
<th>Young’s Modulus GPa (COV)</th>
</tr>
</thead>
<tbody>
<tr>
<td>3D 60/40 ply to ply 24K/24K</td>
<td>Warp</td>
<td>115 (2.18%)</td>
<td>5.31 (4.41%)</td>
</tr>
<tr>
<td>NCF 44/44/11 12K</td>
<td>0°</td>
<td>258 (4.20%)</td>
<td>18.3 (0.98%)</td>
</tr>
</tbody>
</table>

Table 2.7: Summary of mode-I fracture toughness properties

<table>
<thead>
<tr>
<th>Material</th>
<th>Initiation $\frac{J}{m^2}$ (COV)</th>
<th>Propagation $\frac{J}{m^2}$ (COV)</th>
</tr>
</thead>
<tbody>
<tr>
<td>NCF 44/44/11 12K</td>
<td>1097 (8.71%)</td>
<td>2112 (13.5%)</td>
</tr>
</tbody>
</table>

2.3.2 Discussion of tensile test results

The stress-strain response of the 3D woven composite was observed to be bi-linear when loaded on-axis (Figure 2.8b). The slope change, or softening, was much more
pronounced for the weft-direction loading (Figure 2.10b) than for the warp-direction loading. This phenomenon has been observed by other experimenters [1, 63] and has been attributed to the “plastic straightening” of the crimped tows. A crimped tow has the tendency to straighten under tension and exerts strain on the surrounding matrix. When matrix failure occurs, some composite action is lost and specimen softening is observed. Notably, ultimate strain to failure was fairly consistent with the benchmark 2D NCF composite, which exhibited a linear response to failure for both on-axis and off-axis loading (Figures 2.8a, 2.9a and 2.10a).

In the case of off-axis loading of the 3D woven composite (Figure 2.9b), highly ductile nonlinear response was observed. This is consistent with the findings of other experimenters studying 3D woven composites of all architectures lacking bias-direction reinforcement [1, 5, 40, 41].

![Figure 2.8: Tensile test results for 0°/warp loading direction: (a) 2D NCF stress-strain plots and (b) 3D woven stress-strain plots](image-url)
2.3.3 Discussion of compressive test results

Compression tests of specimens loaded in the 0° or warp direction showed quasi-linear behavior to failure (Figure 2.11). Off-axis loading of the 3D woven composite resulted in extremely ductile failure behavior and low ultimate strength (Figure 2.12b) due to the lack of bias reinforcement and the same scissoring deformation mechanism.
discussed in section 2.3.2. The weft-direction compressive testing of the 3D woven composite resulted in a more ductile failure and higher scatter when compared to the 2D NCF baseline (Figure 2.13).

Figure 2.11: Compressive test results for 0°/warp loading direction: (a) 2D NCF stress-strain plots and (b) 3D woven stress-strain plots

Figure 2.12: Compressive test results for 45°/bias loading direction: (a) 2D NCF stress-strain plots and (b) 3D woven stress-strain plots
2.3.4 Discussion of shear test results

Figure 2.14 presents the stress-strain responses obtained for the in-plane shear tests. The 3D woven composite exhibited a highly nonlinear stress-strain curve with no discernible elastic regime. The 2D NCF baseline composite exhibited quasi-linear response to approximately 220 MPa followed by a short ductile failure.
2.3.5 Discussion of fracture toughness test results

The compliance calibration curve for the unidirectional NCF is shown along with the experimental data (two specimens at each crack length) in Figure 2.15. The power fit \( C = 6.344 \times 10^{-7}a^{2.772} \) was very good with a coefficient of determination \( R^2 = 0.9994 \). Panel geometry limited the calibration to the range 50–100 \( mm \).

![Compliance to crack length calibration curve for unidirectional NCF](image)

Figure 2.15: Compliance to crack length calibration curve for unidirectional NCF

The 2D unidirectional NCF specimens exhibited run-arrest fracture behavior due to the presence of the polyester binding threads. A typical load-displacement plot is shown in Figure 2.16a. The load peaks, indicated by blue circles, were the only points processed to produce the \( G_{IC} \) R-curve. An image of the binding threads post-test is shown in Figure 2.16b.

The critical load peaks were processed as described in Section 2.2.8 to produce strain energy release rates, \( G_I \), at each point. The collection of these points for all critical load peaks for all specimens describe the R-curve which is presented in Figure 2.17. The mean propagation toughness value (2112 \( \frac{J}{m^2} \)) was obtained by averaging all points in the indicated range, 80–100 \( mm \). The upper bound of this range was imposed by the range of the compliance calibration, and the lower bound was defined.
Figure 2.16: Run-arrest behavior: (a) typical load-displacement plot and (b) view of specimen mouth showing broken, hanging stitch threads by an iterative convergence experiment which indicated that 80 mm was the lowest boundary for which the mean propagation toughness had stabilized. The dashed lines describe propagation toughness value scatter between specimens by illustrating one standard deviation for the averaged datapoints.

Figure 2.17: $G_{IC}$ R-curve for mode-I fracture testing of 2D NCF material

The point of initiation corresponding to the reported mean initiation toughness $1097 \frac{J}{m^2}$ was taken as the first critical load preceding a force drop of greater than one pound. It was observed that the several failures below this magnitude which
were processed reflected a negligible change in compliance and were likely caused by the failure of inconsequential partial bonds near the tip of the engineered pre-crack. These failures can be observed to trend roughly vertically on the R-curve, indicating that crack extension was negligible for failures below the reported initiation value.

2.4 Conclusions and recommendations

The 2D NCF baseline composite and the 3D ply-to-ply woven composites share comparable strengths and stiffnesses in tension and compression when loaded on-axis. The 3D ply-to-ply woven composite lacks bias-direction reinforcement and correspondingly behaves in a ductile manner with low strength when subjected to off-axis or in-plane shear loading. Extra caution should be exercised when applying this 3D woven architecture in monocoque skin applications or other structural applications with large off-axis load components.

The experimental compliance “Berry” method of mode-I fracture toughness experiment is recommended for convenient testing and data reduction.

To evaluate the mode-I fracture toughness of the 3D woven ply-to-ply composite using the DCB test, it is necessary to increase the strength of cantilevered arms relative to the intended fracture plane. The current study attempted to accomplish this by reducing the area of the fracture plane up to 50%. This quantity proved insufficient, and similar studies using this technique should expect to reduce the gauge region width by greater than 50% in order to successfully propagate a crack without bending failure of the arms. Other studies have successfully used reinforcing tabs adhesively bonded to the specimen faces as described in [6, 42–46]. This method is recommended for future studies characterizing mode-1 fracture toughness of 3D woven composites.
CHAPTER 3
EXPERIMENTAL CHARACTERIZATION OF DAMAGE RESISTANCE

3.1 Introduction

One of the most critical limitations of conventional laminated carbon fiber reinforced polymers (CFRP) is their susceptibility to delamination. In the case of low-velocity impact, this failure mode presents the potential for critical strength and stiffness reduction without commensurate visual indication of damage. In light of this unique response, the design and certification of CFRP primary aircraft structures has required that new methodology be developed. Regulatory agencies such as the U.S. Federal Aviation Administration (FAA) and European Union Aviation Safety Agency (EASA) have published guidance [7, 29], indicating that load-carrying requirements should be directly dependent on damage detectability. Namely, undetectable damage is required to be ultimate-load capable for the structure’s lifetime [8, 10, 16].

The most commonly used design methodology involves the development of zone-dependent impact threat assessments based on in-service damage records, threshold of detectability studies linked to in-service inspection methods [9, 11], and material damage resistance and tolerance characterization. Material damage resistance characterization describes the relationships between incident impact parameters and damage detectability metrics: for example, the indentation depth or ultrasonic C-scan damage area as a function of incident impact energy levels. Damage tolerance characterization describes the relationships between damage detectability metrics and post-damage structural performance: for example, the compression-after-impact strength or fatigue life as a function of damage area.
With these tools, primary structure designs can be produced which ensure against failure with the same or greater confidence as the equivalent metallic structure, all the while utilizing the improved fatigue and corrosion properties of CFRP to significantly increase inspection intervals and decrease scheduled maintenance burdens [18, 64]. Nevertheless, accidental damage and delamination are still of concern, particularly due to ground handling and maintenance [17]. The next generation of composite materials are being developed which eliminate the delamination failure mode and offer enhanced through-the-thickness performance using three-dimensional pre-formed fiber architectures [1, 5]. The use of 3D woven composites may have the potential to further reduce life cycle cost of aircraft structures by reducing the probability of detectable damage and associated unscheduled maintenance occurrence.

This chapter compares the damage resistance performance of the 3D woven composite to that of a conventional multi-directional laminate of comparable thickness, stiffness, and strength. (Material properties were reported in section 2.3.1).

Damage resistance was evaluated using standard low-velocity drop-weight impact testing (AITM 1-0010 with minor adaptations) under Barely Visible Impact Damage (BVID)-type conditions. Tests were performed with impact energy ranging from 20 to 130 Joules using a 16 mm hemisphere striker. Resulting damage was assessed using energy absorption analysis, damage area measurement with ultrasonic through-transmission C-scan imaging, failure mode analysis using micro-computed tomography (µCT), and simple indentation depth measurements.

Comparisons were made between the responses of the two material systems by presenting the energy absorption, dent-depth and damage-area metrics as functions of incident impact energy. This approach yielded a comprehensive understanding of the damage containment properties of the 3D woven composite. Ultrasonic C-scan and µCT scans yielded detailed information on damage mechanisms in the two material systems.
The 3D woven composite was found to absorb greater energy per unit damage area, less total energy for impacts over 50 Joules, and to suffer less damage by the metrics of C-scan area and permanent indentation depth.

3.2 Experimental methods

3.2.1 Specimen preparation

Specimens for low-velocity impact tests were cut to $100 \times 150 \text{ mm}$ in accordance with Airbus Industries AITM 1-0010 [14], a combined test standard for the evaluation of material damage resistance (drop-weight impact and non-destructive inspection), and damage tolerance (quasi-static compression-after-impact testing). Specimens were cut from the material panels using a Flow Mach 3 abrasive water jet cutting at 379 MPa. The specimen dimensions were monitored during machining to ensure adherence with the dimensional tolerances furnished in the test standard. After machining, specimens were stored for a minimum of 24 hours at 21°C, 50% RH before final dimensions were measured and recorded.

In the first batch of manufactured specimens, waterjet piercing was performed too close (12.7 mm) to the specimen edges, causing matrix damage in the corner of both 2D and 3D specimens. This mistake was apparent only after ultrasonic C-scan inspection. Figure 3.1 shows ultrasonic C-scan images of typical pierce damage on early specimens. Because healthy material was present in all cases between the impact damage and the waterjet piece damage, it was decided that the damaged corners would be unlikely to have a measurable effect on the damage resistance test results. Specimens damaged in this way were used without special treatment for damage resistance evaluation. Pierce distance was increased to $25 \text{ mm}$ from the specimen edge in subsequent batches to eliminate the problem.
3.2.2 Drop-weight impact testing

Drop-weight impact testing was carried out using Airbus Industries AITM 1-0010 [14] as the primary reference. This standard was selected over ASTM D7136 [65] because the AITM 1-0010 compression-after-impact (CAI) fixture design allows for more accurate application of boundary conditions [66]; an important consideration for the Chapter 4 evaluation of damage tolerance.

A Ceast Fractovis Plus (9350 series) drop tower was used to perform the testing. Figure 3.2a shows the drop tower apparatus with a test specimen in place on the support fixture. Figure 3.2b shows the important support fixture dimensions in mm [inches], including the standard AITM 1-0010 test specimen, which is shaded. Specimens were clamped lightly with four neoprene-tipped toggle clamps to prevent the specimen from rebounding. As indicated, clamp tips were positioned inside the

Figure 3.1: Ultrasonic C-scan images showing typical pierce damage caused by inadequate tab length on early specimens: (a) 2D and (b) 3D
cut-out of the support fixture so as not to influence specimen response during the impact event.

Figure 3.2: Drop weight impact testing equipment: (a) Fractovis Plus drop-weight impact apparatus and (b) Dimensions of specimen and support fixture

A 16-mm hemispherical striker was used with an embedded 40-kN load cell. A Ceast DAS64K high-speed data acquisition system was used to obtain real-time contact force data from the load cell during impact. Data was recorded at 4000 kHz. Given the total falling mass—striker, carriage, and additional weights—the CEASTVIEW software used numerical integrations of the contact force history to obtain velocity, position, and absorbed energy histories. Equation 3.1 shows the integration used to obtain the displacement history of the striker throughout the impact event. Equation 3.2 shows the integration used to obtain the stored energy history of the specimen throughout the impact event. Additional discussion of the numerical integration scheme is provided in [67].

\[
\varepsilon_i = \int \int_{i} \frac{F(t) - g(M_{\text{total}})}{M_{\text{total}}} d^2t \quad \quad (3.1)
\]
where: $\varepsilon$ is striker displacement, $F(t)$ is the load cell force signal, $g$ is gravitational acceleration, $M_{total}$ is the total falling mass, and $t$ is time.

$$E_i = \int \epsilon F(\varepsilon) d\varepsilon$$

(3.2)

where: $E_i$ is the stored energy at the $i$th data point.

Kinetic energy of the striker is stored by the test specimen primarily as elastic strain energy and released as fracture surface energy. Additional minor losses are incurred through acoustic emission and plastic deformation. The difference between the peak energy storage and the final energy storage, or energy absorption, represents the elastic strain energy used to rebound the striker. Figure 3.3 shows the stored energy plots for 30, 60, and 90 Joule impact tests.

![Figure 3.3: Stored energy vs. time for 30, 60, and 90-J strikes](image)

Single specimens from both material systems were subjected to impacts at 20, 25, 40, 50, 60, 70, 90, 110, and 130 Joules. Three specimens from each material system were subjected to 30-J impacts. Falling mass was adjusted to comply with the drop height (> 0.5 m) requirement set in AITM 1-0010, as well as regulating impact velocity, which was maintained between 3.0 and 4.25 m/s for this study. Table 3.1 shows the impact parameters used during damage resistance screening.
Table 3.1: Impact parameters used for damage resistance experiments

<table>
<thead>
<tr>
<th>Impact Energy (J)</th>
<th>Falling Mass (kg)</th>
<th>Drop Height (mm)</th>
<th>Impact Velocity (m/s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>20</td>
<td>3.998</td>
<td>510</td>
<td>3.163</td>
</tr>
<tr>
<td>25</td>
<td>3.998</td>
<td>637</td>
<td>3.536</td>
</tr>
<tr>
<td>30</td>
<td>5.498</td>
<td>557</td>
<td>3.303</td>
</tr>
<tr>
<td>40</td>
<td>5.498</td>
<td>742</td>
<td>3.814</td>
</tr>
<tr>
<td>50</td>
<td>9.498</td>
<td>537</td>
<td>3.245</td>
</tr>
<tr>
<td>60</td>
<td>10.498</td>
<td>583</td>
<td>3.381</td>
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<tr>
<td>70</td>
<td>10.498</td>
<td>680</td>
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<tr>
<td>90</td>
<td>10.498</td>
<td>874</td>
<td>4.141</td>
</tr>
<tr>
<td>110</td>
<td>20.498</td>
<td>547</td>
<td>3.276</td>
</tr>
<tr>
<td>130</td>
<td>20.498</td>
<td>647</td>
<td>3.561</td>
</tr>
</tbody>
</table>

Immediately following impact, maximum indentation depth was measured to within 0.05 mm using a digital depth micrometer. Maximum indentation was referenced against four datum points located 20 mm from the maximum indentation along each principal material axis. Because of the delayed nature of aircraft inspections, post-impact material relaxation must be accounted for. DIC analysis of the impacted specimens was preferred to fully characterize the dent relaxation behavior, however other tests requiring the DIC equipment had greater priority. To account for dent relaxation, multiple manual measurements were carried out on each specimen over approximately one month following the impact event. Dent depth vs. time was plotted for each specimen. Since the impact testing took place over several weeks, it was not always convenient to take dent measurements at any specific interval after the impact. Taking inspiration from the viscoelasticity approach for isochronous stress-strain curves, the dent depth vs. energy plot was developed for t = 16 days post-impact by interpolating between measured data-points. 16 days was chosen since it was observed that the relaxation occurred primarily in the first week, so stabilization of the depth was expected by 16 days. Additionally, 16 days was the longest interval for which data was collected for all specimens. During the relaxation period, test specimens were allowed to rest in the standard environment (21°C, 50%RH). Over
the 16-day relaxation period, average indentation depths decreased by 6% for the 2D material and 11% for the 3D material.

### 3.2.3 Ultrasonic C-scan

Specimens were ultrasonically scanned using a JSR Ultrasonics DPR300 pulser-receiver with two 0.75 inch, 5.0 MHz transducers in through-transmission mode. The couplant used was deionized water (immersion scan) with a total path-length of 6 inches. Pulse amplitude was 527 Volts using 59 Ohms damping. Pulse repetition frequency was 100 Hz, and receiver filter passband was set between 1.0 and 15 MHz.

A preliminary gain study was carried out on one 3D and one 2D impacted specimen to determine an appropriate receiver gain for the remaining trials. Gains tested varied between 6 and 20 dB. Detected damage area was not found to vary significantly with gain over this range. It was noted that higher gain levels tended to produce images with a wider range of grayscale values which increased resolution and ease of visual differentiation between damaged and non-damaged regions. For this reason, maximum gains were chosen with the constraint that no saturation (white pixels) occurred. These criteria resulted in a receiver gain level of 9 dB for the 2D baseline specimens and 8 dB for the 3D woven specimens.

During trials, C-scan damage area was measured using three techniques, here referred to as “ASTM ray”, “arbitrary polygon” and “pixel counting”. The ASTM ray technique is presented in ASTM D7136 [65] and involves placing rays radially outward from the center of damage at each 30 degrees. A polygon is inscribed between the intersections of each ray and the damage boundary. The damage area is approximated as the area of the inscribed polygon. In practice, it was found that irregular damage geometries such as those occurring in the delaminations of the 2D baseline material were not accurately approximated and large areas might go uncounted. This issue persisted even with more closely (15 degrees) spaced rays, as illustrated in Figure
3.4a. Figure 3.4b shows the arbitrary polygon method, where a user manually selects points which create a border around the visible damage.

![Figure 3.4a](image1.png) ![Figure 3.4b](image2.png)

**Figure 3.4**: C-scan area measurement techniques: (a) Deficiency of ‘ASTM ray’ method illustrated on 50-J 2D specimen with irregular damage shape, and (b) Arbitrary polygon area selection method

The pixel counting method was undertaken in an attempt to automate damage area measurement. This task was a two-step process utilizing an automatic thresholding algorithm to assign pixels below threshold intensity (low signal reception) to black. Continuous regions of black pixels were then automatically selected and counted. To obtain the damage area in $mm^2$, the ratio of counted damage pixels to total pixels was multiplied by the known specimen area.

Five automatic thresholding algorithms were tested: four of them using implementations available in the Fiji distribution of Image-J [68], a popular open-source image-processing program; Huang Fuzziness [69], Maximum Entropy [70], Li Minimum Cross Entropy [71], and Histogram Mean [72]. One additional histogram-based technique was implemented using 3D Composites Studio (3DCS) Verifier, which is an image analysis software developed by AEC. All thresholding methods were found
to reasonably approximate the visible damage boundary, with the exception of the Huang Fuzziness method, which significantly under-represented the damage region in this application.

Three area selection algorithms were tested: two of them using Image-J (‘Wand’ and ‘Lasso and Blow’ tools), and one using 3DCS Verifier. These tools proved effective on simple geometries but failed to accurately capture complex or discontinuous damage regions. Additionally, following an example by Wronkowicz et al. [73] it was desired to count inclusions of non-damaged or apparently non-damaged material within the damaged region, which was not accomplished using these tools. Figure 3.5a illustrates the difficulty encountered in capturing non-continuous, irregular areas such as those occurring on the 130-J 2D impact specimen. Yellow points represent the perimeter of the damage area determined via the automated selection tool operating on the thresholded image. Figure 3.5b shows the arbitrary polygon region, much closer to the intended result.

Figure 3.5: C-scan area measurement techniques: (a) Li Entropy threshold with Image-J Wand automatic region selection, and (b) Unmodified image with Arbitrary Polygon damage measurement
For this study, the arbitrary polygon technique was selected to report C-scan damage areas since it did not suffer the shortcomings of the other two methods.

### 3.2.4 Micro-computed tomography

µCT scans were performed at AEC using 83-micron resolution. 3DCS Verifier was used to perform analysis. Damage profiles are presented and analyzed using cross-sectional views oriented normal to the warp (0-degree), and weft (90-degree) directions and centered in the specimen. Figure 3.6 illustrates the cut sections which are presented.

![µCT cross-sectional views used for damage analysis](image)

Figure 3.6: µCT cross-sectional views used for damage analysis

### 3.3 Results

Damage resistance was characterized by the measurement of indentation depth, C-scan damage area, peak force, and absorbed energy, using single specimens at each impact energy level, except in the case of the 30-J level, where three specimens were evaluated as per AITM 1-0010 [14]. The net performance differences between the two materials varied with impact energy. Specifically, the 2D NCF material exhibited a bilinear response for both dent depth and C-scan area relative to impact energy. The
slope of the dent depth vs. impact energy response increased sharply at 40 J, while C-scan area slope decreased moderately at 65 J. The 3D woven material exhibited linear responses for both metrics to high energy levels.

This discussion section will explain these behaviors, drawing on the evidence of impact force histories, ultrasonic C-scans, and µCT images particularly at the 30 J, 50 J, and 90 J impact energy levels. Summary plots of the damage resistance metrics relative to incident energy are also presented.

3.3.1 Impact force and absorbed energy

In the range 0–50 J, the 2D NCF and 3D woven composites displayed similar force-displacement and kinetic energy storage behavior. The 3D woven composite was slightly more compliant, achieving lower peak force but higher maximum deflection. No significant failures occurred beyond matrix cracking, which is evident by the irregular noise pattern of plate vibration in Figure 3.7, which shows the force-time and force-displacement curves for 30 J impacts.
Figure 3.7: Force histories for 30-J impact tests: (a) Force vs. time and (b) Force vs. displacement.

For impacts of 50 J and higher, contact force on the 2D NCF material reached about 14 kN peak. 14 kN proved to be an incipient force for failure, where significant force drop occurred. Contact force for the remainder of the impact event remained fairly constant in the neighborhood of 9–11 kN until striker velocity decreased to zero (peak deflection.) 3D woven specimens experienced earlier failures between 9 and 13 kN. The contact force after first failure was subsequently regained and exceeded as additional load paths were activated within the material.

First failure for both materials was observed to correspond with breaching of the back face. In the case of the 2D NCF, the failure mode was de-bonding of the back-face 45-degree surface tows, which correspond to the diagonal protrusions seen
in the ultrasonic C-scans shown earlier in section 3.2.3. In the case of the 3D woven composite, the woven architecture necessitated not only the debonding of the woven monolith, but also the rupture of tows, which explains the repeating high loads after first failure. When a given tow ruptured, the load redistributed to neighboring tows, which also loaded to failure.

These behaviors are evident in the force-time histories beginning at 50 J, however they are more obvious at higher energy levels where a relatively long period of continuous material failure occurred. Figures 3.8a and 3.8b show the force-time and force-displacement curves of the 50-J impact tests. Figure 3.9 shows impact plots for 90-J tests, plainly illustrating the steady force plateau corresponding to the progressive delamination of the 2D NCF back-face surface tows, as well as the oscillating force caused by the successive failure of tows in the 3D woven composite.
Figure 3.8: Force histories for 50-J impact tests: (a) Force vs. time and (b) Force vs. displacement
Through increasing energy levels, the 2D NCF specimens accumulated more damage and became more compliant, lengthening the impulse required to stop the striker. There was an inversion of trend at 65 J whereby the normally stiffer 2D NCF composite achieved lower peak force and greater impact duration relative to the 3D composite.

Figure 3.10a shows the absorbed energy by both material systems for all tests, plotted relative to the initial impactor kinetic energies, such that the line $y = x$ represents complete energy absorption with no energy reflected, i.e., no striker rebound. The differences are minimal up to 50 J, where significant failures were observed. In
the range 50–110 J, the 3D woven material stored and reflected a greater proportion of energy as elastic strain as opposed to releasing the energy in damage creation. At 130 J, both materials were fully perforated. In the case of perforation, the striker carried kinetic energy through the plate as it passed. Because energy absorption was calculated based on the integration of force history, the actual absorbed energy could not be calculated and was taken as 100% of incident.

Although total energy absorbed has been a typical measure of impact resistance, Kuboki et al. [74] reported no change in total absorbed energy between impact specimens of greatly varying fracture toughness. The authors suggested an alternative metric for impact resistance, namely absorbed energy per unit damage area. This metric is presented for the present study in Figure 3.10b. Although measurement noise and material scatter are present, linear trends are clearly visible in the range 50–110 J. The 3D woven composite exhibited a steady improvement over the range when compared to the 2D NCF using this metric, owing to the mechanical energy dissipation mode of tow breakage as opposed to delamination.

![Figure 3.10: Absorbed energy plots: (a) total absorbed energy vs. incident energy and (b) absorbed energy per unit C-scan area vs. incident energy](image)

Figure 3.10: Absorbed energy plots: (a) total absorbed energy vs. incident energy and (b) absorbed energy per unit C-scan area vs. incident energy
3.3.2 Ultrasonic C-scan and indentation depth damage analysis

As was previously noted, the 2D NCF exhibited two distinct modes of delamination growth. At low energies, back-face damage was minor, and projected delamination area extended outward radially in a round, to elliptical, to diamond-shaped profile as seen in Figures 3.11a and 3.11b. The second phase of delamination growth occurred beyond the 50-J energy level where critical load for back-face delamination was first achieved. As seen in Figures 3.11b–3.11d, delamination growth between 50 J and 130 J was primarily accumulated by the widening and lengthening of this debonded back-face region, while the area of the diamond-shape core delamination region increased only marginally.

Damage area in the 3D woven material increased linearly to 90 Joules, then decreased markedly up to perforation energy. Figures 3.12a–3.12d show the succession of damage outcomes in the 3D woven composite with increasing energy strikes.

Figure 3.11: Succession of damage outcomes in the 2D NCF composite with increasing energy strikes: (a) 20 J, (b) 50 J, (c) 90 J, and (d) 130 J
Figure 3.12: Succession of damage outcomes in the 3D woven composite with increasing energy strikes: (a) 20 J, (b) 50 J, (c) 90 J, and (d) 130 J

The 3D woven composite outperformed the 2D non-crimp composite in the containment of damage. Figure 3.13a shows the total projected damage area plotted as a function of incident impact energy. The bi-linear response of the 2D NCF clearly summarizes the dual energy absorption modes. The same phenomenon had a significant effect on the permanent indentation. Figure 3.13b shows the permanent indentation depth in its relaxed state for both materials. The 2D NCF exhibits a sharp knee at 40 Joules, 50 Joules being the first energy level with significant back-face delamination. The 3D composite presented a much tougher barrier to striker indentation by forcing the impact loads to be carried by tows. The 3D woven composite indentation increased linearly with energy over the investigated energy range.
3.3.3 Microcomputed tomography damage analysis

In the low-energy range 0–30 J, the 2D NCF and 3D woven composites displayed similar load-displacement and kinetic energy storage behavior, as well as incurring similar damage area and permanent indentation. Despite these similarities, the damage progression in the two materials were already distinctly different. The 2D material exhibited diagonal through-thickness matrix cracking accompanied by significant delaminations, which occurred primarily at interfaces of 90-degree orientation separation. The delaminations were roughly peanut-shaped and aligned with the tows of the lower ply. These observations match the expectations for low velocity impact of composite laminates from Abrate [75] and experimental observations in the literature [76, 77]. Figure 3.14 shows the 0-degree section of a 2D specimen impacted at 30 J. Matrix cracking and delaminations at the interfaces between plies 4 and 5, and plies 14 and 15 were observed on the 0-degree section for all three specimens.
Figure 3.14: $\mu$CT image of 2D NCF specimen impacted at 30 J (0-degree view)

The 3D woven specimens impacted at 30 J exhibited diagonal shear cracking when viewed in both the warp and weft planes, suggesting roughly conical damage morphology as is typical in the case of impact on thick, brittle plates. Figure 3.15a shows a warp cross-section located between weft tows. This cross-section is about 2 mm off-center in the specimen, since the impact was not targeted with respect to the unit cell. At this cross-section, the conical shear cracks progress freely, splitting impregnated warp tows and neat resin pockets. Examining a warp section within a stack of weft tows (Figure 3.15b), no shear cracks are observed, since cracks at this orientation would necessarily sever the weft tows. A similar pattern, i.e. shear cracking occurring between stacks of tows, was observed in the weft section, and was observed for all 30-J, 3D specimens.

Figure 3.15: $\mu$CT images of 3D woven specimen impacted at 30 J: (a) warp view between weft tows and (b) warp view, weft tows visible

As discussed in Section 3.3.1, significant failures occurred in both materials during the 50-J impact test. Back-face delamination was the primary energy absorption mechanism of the 2D NCF material. Successive tow failure was a primary energy
absorption mechanism of the 3D woven composite. Figure 3.15b demonstrates that
tow breakage did not occur at low energy. Figure 3.16 illustrates the failures in the
3D composite when subjected to the 50-J impact.

Figure 3.16a reveals similar diagonal shear cracking in the warp cross section as
that which was seen at the 30 J energy level, although intensified. Weft tow breakage
occurred near the specimen centerline, as well as minor longitudinal cracks tracing
the weft tows (Figure 3.16b.) Minor shear cracks are visible in the weft section, along
with much more pronounced longitudinal cracks which developed along the warp tows
(Figure 3.16c.)

Figure 3.16: \(\mu\)CT images of the 3D specimen impacted at 50 J: (a) warp view
between weft tows, (b) warp view with weft tows visible, and (c), weft view

Figure 3.17 illustrates the increased delamination opening displacement, as well as
increased volume of delaminations within the central core region beneath the striker
on a 2D NCF specimen impacted at 50 J.
Figure 3.17: $\mu$CT images of 2D NCF specimen impacted at 50 J: (a) 0-degree view and (b) 90-degree view

The 3D woven composite’s mechanics of impact energy dissipation are most plainly seen by observing them in an extreme state. Figures 3.18a and 3.18b show $\mu$CT images of the 3D woven specimen impacted at 90 J. Warp tows are firmly rooted in the far field of the specimen and span a long unsupported length due to the AITM 1-0010 or ASTM D7136 asymmetric fixture geometry. During impact, the warp tows straighten and elongate under a tensile-dominated load causing them to progressively de-bond from the surrounding matrix near the impact zone. The 3D ply-to-ply 60/40 reinforcement ratio denotes greater density of warp than weft tows. Since there were fewer weft tows and they were loaded over a shorter unsupported span, they encountered high shear forces and were more easily broken.

With increasing impact energy, additional rows of weft tows were broken. In the case of the 110-Joule 3D impact specimen, which was the highest energy level before perforation occurred, 12 rows of weft tows were ruptured along the specimen centerline. Figure 3.18a indicates that all weft tows through the thickness are broken. Figure 3.18b reveals the extensive warp tow debonding.
Figure 3.18: $\mu$CT images of 3D woven specimen impacted at 90 J: (a) warp view showing shear cracks and weft tow breakage and (b) weft view showing longitudinal cracking/warp tow debonding.

The 90-J $\mu$CT scans are also included in Figures 3.19a and 3.19b which reveal a significant volume of delamination with large permanent deformations.

Figure 3.19: $\mu$CT images of 2D NCF specimen impacted at 90 J: (a) 0-degree view and (b) 90-degree view.
3.4 Concluding remarks

The 3D woven composite exhibited greater damage resistance over the entire experimental range when compared with the baseline 2D composite using metrics of in-plane damage containment, indentation depth, and energy absorption per unit area. In the energy range 50–110 Joules, the 3D composite absorbed lower total energy.

These benefits were achieved by virtue of the through-thickness reinforcement which prevented delamination. Additionally, the woven architecture provided a greater resistance to energy absorption by allowing impact loads to be carried by tows, both before and after initial failure.

For 3D woven composites, low velocity impact damage is characterized by through-thickness cracking as opposed to planar delamination as seen in laminates. Despite this fact, Ultrasonic C-scanning has been observed to be an effective inspection method for detecting impact damage in 3D woven composites, indicating that inspection practices used on laminated composite structures would likely also be applicable to 3D woven structures.

The threshold for detectability (BVID) has been taken in industry as a critical dent-depth which is likely to be spotted during a general visual inspection (GVI); a well-established maintenance routine in the airline industry. The BVID threshold is between 0.5 \( mm \) and 1.3 \( mm \) depending on manufacturer [8–11]. If the BVID threshold is taken as 1 mm, as in the AITM 1-0010 test standard [14], the energy required to generate BVID is significantly increased from 48 J to 77 J by the use of a 3D-woven composite. If the BVID threshold is taken as 1.3 mm, the increase in required energy is even greater; from 48 J to 100 J. Consequently, the use of 3D woven composites in certain applications may reduce the likelihood of detectable damage and non-routine maintenance occurrence.
CHAPTER 4
EXPERIMENTAL CHARACTERIZATION OF DAMAGE TOLERANCE

4.1 Introduction

Quasi-static compression after impact (CAI) strength of structure containing BVID can be a critical factor in determining material design allowables for aerospace applications [11]. CAI strength of structure containing large visible impact damage (LVID) is necessary preliminary information to validate the slow-growth or no-growth damage tolerance approach for repeated loading [7]. This chapter serves to present experimental data on the mechanical behavior of the 3D woven composite under CAI loading as a function of incident impact energy. It will also compare the behavior to that of the “equivalent” NCF multidirectional laminate. Literature review and discussion presented in Chapter 1 identified the domain of impact threats and damage detectability criteria which are most relevant in the industry view of damage tolerance methodology. Findings from Chapter 3 helped to refine this domain by identifying impact energy levels at which to run CAI testing that would likely provide maximum exposure of the performance distinctions between the two material systems. Based on these findings, five specimens of each material system have been subjected to impacts of 30, 60, and 90 J prior to CAI testing. Each of the 24 screening specimens from Chapter 3 also underwent destructive testing by CAI.

Previous studies have compared CAI performance of 2D and 3D composites. Chiu [78] compared the damage resistance and tolerance of a ply-to-ply 3D woven composite with a 2D woven composite, concluding that 3D woven composites exhibited superior damage tolerance at the energy levels investigated (15 and 20 J). The 3D composite
retained 92% and 83% while the 2D woven composite retained 76% and 68% of undamaged strength for the 15 and 20 J energy levels, respectively.

Chen [79] tested a ply-to-ply 3D woven composite in addition to a prepreg laminate and a NCF laminate of similar layup. The 3D woven composite required lower impact energy to generate a 1 mm dent when compared with both the NCF and prepreg laminates, although damage area was consistently lower over the range of energies tested. CAI strength was reported to be considerably lower than the prepreg laminate, which was attributed to the superior in-plane properties of the laminate, which were not reported. Both the 3D woven composite and the NCF were found to maintain compressive strength near 100 MPa over their tested range, 20-45 J for the 3D woven and 20 - 70 J for the NCF, indicating excellent damage tolerance performance.

A comparative study concerning different 3D woven architectures [37] reported that the ply-to-ply architecture is superior to the orthogonal and angle interlock architectures in terms of retained CAI strength.

4.2 Experimental methods

4.2.1 Quasi-static compression-after-impact testing

Compression after impact testing was carried out according to AITM 1-0010 [14], testing each specimen in quasi-static compression to failure using a 250kN Instron servo-hydraulic testing machine with a constant load-head rate of 0.5 mm/min. A self-leveling upper platten (spherical radius approx. 100 mm) was employed to prevent bending moment from being transferred to the test specimen. Figure 4.1 shows the apparatus.

The compression tool was manufactured by Wyoming Test Fixtures (model WTF-CI(Airbus)) to the specifications furnished in the standard. The tool ensures that load is applied through the ends of the specimen and that buckling is avoided by the use of anti-buckling side guides. The primary benefit to the Airbus
Figure 4.1: Experimental setup for CAI testing

(AITM 1-0010) fixture over the Boeing (ASTM D7137) fixture is that it allows positive clamping pressure to be applied to the specimen through the side guides and end-clamps ensuring proper application of boundary conditions [66], completely eliminating out-of-plane displacement at the supported edges. Figure 4.2a shows the standard specification of boundary conditions. Figure 4.2b shows the compression tool used by this work. Care must be taken not to over-torque the tightening bolts for the knife-edge simple supports. The supports must be placed in contact with the specimen face but also must not carry compressive load and must allow vertical motion of the specimen. In this case, preliminary trials using DIC confirmed that specimen binding did not occur when tightening the non-lubricated bolts to “finger-snug.”

As per the AITM standard, verification of the compression tool was accomplished by testing an un-impacted specimen instrumented with strain gauges. Two gauges were applied to the front and two gauges were applied to the back face of the specimen
Figure 4.2: Compression tool illustration: (a) AITM 1-0010 specifications [14], (b) Wyoming Test Fixtures compression tool stock photo [15]

at the locations indicated by Figure 4.3. Verification criteria was agreement of all strain gauges within ±10% at a mean strain of 3000µε. The verification experiment
was repeated four times to test the setup sensitivity to misplacement of the compression tool and torque applied to the side-guide clamping bolts. In all cases the mean front-face and mean back-face strains were within 3% of the target $3000 \mu \varepsilon$. Additionally, strain read from each gauge was repeatable to within 5% of the four-trial mean for that gauge, indicating that the experimental setup was robust to small errors in fixture alignment and small variations in anti-buckling guide pressure.

Unfortunately, an error was made in the machining of this validation specimen with the result that two opposing corners on the load-bearing edges of the specimen were worn down and did not make contact with the compression tool. Due to this error, diagonal in-plane strain patterns were read in the strain gauge and DIC data making it impossible to validate the fixture against in-plane bias. Other preliminary trials using properly machined specimens instrumented with DIC revealed no indication of significant in-plane bias. Figure 4.4 shows the DIC strain map on the instrumented validation specimen, used to validate against out-of-plane bending. The diagonal in-plane strain pattern resulting from the mis-shapen specimen is clearly visible. Figure 4.5 shows one subsequent trial test with a much more symmetric strain profile.

Figure 4.5: Trial specimen CAI test to failure: (a) mean $3000 \mu \varepsilon$, (b) pre-failure, (c) post-failure
Figure 4.3: Strain gauge locations for compression tool validation [14]

Figure 4.4: Instrumented validation specimen and DIC strain map at mean $3000\mu\varepsilon$
4.2.2 Strain measurement technique

The CAI test is different from a typical characterization test in that there is a definitively non-uniform strain profile within the gauge section of the specimen. In this study, a strain value which was representative of the overall specimen displacement was desired. For this reason, an approach was implemented which used the relative displacement of three upper points with respect to three lower points, forming three vertical lines as indicated in Figure 4.6a. At each stage, the value of strain over each of the three lines, left, center, and right, were calculated with the elementary formula

$$\varepsilon = \frac{L + \Delta L}{L}.$$  

The reported strain values here are the average of the left, center, and right strains.

This measurement technique also allowed a more exacting view on the symmetry of loading. Figure 4.6b shows a typical stress-strain response for each of the three lines on a CAI specimen impacted at 60 J. As expected, the center strain is significantly higher at any given load due to the more compliant central damage region. The left-side strain values can also be observed to be greater than the right-side strain. This trend was systematic through all tests, the deviation increasing at higher strain levels and at higher impact levels. The error may stem from minor impact point misalignment relative to the specimen center. The strain deviation from right-to-left was quantified by comparing the strains at peak load. The left side peak strain was systematically higher than the right side peak strain by an average of 3.7% for 30-J tests, 4.4% for 60-J tests, and 6.3% for 90-J tests. These values are well below the 10% deviation limit provided in the compression tool validation process in AITM 1-0010, indicating that the asymmetry did not have a material effect on the strength results. In future work, however, efforts should be made to fine tune the apparatus using this metric as part of the preliminary testing phase.
Figure 4.6: Left-to-right strain deviation: (a) 3-line strain measurement configuration (b) Typical stress-strain responses, left, center, and right for 60-J 3D specimen

4.3 Results and discussion

The stress-strain responses obtained from the 30, 60, and 90 J CAI tests are shown in Figures 4.7a–4.7c, respectively. As is evident in these figures, the effect of increasing impact energy is to decrease ultimate strength and to increase ductility. The dashed lines follow the modulus obtained from the compressive tests of undamaged CAI specimens. The modulus for each specimen was computed based on a linear regression of the noisy raw data, and the reported modulus is the average based on five tested specimens. Since the undamaged coupons exhibited quasi-linear behavior (Figure 4.8), the linear representation in Figures 4.7a–4.7c is a very good approximation of the actual undamaged behavior in this strain range. To illustrate the effect of impact damage on initial stiffness, the tangent moduli evaluated at 0.025% strain are listed in Table 4.1 for comparison with the moduli obtained in the undamaged compression tests and ASTM D6641 compression tests. The moduli reported for both undamaged tests were computed in the range 0.1 to 0.3 % strain. Undamaged compressive
and residual compressive strength values are also listed. Residual strengths are also plotted in Figures 4.9 and 4.10.
Figure 4.7: Stress-strain response for CAI tests: (a) 30-J impact, (b) 60-J impact, and (c) 90-J impact
Figure 4.8: Stress-strain response for undamaged (AITM 1-0010) tests

Table 4.1: Summary of compressive properties after impact

<table>
<thead>
<tr>
<th>Material</th>
<th>Damage Level</th>
<th>Ultimate Strength</th>
<th>Modulus</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td>MPa (COV)</td>
<td>GPa (COV)</td>
</tr>
<tr>
<td>3D 60/40 ply to ply 24K/24K</td>
<td>0 J&lt;sub&gt;(ASTM D6641)&lt;/sub&gt;</td>
<td>399 (10.2%)</td>
<td>75.5 (5.04%)</td>
</tr>
<tr>
<td></td>
<td>0 J&lt;sub&gt;(AITM 1-0010)&lt;/sub&gt;</td>
<td>358 (3.68%)</td>
<td>74.8 (3.06%)</td>
</tr>
<tr>
<td></td>
<td>30 J</td>
<td>272 (5.71%)</td>
<td>69.1 (5.59%)</td>
</tr>
<tr>
<td></td>
<td>60 J</td>
<td>239 (4.22%)</td>
<td>71.7 (3.09%)</td>
</tr>
<tr>
<td></td>
<td>90 J</td>
<td>220 (2.95%)</td>
<td>69.0 (8.09%)</td>
</tr>
<tr>
<td>NCF 44/44/11 12K</td>
<td>0 J&lt;sub&gt;(ASTM D6641)&lt;/sub&gt;</td>
<td>328 (9.13%)</td>
<td>64.0 (2.03%)</td>
</tr>
<tr>
<td></td>
<td>0 J&lt;sub&gt;(AITM 1-0010)&lt;/sub&gt;</td>
<td>335 (3.72%)</td>
<td>65.0 (1.52%)</td>
</tr>
<tr>
<td></td>
<td>30 J</td>
<td>250 (3.60%)</td>
<td>63.5 (5.10%)</td>
</tr>
<tr>
<td></td>
<td>60 J</td>
<td>215 (3.57%)</td>
<td>65.1 (5.18%)</td>
</tr>
<tr>
<td></td>
<td>90 J</td>
<td>194 (2.61%)</td>
<td>59.4 (2.56%)</td>
</tr>
</tbody>
</table>

The mean residual strengths of the 30, 60, and 90-J CAI specimens are plotted in absolute terms against impact energy in Figure 4.9, along with the undamaged ASTM D6641 strengths. Error bars indicate one standard deviation. The undamaged test consisted of eight specimens while the CAI tests consisted of five specimens per configuration. The strength results for the individual screening specimens used in the Chapter 3 damage resistance evaluation are also shown. It should be noted here that, due to a change of instrumentation, the falling mass used in the Chapter 3 screening
impacts was 0.456 kg greater than the mass used in the chapter 4 damage tolerance impacts for any given energy level. For this reason, the screening specimen strengths were treated as a separate sample and not included in the scatter analysis in Figure 4.9. Impact parameters for each set of tests are given in Tables 3.1 and 4.2.

Table 4.2: Impact parameters used for damage tolerance experiments

<table>
<thead>
<tr>
<th>Impact Energy (J)</th>
<th>Falling Mass (kg)</th>
<th>Drop Height (mm)</th>
<th>Impact Velocity (m/s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>30</td>
<td>5.042</td>
<td>607</td>
<td>3.45</td>
</tr>
<tr>
<td>60</td>
<td>10.042</td>
<td>609</td>
<td>3.45</td>
</tr>
<tr>
<td>90</td>
<td>10.042</td>
<td>914</td>
<td>4.23</td>
</tr>
</tbody>
</table>

Scarponi [80] and other researchers have reported residual strengths relative to undamaged strengths using the ratio $\frac{CAI}{CBI}$. Following this example, the mean residual strengths of the 30, 60, and 90-J CAI specimens are plotted in relative terms against impact energy in Figure 4.10. Both material systems seem to approach a static strength reduction factor near 0.5 at perforation energy. The retained strength curves of the two material systems overlap significantly, with the first statistically significant difference occurring at the 90-J impact energy level. Previous studies using various material systems [8, 81, 82] have found that residual strength is negatively correlated with damage extent, particularly damage width, suggesting that the 3D woven composite, which exhibited superior damage resistance, should also display improved residual compressive strength. Given the improved crack arresting/damage localization of the 3D woven composite over the baseline that was discussed in Chapter 3, we would have expected to see a larger improvement in CAI performance, although the marginal improvement observed is in-keeping with expectations.
Figure 4.9: Residual strengths from quasi-static CAI tests
Figure 4.10: Residual strength fractions from quasi-static CAI tests
4.4 Conclusions and recommendations

Five specimens of each material system have been subjected to impacts of 30, 60, and 90 J prior to CAI testing, in addition to the 24 screening specimens from Chapter 3. Strain measurement was accomplished using DIC to track displacement of six discrete points. The 24 screening specimens provided residual strength data for impact energies up to and including perforation energy. Residual compressive strengths for both materials were observed in these trails to follow an exponential decay, reducing to a limit near 50% of undamaged strength. The 3D woven and 2D baseline composites were found to retain similar residual strength percentages. 30 and 60 J C.A.I. tests revealed no statistically significant difference between the performances of the two materials. The 90-J comparison showed a small but statistically significant performance difference at the 0.05 confidence level under a two-sample student’s T-test. The 3D woven composite retained an average of 61.5% of undamaged strength and the 2D NCF baseline composite retained an average of 57.8% of undamaged strength for the 90-J impact testing.

From these results we conclude that while the 3D woven composited eliminated the delamination failure mode and significantly decreased the extent of damage under impact, this did not appear to afford significant changes in compression after impact strength, except at very high energy levels near perforation where a small improvement was observed over the 2D baseline.
CHAPTER 5
CONCLUSIONS AND RECOMMENDATIONS

This thesis provided experimental data to address two broad research questions regarding the superior impact damage resistance and tolerance 3D composites when compared to 2D laminated composites. The first question involved damage resistance and the possible effect of reducing life cycle cost of an aircraft structure by reducing the frequency of component repair as a result of lower probability of detectable damage occurrence. The second question involved damage tolerance and the possible effect on life cycle cost by lengthening the necessary inspection intervals.

The preliminary literature review has discussed the industry and regulatory treatment of damage resistance and damage tolerance to provide a realistic framework for interpreting the experimental results.

The initial intention was to evaluate both materials at AC20-107B damage categories 1, 2, and 3, however literature review demonstrated that the higher level damage categories would not as useful for coupon-level testing because the coupon level testing is aimed at providing insight to designers in their damage tolerance assessments. Category 3 damage, especially, is classified as a definite repair scenario and is not of primary interest to the study. Furthermore, the definitions of the AC20-107B damage categories vary with structural applications, which are beyond the scope of the study. Screening impact tests were performed on both material systems between 20 and 130 Joules. The screening test spanned the range of likely in-service impact threats to aircraft presented in literature. Preliminary screening results indicated that 30, 60, and 90 Joules were appropriate impact levels to use for the damage tolerance assessment with the primary goal being to expose the performance differences between the two material systems.
The first experiments, presented in Chapter 2, characterized tension, compression, in-plane shear, and mode-1 fracture toughness behavior of the materials. The 2D NCF baseline composite and the 3D ply-to-ply woven composites share comparable strengths and stiffnesses in tension and compression when loaded on-axis. The 3D ply-to-ply woven composite lacks bias-direction reinforcement and correspondingly behaves in a ductile manner with low strength when subjected to off-axis or in-plane shear loading. Extra caution should be exercised when applying this 3D woven architecture in monocoque skin applications or other structural applications with large off-axis load components.

The experimental compliance “Berry” method of mode-I fracture toughness experiment is recommended for convenient testing and data reduction.

To evaluate the mode-I fracture toughness of the 3D woven ply-to-ply composite using the DCB test, it is necessary to increase the strength of cantilevered arms relative to the intended fracture plane. The current study attempted to accomplish this by reducing the area of the fracture plane up to 50%. This quantity proved insufficient, and similar studies using this technique should expect to reduce the gauge region width by greater than 50% in order to successfully propagate a crack without bending failure of the arms. Other studies have successfully used reinforcing tabs adhesively bonded to the specimen faces as described in [6, 42-46]. This method is recommended for future studies characterizing mode-1 fracture toughness of 3D woven composites.

Chapter 3 experiments evaluated damage resistance behavior of the two materials. The 3D woven composite exhibited greater damage resistance over the entire experimental range when compared with the baseline 2D composite using metrics of in-plane damage containment, indentation depth, and energy absorption per unit area. In the energy range 50–110 Joules, the 3D composite absorbed lower total energy.
These benefits were achieved by virtue of the through-thickness reinforcement which prevented delamination. Additionally, the woven architecture provided a greater resistance to energy absorption by allowing impact loads to be carried by tows, both before and after initial failure.

For 3D woven composites, low velocity impact damage is characterized by through-thickness cracking as opposed to planar delamination as seen in laminates. Despite this fact, Ultrasonic C-scanning has been observed to be an effective inspection method for detecting impact damage in 3D woven composites, indicating that inspection practices used on laminated composite structures would likely also be applicable to 3D woven structures.

The threshold for detectability (BVID) has been taken in industry as a critical dent-depth which is likely to be spotted during a general visual inspection (GVI); a well-established maintenance routine in the airline industry. The BVID threshold is between 0.5 mm and 1.3 mm depending on manufacturer [8-11]. If the BVID threshold is taken as 1 mm, as in the AITM 1-0010 test standard [14], the energy required to generate BVID is significantly increased from 48 J to 77 J by the use of a 3D-woven composite. If the BVID threshold is taken as 1.3 mm, the increase in required energy is even greater; from 48 J to 100 J. Consequently, the use of 3D woven composites in certain applications may reduce the likelihood of detectable damage and non-routine maintenance occurrence.

Chapter 4 experiments evaluated damage tolerance behavior of the two materials. Five specimens of each material system have been subjected to impacts of 30, 60, and 90 J prior to CAI testing, in addition to the 24 screening specimens from Chapter 3. Strain measurement was accomplished using DIC to track displacement of six discrete points. The 24 screening specimens provided residual strength data for impact energies up to and including perforation energy. Residual compressive strengths for both materials were observed in these trails to follow an exponential
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From these results we conclude that while the 3D woven composites eliminated the delamination failure mode and significantly decreased the extent of damage under impact, this did not appear to afford significant changes in compression after impact strength, except at very high energy levels near perforation where a small improvement was observed over the 2D baseline.
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