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Review of Damage Tolerance for Composite Sandwich Airframe Structures

August 1999

Final Report

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16. Abstract

The use of composite sandwich construction is rapidly increasing in current and future airframe designs especially for general aviation aircraft and rotorcraft. Typically, sandwich constructions for these applications use thin-gage composite facesheets (0.020″ to 0.045″) which are cocured to honeycomb and foam cores. Due to the nature of these structures, damage tolerance is more complex than conventional laminated structures. Besides typical damage concerns such as through penetration and delamination, additional modes including core crushing and facesheet debonding must also be addressed. This complicates the certification process by introducing undefined Allowable Damage Limits (ADL) and Critical Damage Thresholds (CDT) as related to the ultimate and limit load carrying capability of the structure. This document provides a background review of previous damage tolerance investigations including an overview of traditional metallic damage tolerance methodologies. Illustrative summaries are presented which show the scope of previous investigation parameters such as impact energy, facesheet thickness, and core thickness of typical sandwich constructions. Also included is a compilation of damage tolerance certification procedures and regulations taken from FAR Part 23-29 for composite damage tolerance as well as recommendations from associated Advisory Circulars. Past and current airframe industry sandwich constructions which show the scope of current and future sandwich designs were also surveyed. In conclusion, a proposed future research approach and its methodology are presented which should aid in establishing certification guidelines and confidence involving the damage tolerance of sandwich constructions as they apply to general aviation aircraft and rotorcraft.

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EXECUTIVE SUMMARY

The use of composite sandwich construction is rapidly increasing in current and future airframe designs especially for general aviation aircraft and rotorcraft. Typically, sandwich constructions in these applications use thin-gage composite facesheets (0.020″ to 0.045″) which are cocured to honeycomb and foam cores. Due to the nature of these structures, damage tolerance is more complex than conventional laminated structures. Besides typical damage concerns such as through penetration and delamination, additional modes including core crushing and facesheet debonding must also be addressed. This complicates the certification process by introducing undefined Allowable Damage Limits (ADL) and Critical Damage Thresholds (CDT) as related to the ultimate and limit load-carrying capability of the structure.

This document provides a background review of previous damage tolerance investigations including an overview of traditional metallic damage tolerance methodologies. Illustrative summaries are presented which show the scope of previous investigation parameters such as impact energy, facesheet thickness, and core thickness of typical sandwich constructions. Also included is a compilation of damage tolerance certification procedures and regulations taken from FAR Part 23-29 for composite damage tolerance as well as recommendations from associated Advisory Circulars.

Past and current airframe industry sandwich constructions which show the scope of current and future sandwich designs were also surveyed. In conclusion, a proposed future research approach and its methodology are presented which should aid in establishing certification guidelines and confidence involving the damage tolerance of sandwich constructions, particularly as they apply to general aviation aircraft and rotorcraft.

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1. INTRODUCTION.

Sandwich structures provide an efficient method to increase bending rigidity without a significant increase in structural weight. Thin-gage facesheets (0.020" to 0.045") are cocured or bonded to honeycomb (aluminum or Nomex) or syntactic foam cores. Thus, structures, based upon a minimum gage thickness adequate to carry the in-plane loads, can be made to carry out-of-plane loads and to be stable under compression without a significant weight penalty. The potential of sandwich structures is substantial. Helicopter blades, optical benches for space applications, and nonferrous ship hulls are some of the current applications. In general aviation, skin/stiffener structures can be replaced with sandwich structures. Design is based on several loading regimes including pressurization, gust, and landing loads. Sandwich construction figures prominently in future aerospace applications such as Raytheon's Premier I, Lockheed-Martin's X-33, and future tilt rotors by Boeing Defense & Space Group, Helicopters Division.

Damage tolerance of such sandwich structures is substantially more complex than conventional laminated structures. Besides typical damage concerns such as through penetration and delamination, additional modes including core crushing and facesheet debonding must be addressed. Often damage may not be characterized as uniform through the thickness. An impact may penetrate or damage only one facesheet while the other remains intact. Manufacturing flaws or in-service loads will also result in an unsymmetrical damage state. Cores tend to absorb and retain water which often reduces mechanical properties as well as increasing the structural weight.

To fully realize the weight-saving potential of sandwich structures, one must first understand the damage tolerance of such structures. This is essential in the design process to develop more efficient structures and to reduce the extent and frequency of in-service repair.

Composite facesheets fail as a result of an interaction among matrix cracks, fiber fracture, fiber kinking, and delamination. Sandwich structures also exhibit core crushing and facesheet debonding. Damage may result from low-velocity impact such as tool drops or high-energy events such as ballistic penetration. In some instances, penetration may not be complete and only one facesheet may be damaged significantly causing a redistribution of stresses in the plane of the damaged facesheet. Damage inspections are more difficult because the core can mask the damage or otherwise impede the nondestructive technique.

Current research can be divided into two major areas: damage resistance and damage tolerance. Damage resistance is concerned with the creation of damage due to a specific impact event. Here the variables include the material and layup of the facesheet, the type and thickness of the core material, and the boundary conditions of the sandwich structure. Damage tolerance is concerned with the structural response and integrity associated with a given damage state of a structure. Here the variables include the type, extent, and location of the damage. For plain laminates, the damage due to impact is contained in the laminate itself. The nature of sandwich construction increases the complexity of the problem. The presence of two load paths separated by a core that is responsible for shear load transfer combined with unsymmetrical damage requires a better

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understanding of damage progression and residual strength. This is not to say that for built-up curved stiffened panels the damage progression is not as complex.

For aircraft design, the technical challenge is to adequately predict the residual strength of a damaged composite sandwich structure. Discussions with airframe manufacturers lead one to believe there is a general lack of understanding of the failure mechanisms and damage tolerance of sandwich structures. The structural response of sandwich structures is determined empirically based upon component and full-size test articles. This approach may not allow for adequate design tradeoffs in a timely manner and can result in significant rework if problems materialize during full-scale testing.

The lack of understanding is partly due to the nonlinear behavior exhibited by composite sandwich structures. Out-of-plane deflections on the order of the facesheet thickness (rather than the sandwich thickness) can result in nonlinear response. Structural features such as closeouts and tapered sections to accommodate fastening and unsymmetrical damage result in out-of-plane deformations and additional sources of nonlinearities. Thus, linear models cannot adequately predict the structural response let alone damage progression and residual strength. Without adequate models, the ability to predict damage onset due to overloading or growth of existing damage is nearly impossible. Design methodology is thus based on limited coupon tests and full size test articles to either substantiate the design or indicate hot spots that need to be reworked.

Understanding the damage tolerance of composite sandwich structures will improve the structural efficiency of the aircraft. Moreover, the accurate assessment of damaged structures will reduce the frequency and cost of repair and thus improve the serviceability of the aircraft.

2. DAMAGE TOLERANCE OF METAL AIRCRAFT.

The purpose of this section of the report is to briefly review the general concepts associated with the damage tolerance evaluation of metal airframes and to use this review to extract some initial ideas that will help lead to developing damage tolerance guidelines for composite sandwich construction airframes. There is no intent to copy the metal damage tolerance philosophy for composites, but to make use of the experience and technology associated with metal aircraft to provide a head start in developing a damage tolerance philosophy for composites. For this section of the report, liberal use has been made of Swift [1] which contains an excellent review of damage tolerance practices for metallic airframes.

2.1 PURPOSE OF DAMAGE TOLERANCE.

The purpose of damage tolerance for metal airframes is to provide an inspection plan for each principal structural element so that cracking (initiated by fatigue, accident, or corrosion) will never propagate to failure prior to detection. This applies to all principal structural elements except those for which the manufacturer can show that damage tolerance is impractical, in which case a safe-life evaluation must be made using appropriate scatter factors.

The crack propagation life must be determined for each principal element under the spectrum of stresses expected in service, taking into account the environment to which the element is

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subjected in service. A threshold for inspecting each principal structural element must be established together with recurring frequency of inspection based on growth from maximum crack size, which can remain undetected after an inspection. Inspection frequencies must be determined based on crack growth life from a detectable length to a critical length at limit load.

2.2 PRINCIPAL STRUCTURAL ELEMENTS.

A principal structural element is one whose failure, if undetected, would lead to catastrophic failure (loss of aircraft). In general, there may be approximately 100 such elements for the entire airframe. The number of locations varies depending on the particular aircraft, but is not necessarily a function of aircraft size. The following conditions are examples of those used to identify principal structural locations: high tensile loading, low margins of safety, high stress concentration, high spectrum severity, high crack growth rates, low fracture toughness, multiple site damage, high load transfer, locations with high stress in secondary members after primary member fails, locations prone to accidental damage, and locations shown from full-scale fatigue testing to have problems.

2.3 DAMAGE TOLERANCE EVALUATION TASKS.

The major tasks associated with damage tolerance evaluation prior to certification are as follow:

- Select the principal structural elements to be evaluated.
- Develop the stress spectrum for each location.
- Define the environment for each location.
- Develop crack growth rate data for each location.
- Determine fracture toughness for each location.
- Determine the structural category for each location.
- Determine critical damage size under limit load and in-service load for each location.
- Develop crack growth curves from spectrum loading for each location.
- Establish initial damage size for each location.
- Establish damage threshold of detectability for each location.
- Validate residual strength and crack growth analysis methods with testing.
- Meet with operators, manufacturers, and certifying agencies collectively to approve plans.

Since there is a large number of locations on each airframe to be evaluated, it is impractical to accomplish a damage tolerance evaluation for each location with testing. Therefore, reliance must be placed on analysis. The analysis procedures associated with damage tolerance evaluation of metallic aircraft are based on stress analysis oriented fracture mechanics including stress intensity, fracture toughness, and crack propagation growth rate. These, in turn, are used in (1) residual strength calculations and (2) fatigue crack propagation analyses under spectrum loading, which are two major ingredients in damage tolerance evaluation. However, enough testing must be done to validate analysis methods, primarily to ensure that they do not produce unconservative results. Full-scale testing is too late to be useful in the design phase; however, it can be used later to update initial damage tolerance evaluations.

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2.4 STRUCTURAL CATAGORIES.

Every metal airframe has a variety of structural element categories. Several of the most common of these are listed and briefly described below.

2.4.1 Single Load Path - Safe Life.

This category is only allowed if the manufacturer can show that damage tolerance is impractical. Typical examples are landing gears and engine mounts. For this type of structural element, the critical crack size at limit load is smaller than the detectable crack size (see figure 1). Therefore, there is no safe crack growth period for inspection. The time (flights) to reach critical damage size must well exceed the life of the airframe

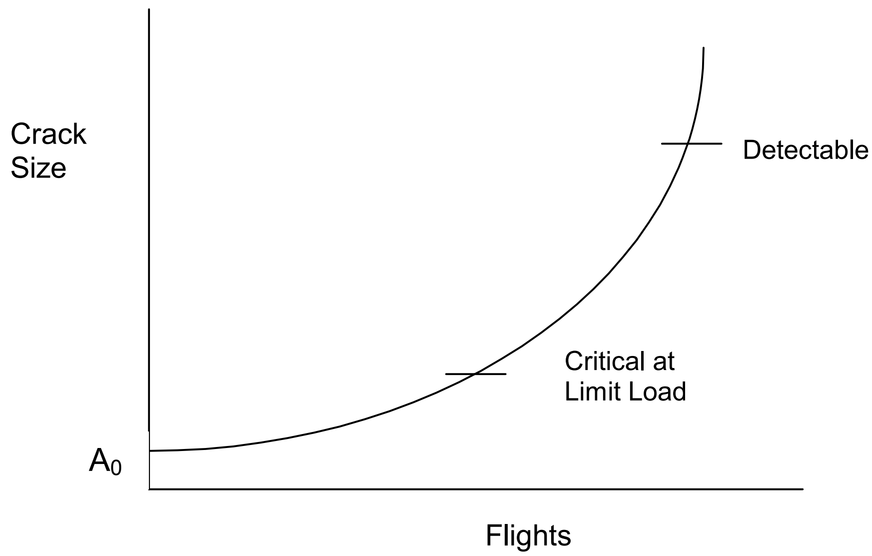

FIGURE 1. SINGLE LOAD PATH - SAFE LIFE

2.4.2 Single Load Path - Damage Tolerant.

This type of structure (single load path) is allowed but not recommended. For this type of structure, the detectable crack size is significantly less than the critical size at limit load (see figure 2). Therefore, there is a safe crack growth period for which adequate inspection can take place to detect the crack before it becomes critical and causes catastrophic failure. This type of structure would be inappropriate for brittle materials with low fracture toughness, since the critical crack size would be too small to allow an adequate safe crack growth interval for inspection.

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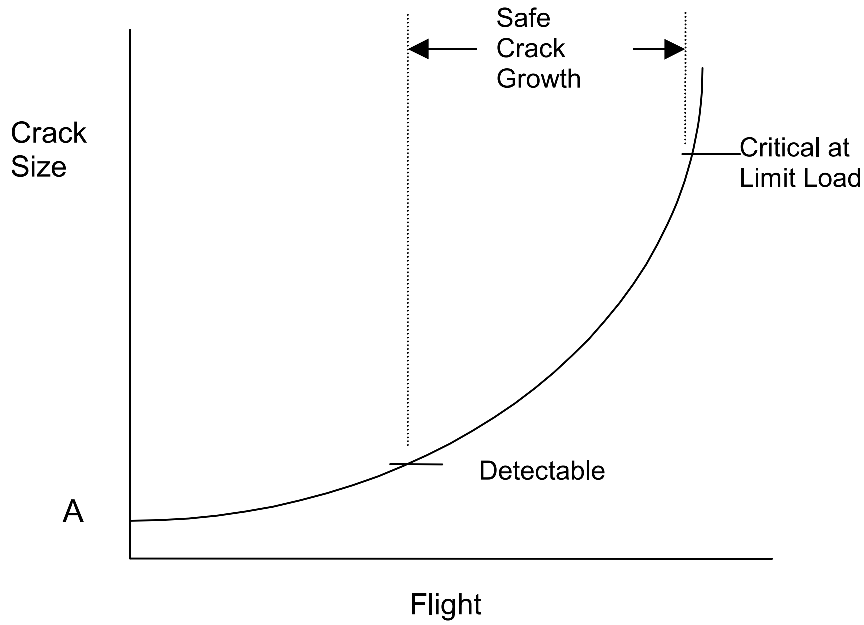

FIGURE 2. SINGLE LOAD PATH - DAMAGE TOLERANT

2.4.3 Multiple Load Path - Externally Inspectable.

This is the category that is used for most basic structure in metallic aircraft. Initial cracks are assumed to exist in both the primary member and the secondary member. The primary member is hidden (not externally inspectable). After the primary member fails during service, additional load is transferred to the secondary member, which is externally inspectable (see figure 3). The secondary member eventually fails when its crack reaches critical size at limit load. The detectable crack size in the secondary member is identified, and the safe crack growth period for inspection is between detectable and critical, as shown.

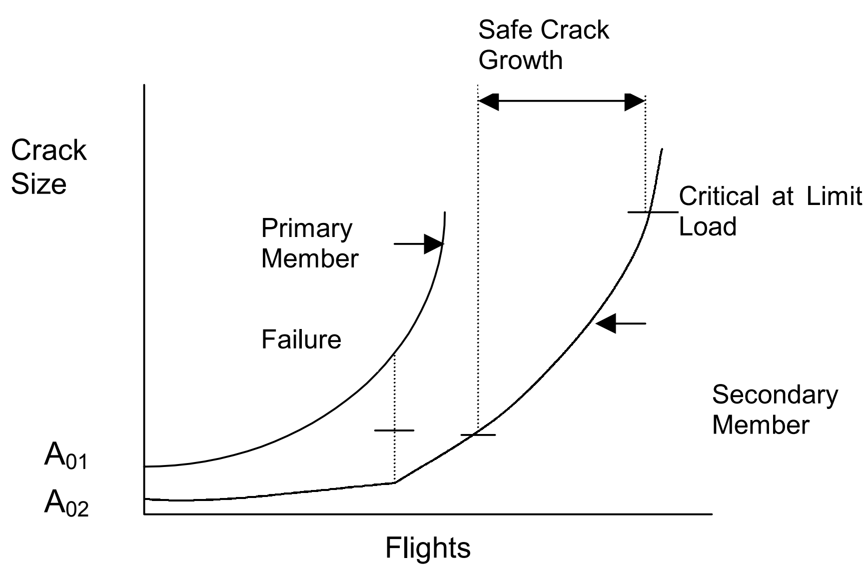

FIGURE 3. MULTIPLE LOAD PATH - EXTERNALLY INSPECTABLE

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2.4.4 Multiple Load Path - Inspectable Prior to Load Path Failure.

For this category of structure, the primary member is inspectable before failure. Therefore, the detectable crack size in the primary member is identified, and the safe crack period for inspection is between the detectable crack in the primary member and the critical crack in the secondary member (see figure 4).

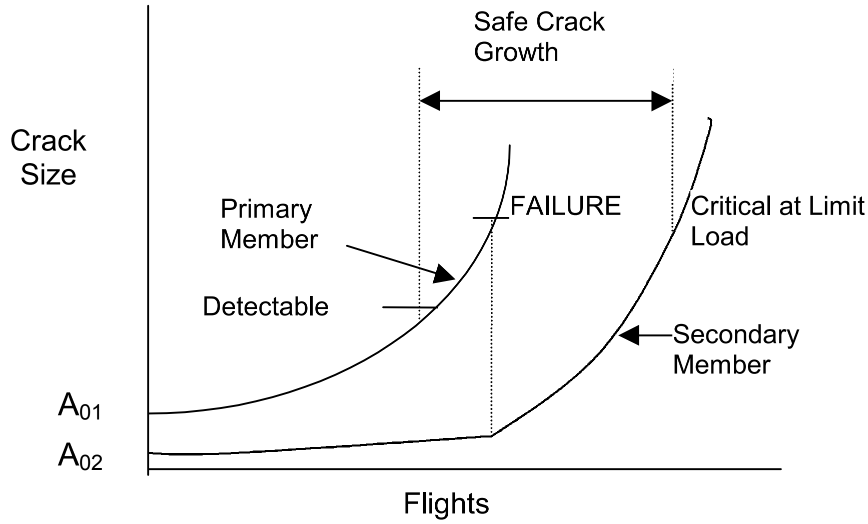

FIGURE 4. MULTIPLE LOAD PATH - INSPECTABLE PRIOR TO LOAD PATH FAILURE

Damage tolerant structural elements, such as those described in figures 2, 3, and 4, have the following in common: (1) the existence of initial cracks, (2) the establishment of threshold values for detecting cracks based on a specified inspection methodology, (3) critical crack sizes developed from static (residual strength) tests based on limit load, (4) crack propagation curves from spectrum loading expected in service, and (5) safe crack growth intervals for inspections to detect cracking.

2.5 ISSUES/CONCERNS ASSOCIATED WITH DAMAGE TOLERANCE OF COMPOSITE STRUCTURES.

2.5.1 Principal Structural Elements.

Criteria must be developed to identify principal structural elements for damage tolerance evaluation. For a monocoque sandwich fuselage, there may be many fewer principal structural elements than for metallic airframes which may undergo a variety of loading types.

2.5.2 Structural Categories.

The different types of composite structure must be categorized, and each principal structural element must be identified with a particular category to enable its damage tolerance program to be developed. For a monocoque sandwich fuselage, there may be fewer categories of structure than for metallic airframes.

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2.5.3 Damage.

Damage from manufacturing processes, accidental damage from assembly and handling, and accidental damage from in-service operation must be defined. From each of these sources, the modes (porosity, delamination, puncture, core crush, etc.) of damage must be identified. For each mode of damage, criteria must be established for determining characteristics such as initial size, threshold of detectability, critical size, and whether growth during spectrum loading will be produced.

2.5.4 Important Loading Modes.

The most critical mode of loading for metallic structure is tension, which causes crack opening. The critical modes of loading for composite structure must be identified for purposes of selecting principal structural elements and for developing appropriate testing requirements.

2.5.5 Testing.

Requirements for static testing of structural elements with various modes of damage and various modes of loading should be defined to provide residual strength information. Requirements for cyclic testing of structural elements with various modes of damage and various modes of loading should be defined to provide damage growth (or no growth) information. Requirements for cyclic testing of important full-scale components should be defined to verify that damage would not grow to a critical size in the full-scale article during the life of the airframe. Requirements for the static testing of important full-scale components should be defined to verify that they would carry required loads with any damage growth inflicted from cyclic loading. In addition, appropriate testing must be identified to validate any analyses methods used to demonstrate certification requirements.

3. PREVIOUS INVESTIGATIONS INTO DAMAGE TOLERANCE.

3.1 RECENT ADVANCES IN SANDWICH COMPOSITES.

Sandwich construction composites are used in a wide variety of structural applications largely because of their relative advantages over other structural materials in terms of improved stability and weight savings as well as a number or other factors cited previously. While the initial design of structures comprised of sandwich construction composites is at a fairly mature stage of development [2, 3], less progress has been made in understanding the long term response of such structures subjected to adverse in-service impact events or environmental influences. Such an understanding is critical if there is to be widespread use of sandwich composites in applications where structural durability and damage tolerance is a primary consideration (e.g., aerospace or automotive applications). The focus of this section is to provide a broad overview of recent research aimed at characterizing the thermomechanical response of various sandwich construction composites subjected to low-to-moderate energy impact events as well as environmental influences. Such impacts may induce *localized* damage in sandwich composites (fiber breaks, resin cracking, face sheet-core delamination, core crush, puncture, etc.) and can be attributable to a number of fairly common discrete sources (hail, tool drops, runway projectiles,

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bird strikes, or other unintentional impacts). Any reference to impact damage used herein will suggest the damage associated with low-to-moderate energy impact events, unless stated otherwise. The effect of intrinsic processing induced defects (porosity, voids, small disbonds, etc.) and catastrophic damage associated with high-energy impacts resulting from airplane crashes or similar events are not considered in this discussion. Widespread processing induced defects, perhaps, may best be addressed using continuum damage mechanics [4]; Abbott [5] noted that evolution of processing induced defects is likely not an issue for many structural applications involving sandwich composites.

Recent efforts aimed at clarifying the thermomechanical response of sandwich composites subjected to low-to-moderate energy impacts can be loosely categorized into three areas:

a. Analysis of the impact dynamics between the indentor (projectile) and the target (sandwich panel).

b. Characterization of impact damage modes and mechanisms.

c. Application of durability and damage tolerance principles to sandwich composites.

A summary of key research in each of these areas is presented in the following discussion. See Abrate [6] for an excellent review of recent investigations concerning the effect of impact on sandwich structures with laminated facings.

3.1.1 Analysis of Impact Dynamics for Sandwich Laminates.

Foreign object impact damage to composite structures can result in drastic reductions in composite strength, elastic moduli, and durability and damage tolerance characteristics. The problem of impact damage in composite laminates has received the bulk of the treatment in the literature [7, 8]; an understanding of the effect of impact damage on the mechanical properties and residual strength of sandwich composites requires further development. Indeed, it may be asserted that the issue of foreign object damage to sandwich composites is somewhat distinct from that of laminated composites; Abrate [6] noted that the impact response of sandwich composites is largely dominated by the core material. The contact laws defining the contact force versus indentation relationship during impact are significantly different for laminated and sandwich composites. Here, the indentation is defined as the relative displacement between the indentor (projectile) and target (composite panel).

During impact of a projectile and a target, the target experiences overall structural deformation as well as localized deformation in the vicinity of the point of impact. Classical beam, plate, and shell theories generally do not account for the local deformation near the impact site. The localized deformation in a sandwich panel may be estimated from large-scale numerical analysis of the dynamic contact problem [9] or through the introduction of a contact law that accounts for the effect of inelastic deformation and damage induced in the impact zone. The contact law is typically a nonlinear function displaying different loading and unloading characteristics and generally requires experimental verification [6].

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Sun and Wu [10] presented experimental results for low-velocity indentation of sandwich panels, constructed of AS4/3501-6 graphite/epoxy facesheets with either aluminum honeycomb or Rohacell foam cores, impacted by cylindrical and spherical indentors. Bilinear and linear-quadratic expressions were introduced to describe the loading portion of the contact law for panels constructed with honeycomb and foam cores, respectively. Power-law expressions were introduced to characterize the unloading portion of the contact law. For both types of core materials, the assumed contact laws provided a fairly accurate fit of the experimental data, although it must be emphasized that the contact force-indentation relationship was a strong function of the indentor shape used in the impact test. This latter observation is fairly typical of results reported throughout the literature [6]. It should be noted that the form of the contact laws for sandwich composites [10] is fundamentally different from that of the widely accepted contact relationship of power-law type for *laminated* composites developed by Yang and Sun [11]. Even if a power-law type contact force relationship is assumed for sandwich composites with a given facesheet configuration, the empirically determined power law coefficients and exponents, in general, will vary significantly from those of a composite laminate with similar layup [12, 13, 6]. Abrate [6] noted that the mechanical behavior of the core material must be well characterized in order to predict the contact law as well as the extent of damage.

In most applications the core is constructed from either foam or honeycomb material, although corrugated cores and balsa cores are not uncommon. The compressive stress-strain behavior of foam cores can be idealized as consisting of (1) a linear portion, (2) a plateau region corresponding to progressive crushing at nearly constant stress level, and (3) when densification is complete a region where the stresses increase rapidly with further deformation as indicated in figure 5 [6]. Of course, there are many factors that influence the material response of foam cores including temperature, strain rate, porosity, and binder content. Abrate [6] summarized a number of recent works aimed at establishing constitutive equations for various foam core materials that account for such influences. Hiel et al. [14] noted that foam core damage in sandwich composites may be sometimes limited to a small semihemispherical region near the top face sheet. Hence, a three-dimensional failure criterion is necessary in order to assess the likelihood of foam failure for a given impact event. Ideally, such a criterion should account for large deformations since 70% compaction is not unusual for both foam and honeycomb cores [6].

Honeycomb cores are also frequently used in sandwich panel construction. Similar to the preceding case, the idealized compressive stress-strain curve for honeycomb cores can be divided into three distinct regions: (1) a linearly elastic region, (2) a region corresponding to progressive crushing at nearly constant stress level (small oscillations may occur about the mean stress value due to consecutive local buckling), and (3) a region of rapidly increasing stresses with further deformation [6]. For low-density honeycombs, the linearly elastic region ends when the honeycomb cell walls buckle. The initiation of elastic buckling does not cause a complete loss of stiffness and a smooth transition to the post-buckled regime is generally observed. For intermediate- and high-density honeycombs, *fracture* of the cell walls is generally observed and a sudden drop from a collapse stress to a steady crush strength occurs as shown in figure 6 [15, 6].

Williamson and Lagace [16] suggested that impacted sandwich panel specimens that are either supported along their edges or are continuously supported by a rigid platen along the backface of

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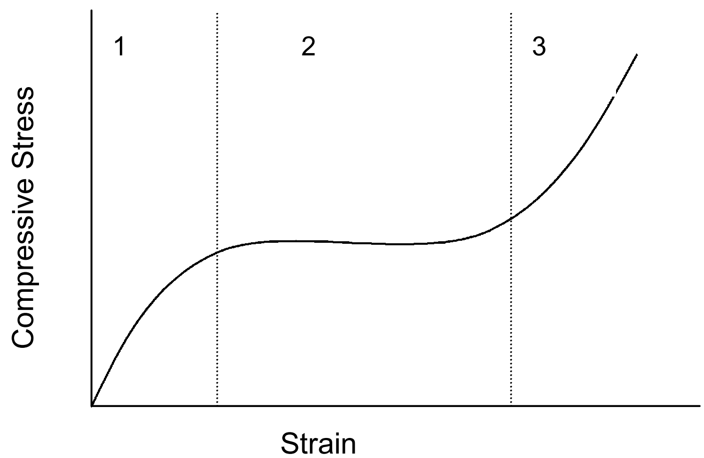

FIGURE 5. IDEALIZED COMPRESSIVE STRESS-STRAIN CURVE FOR FOAM CORE

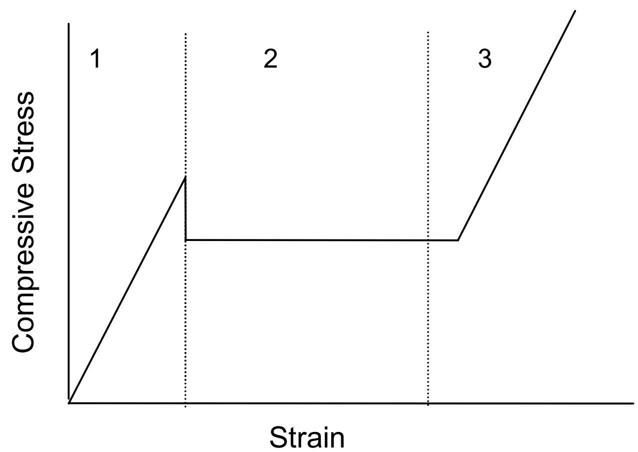

FIGURE 6. IDEALIZED COMPRESSIVE STRESS-STRAIN CURVE FOR HONEYCOMB CORE

the panel produce similar indentation laws. Thus, sandwich panels may be idealized mathematically as either beams or plates on inelastic foundations in order to derive expressions defining the contact law for a given indentor [12, 13, 6]. Once the contact law and the constitutive behavior of the core material have been characterized, it may be desirable to model the overall dynamic response of the impacted sandwich panels in order to establish the contact force history. In general, solution of the dynamic impact problem for sandwich panels requires use of the finite element method or other approximate technique since relatively few analytic solutions exist. See Abrate [6] for an excellent discussion on characterization of the dynamic impact response of sandwich panels using various two-dimensional spring-mass models for the case where the sandwich panel behaves quasi-statically as well as more refined models predicated on higher-order plate theory. Naturally, accounting for localized deformation and core material nonlinearity as well as selection of appropriate boundary conditions are key issues in adopting any of these approaches.

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In the preceding discussion, it was assumed that impact damage was primarily limited to the core material and upper facesheet. In the presence of large-scale damage involving both upper and lower facesheets and/or projectile penetration, the aforementioned analytic techniques either cease to be valid or require substantial modification. However, the result of such an event is a through-thickness hole with relatively less damage in the vicinity.

3.1.2 Impact Damage Modes and Mechanisms.

In this section, the failure modes involved in impact damage of sandwich composites will be described and the influence of several important factors on damage initiation and development will be discussed. Foreign object impact damage on sandwich structures can result in localized damage to the facings, core, and core-facing interface. The nature of the damage induced depends on a multitude of factors including the facesheet layup configuration and thickness, core material and thickness, interface properties between facesheet and core, fabrication techniques, impact velocity and energy, indentor shape, temperature, boundary conditions, and environmental factors. Damage initiation thresholds as well as the damage size depend on the properties of the core material and the relationship between the properties of the core material to those of the facings. Abrate [6] provided an excellent review of the failure mechanisms associated with impact to sandwich composites comprised of honeycomb, foam, balsa, corrugated, as well as *interleaved* cores. This discussion will focus on sandwich structures comprised of honeycomb and foam cores because of their widespread use in aerospace applications (see reference 6 for an overview of damage associated with balsa and corrugated cores as well as interleaved construction). Note that a variety of experimental techniques including ultrasonic inspection, X-rays, micrography, thermography, shearography, and the deply technique may be employed to characterize impact damage in sandwich structures.

A number of studies have been conducted to examine the effect of low-velocity impact damage on sandwich beams and plates comprised of carbon-epoxy facings and honeycomb cores; a summary of key results will be presented here (see Abrate [6] for an exhaustive list of references associated with these efforts). For such impacts, damage is typically confined to the top facing, the core-top facing interface, and the core. The lower facing remains generally undamaged. Damage is comprised of five different failure modes [17, 6]: (1) core buckling, (2) delamination in the impacted face sheet, (3) core cracking, (4) matrix cracking, and (5) fiber breakage in the facings. Damage in the upper facing consists primarily of matrix cracks and some fiber failures, similar to that observed in laminates [18]. Numerous researchers have reported that skin damage in sandwich structures increases almost linearly with impact energy until a maximum value is reached [6]. At this saturation damage state associated with low velocity impact, skin damage may become visible and the core-facing delamination size remains constant [19, 20, 6]; ostensibly, this may also denote the onset of barely visible impact damage (BVID). Damage in the honeycomb core material consists of crushing or buckling of cell walls in a region surrounding the impact point. Composite honeycomb cores (Nomex, glass thermoplastic, glass-phenolic, etc.) generally have relatively thick cell walls and typically fail due to cell wall fracture. The failure mechanisms associated with aluminum honeycomb cores were discussed earlier.

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For sandwich panels with foam cores, the core-facing interface may debond in a region surrounding the point of impact and the core may experience permanent deformation. Crack-like core defects may result from low-energy impacts whereas regions of compressed core can result from high-energy impacts [21, 6]. Foam core damage reaches a maximum value as the impact energy increases [17, 20, 6]. Bernard and Lagace [21] noted that the amount of interfacial debonding varies depending on the type of core material. For a given impact event, the debond size for a typical foam core may be comparable to that of a high-density aluminum honeycomb core and one order of magnitude smaller than that for a Nomex honeycomb core (the debond size may be virtually nonexistent for Rohacell foam cores). Increasing the number of facing plies may result in more core damage [19]. Compression after impact tests, however, indicated that cores providing more face sheet support had higher compressive residual strength even though delamination areas were larger. Hence, an understanding of the core-facing interaction is essential in order to discern the effect of impact damage on the subsequent thermomechanical response of sandwich composites.

Abrate [6] noted that a majority of experimental studies found in the literature considered relatively few sandwich configurations and examined the effect of projectile parameters (mass, shape, impact energy, etc.) on the impact damage induced. The results from many of these studies often were in conflict with each other. Consideration of a small number of panel configurations, however, does not allow for adequate treatment of the complex coupling between facing and core that largely dictates the impact damage development for a given loading.

Triantafillou and Gibson [22] studied the failure of foam core sandwich panels with aluminum facings subjected to three-point bending without impact. The study suggested that the dominant failure mode was highly material and configuration dependent. Seven failure modes in a foam core sandwich beam with a face that yields were identified: (1) face yielding, (2) face wrinkling/buckling, (3) core shear, (4) core tensile yield, (5) core compressive yield, (6) core indentation, and (7) face/core debonding. Expressions were derived for the first six failure modes in terms of the relative core density, ρ_c/ρ_s, and the ratio of the facing thickness to span length t/L. Here ρ_c is the density of the core material, and ρ_s is the density of the foam solid cell wall material. The study suggested that when both ρ_c/ρ_s and t/L are small, face wrinkling was the primary failure mode. When ρ_c/ρ_s is small and t/L is large, then core shear was the primary failure mode. Face yielding was observed for large values of ρ_c/ρ_s. Hence, the *fundamental failure modes for sandwich composites are configuration dependent for the undamaged case*. While the given material system may be fairly atypical, this study indicates that the influence of impact damage on the fundamental failure modes and load for a given sandwich configuration remains to be fully explored.

In a similar study, Mines et al. [23] examined the static and impact behavior of polymer composite sandwich beams comprised of several types of composite skins (chopped strand mat (CSM) glass fibers, woven fabrics of glass, carbon, and aramid fibers in either an epoxy or polyester resin) and core materials (Coremat™ resin impregnated nonwoven polyester and aluminum honeycomb). The study identified four types of failure modes (cf. figure 7): (I) upper skin compression failure followed by either stable core crushing then lower skin tensile failure or core shear failure, (II) upper skin crushing failure, (III) core shear failure, and (IV) lower skin

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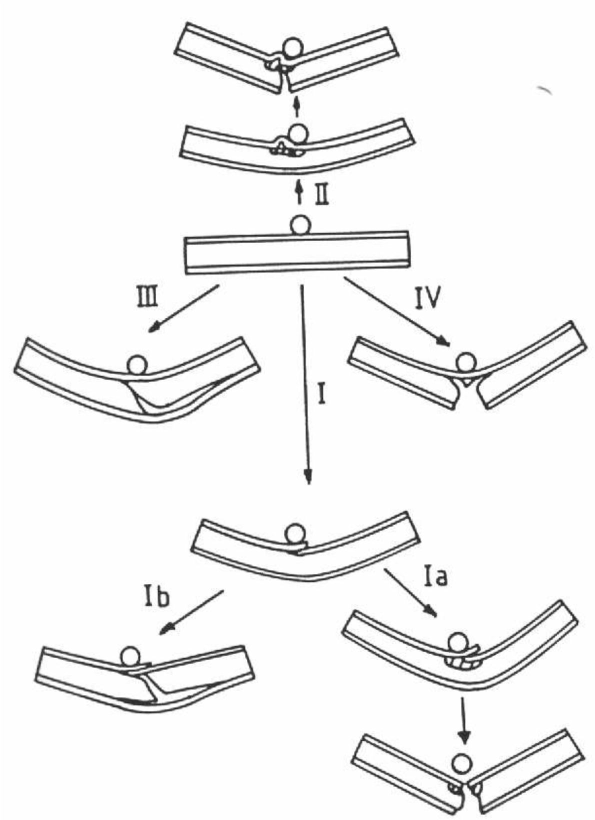

FIGURE 7. SANDWICH STRUCTURE FAILURE MODES FROM MINES ET AL. (1994). (I) UPPER SKIN COMPRESSION FAILURE, (II) UPPER SKIN CRUSHING FAILURE, (III) CORE SHEAR FAILURE, AND (IV) LOWER SKIN TENSILE FAILURE.

tensile failure. The specimens with skins comprised of woven carbon, woven glass, and CSM exhibited mode I failure; the skin compressive strength was lower than the tensile strength for these cases (the upper skin experienced true compressive failure rather than wrinkling). Specimens comprised of woven aramid skins and Coremat™ core exhibited mode II failure, likely a result of the low compressive strength of aramid. CSM skin/Coremat™ specimens exhibited mode IV failure because the tensile strength of the CSM is less than that in compression. Finally, specimens comprised of a combination woven/CSM glass skin and Coremat™ core exhibited both mode I and mode III failures. The latter failure mode resulted from reduced core/facing bondline toughness for this configuration. Llorente et al. [24] demonstrated that face wrinkling was the mode of failure for both undamaged and damaged thin-gauge sandwich panels being considered for use in Boeing helicopter primary fuselage structure.

The preceding studies are fairly typical and illustrate that both the fundamental failure modes and impact damage development in sandwich composites are material and configuration dependent [6]. Hence, the manifest durability and damage tolerance characteristics of a damaged sandwich composite are also dependent on layup configuration, fabrication techniques, environmental conditions, and impact parameters. In addition, there likely is a structural size effect that influences impact damage development and subsequent sandwich panel failure modes.

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It is important that any impact damage falling below the threshold of detectability (TOD) using standard detection techniques not result in a serious degradation in structural integrity. Rhodes [25] showed that graphite/epoxy and kevlar/epoxy sandwich panels failed at very low loading when the structure was impacted at energy levels well below those that would produce visible damage. A number of studies have been directed at investigating the facing dent depths as a function of indentor shape and impact energy [25, 20, 26, 6]. Increasing the facesheet thickness does not necessarily improve the impact resistance of a given sandwich structure and may result in surface damage below the TOD for a particular impact. Again, increasing the crushing strength of the core material generally improves the impact resistance [6].

Fabrication techniques also have significant effects on the impact resistance of sandwich panels. Considerable degradation of the facesheet/core interface region can occur when a single-step process is used to cure the facesheets and bond the facesheets to the core [6]. Use of a two-step process involving prefabrication of the facesheets and subsequent facesheet/core bonding can result in enhanced damage tolerance characteristics in comparison to that obtained using a single step fabrication process, although the effect of the adhesive properties on damage tolerance becomes an important issue [27, 6]. Adhesive bondline degradation due to moisture ingression may lead to poor impact damage resistance [28, 6]. Baron et al. [28] developed a sandwich construction using fusion bonding of a thermoplastic foam and thermoplastic facesheets that was relatively impervious to moisture ingression. Further study of the effect of fabrication techniques on the damage tolerance properties of sandwich composites is warranted.

Thus far, the discussion has been limited to impact damage without facesheet penetration. Higher energy impacts resulting in partial or complete penetration of the sandwich structure have received fairly little attention in the literature. Such efforts have largely been limited to experimental studies [29]; no contact law exists to predict the complex force history during a penetration event. Abrate [6] noted that it is likely the penetration resistance is governed by the overall rigidity of the target and the resistance of the facings to penetration; core properties have little influence on perforation resistance.

3.1.3 Durability and Damage Tolerance of Sandwich Composites.

Numerous experiments have shown that low velocity impact damage in sandwich structures results in significant reductions in the residual strength in tension, compression, shear, and bending [6]. Typically, the strength after impact is unaffected until the impact energy exceeds a threshold value after which there is a marked reduction from the virgin strength. As mentioned previously, the nature and degree of sandwich structure impact damage is a function of a number of material system and impact parameters. It is highly desirable that impact damage resulting in an unacceptable reduction in mechanical properties be detectable via standard inspection techniques. Hence, special care must be used in selection of a layup configuration in order to ensure that critical internal damage due to impact will exceed the TOD and/or have accompanying BVID. This is especially a concern for sandwich structures with thick facesheets.

Hiel et al. [14] considered impact damage to sandwich structures with a syntactic foam core and skins comprised of glass fiber reinforced plastic (GFRP)/carbon fiber reinforced plastic. The

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outer layer of GFRP acted as a sacrificial protective coating and served to aid impact damage detection due to the localized discoloration that occurs in the impact region. The discoloration region corresponded to the amount of underlying facing/core delamination and was generally elliptic in shape with major diameter, *D*. Hiel et al. [14] showed that for a given range of discoloration zone sizes, *D*, the failure modes and residual strength of the sandwich composite were analogous to those of an undamaged panel with an open circular hole of comparable size. Hence, the residual strength of the facing may be estimated using available methods for predicting the residual strength of open-hole composites.

Saczalski et al. [30] used an innovative factorial-based design of experiments technique to determine the effect of foreign object impact on the damage induced and residual strength of sandwich composites comprised of graphite/epoxy facesheets and metallic honeycomb core. Based on a number of carefully selected experiments, statistically reliable polynomial expressions for the residual strength, dent depth, and damage area were determined as a function of the layup configuration (core thicknesses, facesheet thicknesses) and impact energy. One key advantage of this approach is that it allows for estimation of the impact damage and residual strength associated with sandwich panel configurations and/or impact energy levels not considered in the original test program. In addition, the nonlinear interaction effects between critical design parameters (residual strength, impact energy, panel configuration, etc.) may be readily obtained. This study represents one of the few comprehensive treatments of the sandwich composite impact problem that addresses a wide range of governing material system and impact parameters.

Russell et al. [31] summarized relevant FAA requirements, analysis methodologies, and selected test data for impact damage tolerance and fail safety of composite sandwich panels. Characteristic impact damage, failure modes, and residual strength prediction methodologies were discussed for typical nonmetallic honeycomb core sandwich panels. Test data confirmed the presence of facesheet and core damage zones of differing sizes along with a multiplicity of potential failure modes that occur in a specific sequence during panel failure. A number of guidelines for impact damage tolerant design were presented including: (1) sandwich panels should fail initially by facesheet compression (rather than facesheet buckling) so that the full strength of the composite facesheets can be realized, and (2) extensive core damage should not develop at a lower impact level than detectable facesheet damage since this can lead to local facesheet buckling at extremely low panel strain levels. A tear strap methodology for fail-safe composite sandwich panel design was discussed, along with pertinent residual strength test data for composite sandwich tear strap panels. Tear straps can be formed by interleaving extra plies of material into composite facesheets at specified spacings and may provide load redistribution capability in the presence of large damage.

McGowan and Ambur [32] conducted an experimental study of the impact damage characteristics and residual strength of composite sandwich panels (AS4/8552 graphite epoxy tape/woven fabric facesheets and Korex® honeycomb core) with and without a compressive preload. The study suggested that use of dent depth only as a measure of the extent of damage in an impacted structure might not always be reliable or realistic. McGowan and Ambur [33] obtained similar results in the study of composite fuselage sandwich structures with thick

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facesheets. For thick facesheets, one must consider the use of energy cut offs that represent realistic threats to fulfill ultimate load, BVID requirements.

Rouse et al. [34, 35] performed combined experimental and analytic studies of composite fuselage side panels (AS4/8552 graphite epoxy tape/woven fabric facesheets and Korex® honeycomb core) with window cutouts and underlying reinforcing frames for use in transport aircraft. The studies suggested that inclusion of composite sandwich panels in fuselage design may allow for average frame spacings that are nearly double those used in conventional metallic fuselage design, thus resulting in a potentially significant weight and cost savings.

Kassapoglou conducted several studies related to the damage tolerance of composite laminates and thin-gage sandwich panels where damage is typically indentation, debond, and delamination [36, 37, 38]. He demonstrated a method for analyzing laminated composite panels with elliptical delaminations [36]. This method predicted the failure of the panel to occur when the stress in the resin layer at the edge of the delamination reaches an ultimate load based on von Mises failure criteria. He also defined delamination threshold size as the delamination size at which the failure of the panel in compression changes form global column buckling to local delamination buckling.

Kassapoglou presented a method to estimate the failure loads of sandwich panels with threshold of detectability (TOD) damage [37]. The author postulated that the complex geometry, matrix cracking, indentations, and delaminations resulting from an impact can be modeled using a single equivalent elliptical delamination. The size of this equivalent delamination can be related to the analytically determined delamination threshold size. Experimental data showed that the equivalent delamination size for TOD damage was 1.22 times the delamination threshold for the panels used. This number is dependent on the matrix material and damage size. A strength ratio of 1.49 was also proposed (ratio of the strength of the undamaged sandwich panel to the strength of the TOD damaged panel) [38]. This number is based on empirical results of the ratio of equivalent delamination size of a TOD damaged panel to the delamination threshold size of the intact panel (1.22) and the analytical methods used to predict failure of a panel with a delamination. This number is independent of geometry or layup but is dependent on material. These prediction methods have good correlation with experimental results. Kassapoglou demonstrated simplified methods for predicting the compressive failure of damaged composite panels, but the methods may be limited in their application due to their material and geometry dependence. The analysis and experimentation was limited to flat, thin-gage sandwich structures. In addition, all damage was modeled using only an equivalent delamination, indentation was not considered in the analysis models.

Minguet presented a different approach to modeling the residual strength of damaged sandwich structures [39]. He used the indentation depth as a parameter for evaluating the buckling load of the damaged panel. The core is modeled as a generally orthotropic material with a linear stress-strain relation up to the stress that causes crushing of the core. The crushed core then supports a constant stress lower than its crushing load. The undamaged back facesheet was restrained out of plane to force local buckling at the impact indentation. Results from the model show failure as a growth of the initial indention perpendicular to the applied compressive load and show good

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correlation with experimental data. This analysis is limited to thin-gage facesheets where fracture of the facesheets due to bending will occur only at large deflections. The method for modeling the initial impact damage considers facesheet indentation and core damage and neglects the effects of delaminations and matrix fracture.

3.2 THE ADVANCED COMPOSITES TECHNOLOGY (ACT) PROGRAM.

The damage resistance and tolerance of two composite design concepts for the fuselage structure of a wide body transport aircraft was investigated as part of the Boeing Advanced Technology Composite Aircraft Structures (ATCAS) program which was funded under the NASA-Advanced Composites Technology (ACT) initiative. The primary objective of the ACT/ATCAS program was to develop and demonstrate the integrated technology base required to make cost and weight-effective use of advanced composite materials in pressurized fuselage primary structure. The representative fuselage section just aft of the wing-to-body intersection of a commercial wide-body transport aircraft (see figure 8) was chosen for the study. This section was selected because it contained many of the structural details found throughout the fuselage.

Two composite design concepts were identified for the ATCAS side quadrant. The first design consisted of a skin/stringer/frame configuration with mechanically fastened circumferential frames, stringer clips, and J-section stringers cobonded to the laminate skin. The side panel assembly for this design concept is illustrated in figure 9. The second design was a sandwich configuration with cobonded circumferential frames and is illustrated in figure 10. The global cost and weight evaluations of these two composite design configurations and a baseline aluminum design were conducted to compare the weight reductions and cost-effectiveness of these designs and are reported elsewhere [40]. This baseline concept selection and global evaluation activities resulted in the selection of a skin/stringer configuration for the crown and sandwich construction for the keel and side quadrants. Additional final documentation of the ATCAS program can be found on materials and manufacturing processes [41], manufacturing [42] structural performance [43], repair and damage assessment supporting maintenance [44], impact damage resistance [45], and a program overview [46].

The damage resistance of composite structures was of particular concern because they have historically been susceptible to foreign object impact damage. Damage tolerance evaluations were performed to ensure adequate structural performance in the presence of potential large-scale damage. The impact damage resistance and post impact load carrying capability of stiffened skin and sandwich structures were investigated in the ACT/ATCAS program and are summarized in the following sections.

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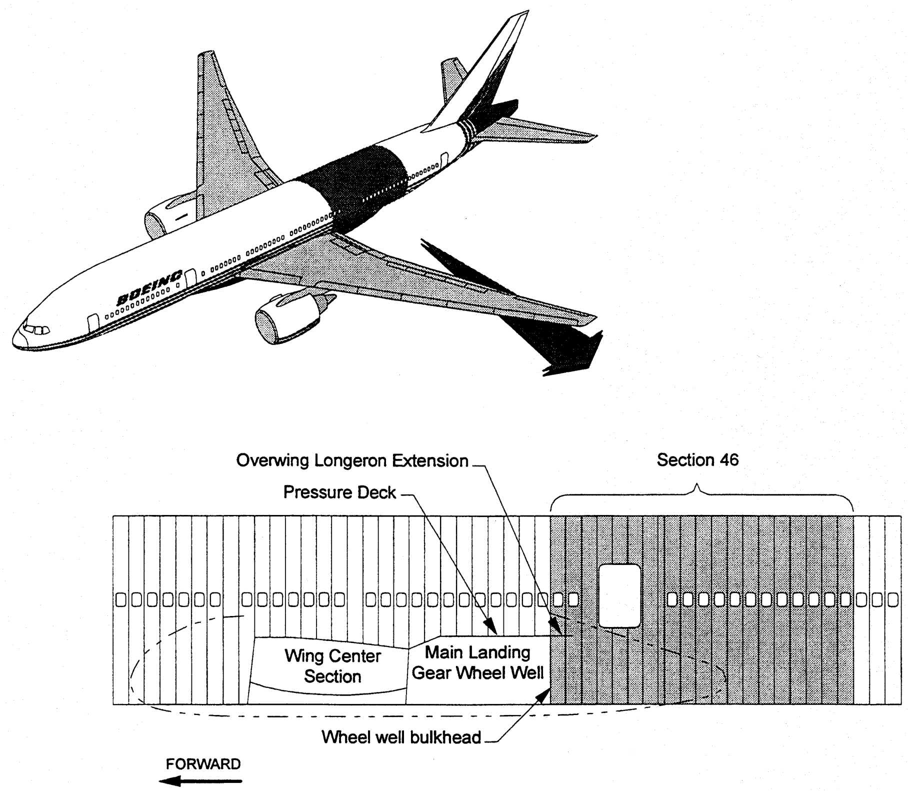

FIGURE 8. BASELINE VEHICLE AND STUDY SECTION

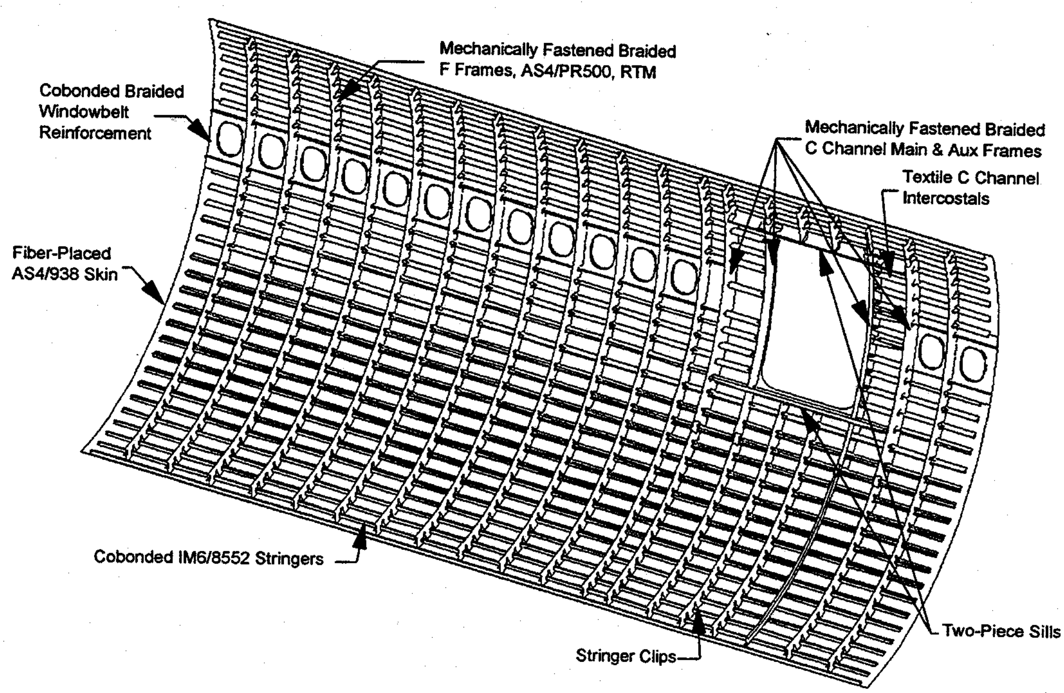

FIGURE 9. THE COMPOSITE SKIN/STRINGER CONFIGURATION SIDE PANEL

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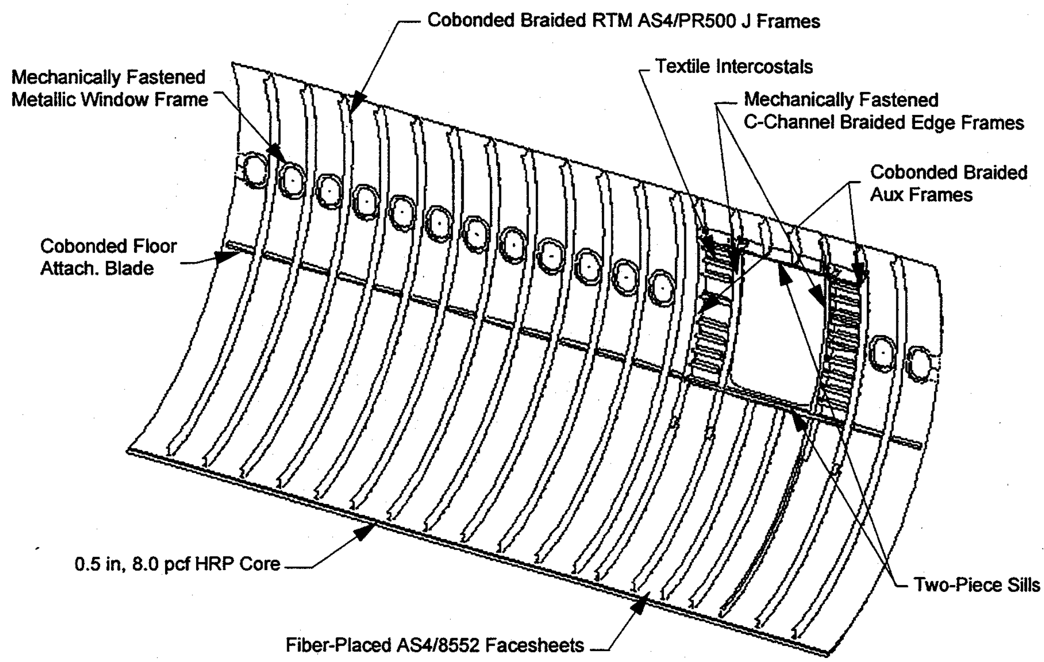

FIGURE 10. THE COMPOSITE SANDWICH CONFIGURATION SIDE PANEL

3.2.1 Stiffened Skin/Laminate Structures.

The impact damage resistance of stiffened skin structure was studied experimentally using a statistically based designed experiment. A total of 32 different stiffened panels were tested to assess the impact damage resistance/response of representative laminated composite transport aircraft fuselage structure. The skin/stiffener panels used in the experiments were representative of potential fuselage crown, keel, and lower side panel designs. A number of intrinsic and extrinsic variables were identified that affect the impact damage response of the panels. A total of 14 such parameters were selected for investigation and are listed in tables 1 and 2.

A design of experiments approach was used to investigate the coupled effect of intrinsic and extrinsic parameters on the structural impact damage response. The experiment represented a 32 run, resolution IV, fractional factorial design, allowing the study of the large number of variables with a relatively small number of specimens. This methodology provided information on the effect of each variable on the damage created and indicated variable interactions, if any. The use of this experimental approach limited the number of variable levels to two for each intrinsic and extrinsic variable; the levels chosen in the investigation are listed in the tables 1 and 2. The range of intrinsic an extrinsic parameters considered in the study were based on practical considerations and the knowledge gained during the ACT/ATCAS program.

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TABLE 1. INTRINSIC VARIABLES

Variable	Low Level	High Level
Fiber Type	AS4	IM7
Matrix Type	938	977-2
Fiber Volume	48%	56.5%
Material Form (Stiffener Layup)	Tow (Soft)	Tape (Hard)
Skin Layup	Soft	Hard
Stiffener Type	Blade	Hat
Stiffener Spacing	17.8 cm (7 in)	30.5 cm (12 in)
Laminate Thickness	2.26 mm (0.0888 in)	4.51 mm (0.1776 in)
	Layups	
Hard Skin (Thin)	$(45/90/-45/0/90/0)_S$	
Soft Skin (Thin)	$(45/90/-45/45/0/-45)_S$	
Hard Skin (Thick)	$(45/90/-45/0/45/90/-45/0/90/0/90/0)_S$	
Soft Skin (Thick)	$(45/90/-45/45/0/-45/-45/0/45/-45/90/45)_S$	
Hard Stiffener (Thin)	$(22.5/90/-22.5/0)_S$	
Soft Stiffener (Thin)	$(30/90/-30/0)_S$	
Hard Stiffener (Thick)	$(22.5/90/-22.5/0)_{2S}$	
Soft Stiffener (Thick)	$(30/90/-30/0)_{2S}$	

TABLE 2. EXTRINSIC VARIABLES

Variable	Low Level	High Level
Impactor Stiffness	2.8 GPa (0.4 Msi)	210 GPa (30 Msi)
Impactor Mass	0.28 kg (0.62 lbm)	6.31 kg (13.9 lbm)
Impact Energy		
Midbay Impacts	23 J (200 in-lb)	136 J (1200 in-lb)
Stiffener Impacts	40 J (350 in-lb)	181 J (1600 in-lb)
Impactor Shape	Flat	Spherical
Impactor Diameter	6.35 mm (0.25 in)	25.4 mm (1.00 in)
Temperature at Impact	21°C (70°F)	83°C (180°F)

3.2.1.1 Impact Testing.

The impact testing consisted of high mass (6.31 kg/13.88lbs), low mass (0.28 kg/0.62lbs), and hail simulation tests. Different equipment was used for simulating these impact conditions.

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3.2.1.1.1 Low-Mass Impacts.

Low-mass impacts were performed using a horizontally oriented nitrogen gas gun. The projectile consisted of steel or graphite tup, an Aluminum shaft, load cell and a torsion shell. A typical projectile used in the experiments is shown in figure 11.

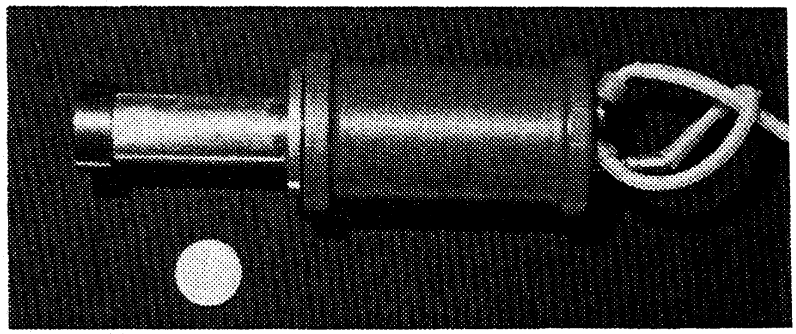

FIGURE 11. A TYPICAL LOW-MASS IMPACTOR

3.2.1.1.2 High-Mass Impacts.

High-mass impacts were conducted using a Dynatup 8250 drop weight impact tester. The load cell and the tups used in the experiments are shown in figure 12.

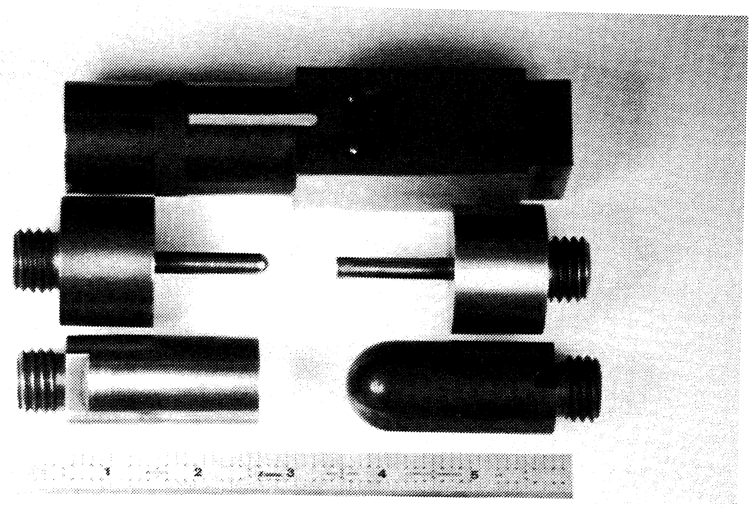

FIGURE 12. HIGH-MASS IMPACTORS USED IN THE INVESTIGATION

3.2.1.1.3 Hail Simulation.

The damage created by a 63.5-mm (2.5-in) hail was studied experimentally. Hail of this size was found to have an energy of 56.7 J (500 in-lb) at terminal velocity. A 63.5-mm lead ball was dropped from a height of 3.43 m (11.25 feet) to achieve an equivalent energy.

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3.2.1.2 Support Conditions.

The support conditions applied to all panels were designed to simulate the circumferential frames found in aircraft fuselages. Supports were placed with a span of 0.51 m between their inner edges to simulate the fuselage 0.56-m (22-inch) centerline-to-centerline frame spacing.

3.2.1.3 Impact Tups.

Four different impact tup geometries were considered. Each geometry was manufactured using both hard and soft materials. The material selected for hard impactor was steel, with a modulus of elasticity of 210 Gpa. A2 steel was used for high-mass impact tups while 4140 steel was used for low-mass impact tups. All steel tups were hardened to 55-60 Rockwell. Graphite/Epoxy cut transversely to the plane of lamination was used for soft impactors. The graphite/epoxy tups were rough cut from 31.75-mm (1.25-inch) -thick blocks of tool grade graphite-epoxy (Toolrite MXG-7650 style 2577) using a water jet cutter. These rough cut tups were then machined appropriately to achieve required tolerances.

3.2.1.4 Damage Characterization.

The design of experiments technique required quantifiable response measurements for all 32 runs to evaluate the effects of the variables and their interactions. Responses measured and studied are indentation depth, planar damage area, stiffener separation area, fiber damage average length and through-thickness distribution, and local flexural stiffness. The experimental techniques used to obtain the required responses are described below.

3.2.1.4.1 Residual Surface Indentation/Visibility.

The indentation depth at the impact sites was measured with a dial indicator. The dial indicator was mounted in a specially designed fixture that used the undamaged region surrounding the impact site to provide a reference to the original panel surface. This allowed measurement of the indentation with respect to the original undamaged surface.

3.2.1.4.2 Matrix Damage.

A pulse–echo ultrasonic technique was used to locate the impact induced internal delaminations by examining the amplitude and time of flight of a reflected high-frequency (10-MHz) short-duration pulse perpendicular to the laminate surface. All panels were initially subjected to a low spatial resolution planar inspection to map the extent and location of damage created during impact. Damage sites found in the overall panel C-scans were then inspected using higher-resolution ultrasound scans. The damage found in these detailed scans provided information on the first levels of delaminations in the laminate. Destructive cross section examination was also performed on an experimental panel to determine the through-thickness location of matrix damage hidden from ultrasound inspection. A cylindrical sectioning technique was used to visualize the matrix damage. A fluorescent dye penetrant was used to aid visual inspection.

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3.2.1.4.3 Fiber Failure.

The fiber failures in the laminates were measured by thermally deplying the damage sites from the test panels and measuring the length of broken fibers in each ply. The length and orientation of failed fibers in each ply was recorded and fiber failure maps were generated for all deplied coupons. A typical ply-by-ply fiber failure map is shown in figure 13.

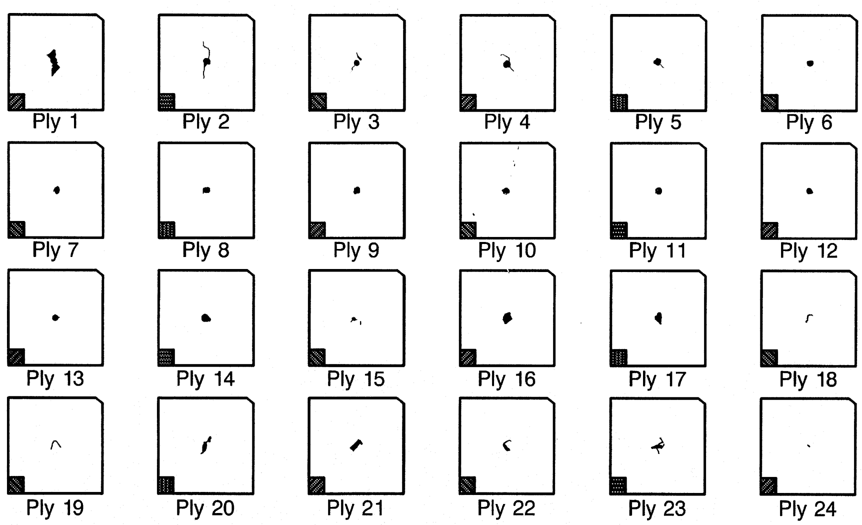

FIGURE 13. TYPICAL PLY-BY-PLY FIBER FAILURE MAP

In addition to the discrete measurements of damage discussed above, nondiscrete measurements of the damage were also conducted during the investigations. The nondiscrete measurements relate directly to the structural response. The reduction in local flexural and transverse shear stiffness of the damage zone was used as a measurement of the damage. The particular technique used is summarized below.

3.2.1.4.4 Flexural Wave Stiffness Determination.

An inspection method based on flexural (Lamb) wave propagation was used to estimate the stiffness reduction associated with simulated hail impact damage. Flexural wave phase velocity measurements for propagation at four different frequencies (14, 25, 39, and 84 kHz) in both undamaged and damaged regions were made using a ZETEC Sondicator model S-9.

Dispersion curves (phase velocity versus frequency) were generated for each panel and the measurements for the undamaged portions of the panel were compared to theoretical values based on laminated plate theory, which includes the effects of transverse shear deformation and rotary inertia. The agreement between predicted and measured phase velocities was typically within 5% for all 32 panels. Similar phase velocity measurements were also conducted in damaged regions. The stiffness of the damaged region was back-calculated from dispersion measurements and laminated plate theory. The measurements were made in the 0° direction, and

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thus the decrease in stiffness was attributed to a reduction of the bending stiffness (D_{11}) and the transverse shear stiffness (A_{55}).

3.2.1.5 Results.

The experimental data generated were subjected to statistical analysis. Due to the highly fractionated and nonreplicated nature of this experimental design, the use of a straightforward analysis of variance (ANOVA) model to analyze the data was impossible. Half-normal plots of the effect estimates were constructed to analyze the data. The details of these analyses may be found in NASA-CR-4658.

The experiments indicated that the severity of damage was a strong function of impact variables and variable interactions. An inverse relation observed between the damage severity and visibility relating to the impactor geometry suggested that the BVID or similar criteria might be flawed. More comprehensive material and design-screening approach for composite structures is warranted based on the strong coupling observed between the extrinsic impact variables and the damage characteristics. *This study illustrated that the use of a single arbitrary value for an extrinsic variable in a test program may lead to conclusions which may not apply over the full range of potential impact conditions.* The effects of the variables considered in this investigation are summarized below.

3.2.1.5.1 Fiber Type.

- Laminates fabricated with IM7 fibers increased the local flexural stiffness in the damage region for hail simulation midbay impacts.

3.2.1.5.2 Matrix Type.

- Tougher matrix materials decreased local matrix damage area for the midbay, flange edge, and web impacts.
- Tougher matrix decreased flange separation area for web impacts.
- Tougher matrix increased damage region local flexural stiffness for hail simulation midbay impacts.

3.2.1.5.3 Material Form (Stiffener Layup).

- Tape laminates (hard stiffeners) had a more uniform fiber failure distribution through the thickness for midbay impacts.
- Tape laminates had decreased fiber failure average width for stiffener web impacts.

3.2.1.5.4 Stiffener Type.

- Hat stiffeners increased local matrix damage area for the web impacts.
- Hat stiffeners decreased flange separation area for web impacts.

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3.2.1.5.5 Laminate Thickness.

- Thick laminates decreased indentation depth for hail simulation midbay impacts.
- Thick laminates increased flange separation area for the flange impacts.
- Thick laminates decreased fiber failure average width for midbay and web impacts.
- Thick laminates increased damage region local flexural stiffness for hail simulation midbay impacts.

3.2.1.5.6 Impactor Mass.

- Higher mass increased indentation for midbay impacts.

3.2.1.5.7 Impact Energy.

- Higher energies increased the indentation for the midbay, flange, and web impacts.
- Higher energies increased local matrix damage area for the midbay, flange, and web impacts.
- Higher energies increased flange separation area for the flange and web impacts.
- Higher energies had fiber failures concentrated towards the back surface in the fiber failure through-thickness distribution for the midbay impacts.

3.2.1.5.8 Impactor Shape.

- Spherical impactor increased indentation for midbay impacts
- Spherical impactor increased fiber failure average width for midbay impacts.

3.2.1.5.9 Impactor Diameter.

- Larger diameter impactor decreased indentation for the midbay, web, and flange impacts.
- Larger diameter impactor increased local matrix damage area for the midbay, web, and flange impacts.
- Larger diameter impactor increased flange separation area for the flange and web impacts.
- Larger diameter impactor increased fiber failure average width for the midbay impacts.

3.2.1.5.10 Temperature.

- Higher temperatures increased indentation for the midbay impacts.

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This investigation may provide an excellent guideline for future in-depth studies. The variables that most greatly influence the impact damage response must be isolated in order to conduct a more detailed investigation. Further investigation of the appropriate ranges of intrinsic and extrinsic variables may be warranted. Note that establishing a threshold for BVID was not addressed in this study.

3.2.2 Sandwich Structures.

The impact damage resistance of sandwich structures was also investigated as part of the ACT/ATCAS program [41]. Initially, candidate core materials were subjected to a screening process. The impact damage resistance and compression after impact (CAI) strengths of sandwich panels representative of minimum-gage fuselage structure with a variety of honeycomb core materials were characterized to aid core material selection.

The impact survey coupons were constructed of 8-ply quasi-isotropic AS4/8553-40 facesheets and 0.5-inch-thick core. Developmental and off-the-shelf products were considered for the cores. Cost/weight design trade studies resulted in the identification of Hexcel's HRP honeycomb as the baseline core material for primary structural applications. The candidate core materials are listed in table 3. The AS4/8553-40 facesheet material had a fiber aerial weight of 190 g/m^2 and 40% resin content. CYTEC's Metalbond 1515 film adhesive was used for cocuring the facesheets to the core materials.

TABLE 3. CANDIDATE CORE MATERIALS

Material Designation	Description	Cell Size (in)	Density (lbs/ft^3)
HRP-3/16-5.5	0/90 glass/phenolic honeycomb	3/16	5.5
HFT-1/8-5.5	±45 glass/phenolic honeycomb	1/8	5.5
TPC-3/16-5.5	±45 glass/polyamide-imide honeycomb	3/16	5.5
HFT-G-3/16-6.0	±45 carbon/polyimide honeycomb	3/16	6.0
HRH-10-1/8-5.0	Aramid paper/phenolic honeycomb	1/8	5.0
HRP-3/16-5.5 K-foam	0/90 glass/phenolic honeycomb cyanate-ester foam	3/16	5.5 [6.0*]
HRH-10-1/8-5.0 K-foam	Aramid paper/phenolic honeycomb cyanate-ester foam	1/8	5.0 [6.0*]

*Density of foam

Sandwich test coupons were impacted on the toolside surface over a 4- by 4-inch simple support with 1-inch-diameter steel impactor at impact energies of 80, 100, 200, and 500 inch-pounds. The tested coupons were then subjected to nondestructive inspection using through transmission ultrasonic (TTU) C-scan, and the indentation depth was also measured. CAI tests were also performed on 5- by 10-inch coupons cut from the impacted panels. NASA's ST-1 method was adopted for performing the CAI tests.

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3.2.2.1 Screening Test Results.

3.2.2.1.1 Impact Dent Depth.

The performance of different core materials as characterized by the impact dent depth incurred due to variable energy level impacts is shown in figure 14. The impact dent depth was essentially unaffected by the core material type at low impact energies (≤ 200 in-lbs.). Also, none of the panels sustained a visible damage (dent depth ≥ 0.05″) from the impacts of ≤ 100 in-lbs. The 500-in-lbs. impacts resulted in perforation of the first facesheet in most plots as is evident in figure 14 and subsequent impact of the farside sheet in most cases.

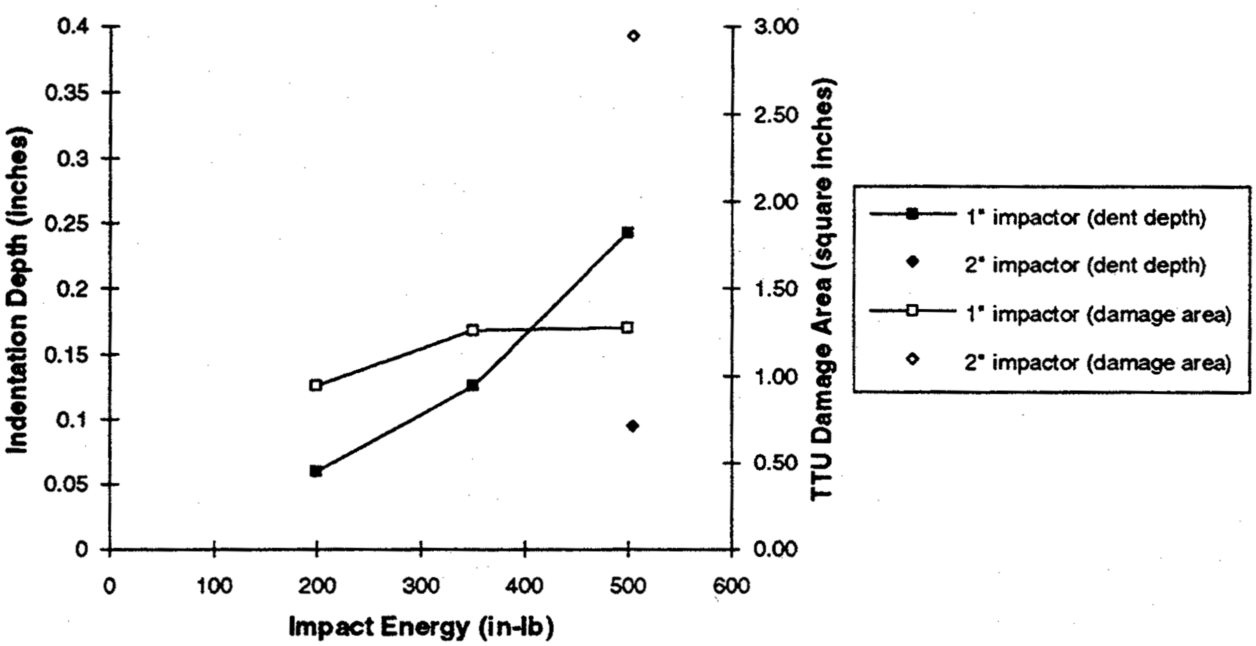

FIGURE 14. IMPACT DAMAGE RESISTANCE OF BASELINE AFT KEEP SANDWICH STRUCTURE

3.2.2.1.2 Planar Damage Area.

The planar extent of damage appeared to approach an asymptote for increasing impact energy as the impacted facesheet was perforated at high energy levels as shown in figure 15. The resin matrix type of the core material was observed to have the greatest influence on damage resistance. The thermoplastic matrix core provided greater impact damage resistance compared to the rest. The damage area of foam-filled honeycomb was found to be slightly greater than that of the unfilled ones as shown in figure 16.

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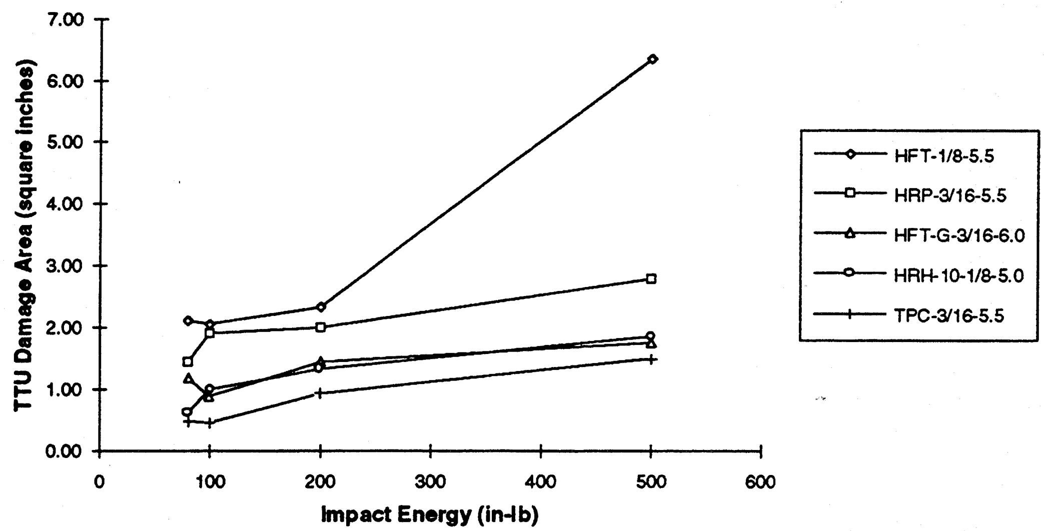

FIGURE 15. PLANAR DAMAGE AREA FOLLOWING IMPACT OF SANDWICH STRUCTURE WITH HONEYCOMB CORE MATERIALS

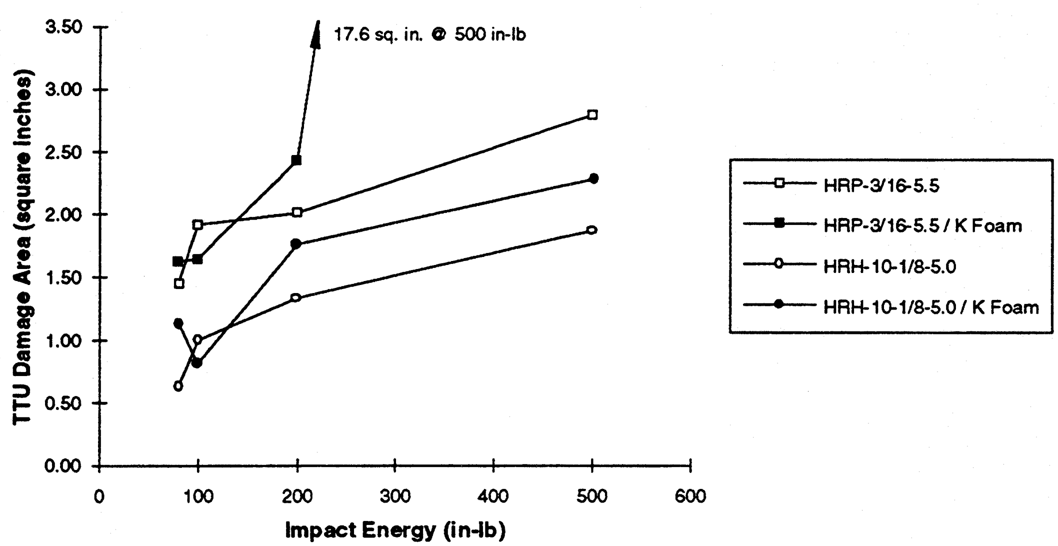

FIGURE 16. PLANAR DAMAGE AREA FOLLOWING IMPACT OF SANDWICH STRUCTURE WITH FOAM-FILLED HONEYCOMB CORE MATERIALS

3.2.2.2 Effects of Intrinsic and Extrinsic Variables.

The influence of extrinsic and intrinsic variables on the impact damage resistance and damage tolerance of composite sandwich structures were investigated. The variables considered were:

- Core material type and density [intrinsic]
- Facesheet thickness[intrinsic]

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- Impact event support type [extrinsic]
- Impactor size [extrinsic]
- Impact energy [extrinsic]

3.2.2.2.1 Test Matrix.

The test matrix shown in table 4 was used in the investigation. A total of five test panels were constructed of AS4/8553-40 facesheets and 0.9-inch-thick honeycomb core. The core cell size was 3/16 inch for all panels.

TABLE 4. TEST PANEL CONFIGURATIONS

Facesheet Layup	Facesheet Thickness (in)	Core Material	Core Density (lb/ft^3)
$[45/90/0/-45/0]_S$	0.078	HRP-3/16-5.5	5.5
		HRP-3/16-12.0	12.0
		TPC-3/16-5.5	5.5
$[45/90/0/-45/0]_{3S}$	0.234	HRP-3/16-5.5	5.5
		HRP-3/16-5.5	5.5

The test panels were impacted with the 1″ and 2″ diameter steel tups at various impact energies over simple and rigid supports. The simple and rigid supports were intended to simulate impact over and between stiffeners. The depth of resulting indentation was measured for each impact and the panels were inspected by TTU C-scan to determine the planar extent of core damage.

Several key experimental results are as follows:

- The impact support type had a minor influence on damage resistance.

- Greater dent depth was observed for impact with rigid supports.

- Impact with the larger tup resulted in a smaller indentation and greater damage area in almost all the cases.

- Core type and density was observed to affect the damage resistance significantly. The denser (12 lb/ft^3) core sandwich panel sustained less indentation in most cases and less damage area in all cases than the lower density (5 lb/ft^3) core sandwich panel.

- The panels with thicker facesheets sustained smaller indentations and damage over a smaller area than those with thin facesheets for a given energy level.

3.3 TECHNOLOGY LABORATORY FOR ADVANCED COMPOSITES (TELAC).

The program at the Technology Laboratory for Advanced Composites (TELAC) has concentrated on separating the commonly used indicator of compression after impact into two separate concepts: damage resistance and damage tolerance. This offers the advantages of separating

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exclusive effects and being able to isolate and study damage modes and failure mechanisms. In particular, the work has concentrated on both analytical and experimental research on the damage resistance of laminated plates and was then extended to include sandwich structures. Their damage tolerance work has been mostly limited to experimental development.

Cairns conducted an extensive analysis of the impact response of composite laminates [47]. The issues of damage resistance and damage tolerance are treated separately. The impact event was modeled with global and local analytical models. Experimental data were used to validate the models using a range of parameters including impact energy, material, impactor mass, and boundary conditions. The residual strength of the damaged laminates was predicted using calculations based on Lekhnitskii's work on an anisotropic plate with an elliptical inclusion. The compliance matrix of the inclusion is found by using the damage resistance analysis to determine which plies of the damaged zone have failed, and then performing a Classical Laminated Plate Theory based evaluation of the remaining laminate. The calculations result in a stress concentration factor for the material near the inclusion. Failure due to uniaxial tension is predicted by averaging the stresses in a region near the inclusion (the size of the region is a material dependent parameter). This stress value is compared to the far field stress in the panel and results in a strain ratio (S.R.). The S.R. represents the ratio of residual strength of the damaged panel to the strength of the undamaged panel. Cairns' work on laminated composite panels is considered important and has been referenced in many other papers.

Lie developed analytical models to describe the impact event and compression failure of thin-gage composite sandwich panels (graphite/epoxy facesheets with a Nomex honeycomb core [48]. As an extension to the work by Cairns, the damage event analysis also uses a local and global model, but this model is restricted to orthotropic facesheets. For damage tolerance analysis, the damaged panels are modeled as an intact panel with an elliptical hole representing the damage. Capacity of the damaged panels is calculated by considering the notch sensitivity of the layups of the facesheets. Failure of notch insensitive layups (±45° plies) is estimated using a net section method. Failure of notch sensitive layups (0°/90° plies) is estimated using the Mar-Lin relation. This work also focuses on the impact event and predicting damage resistance rather than damage tolerance.

Lagace et al. [49] investigated the damage resistance and damage tolerance of AMOCO's ERLX-1983 3K-70 PW graphite/epoxy plain weave fabric used in a thin-gage sandwich panel with a Nomex honeycomb core. Test panels were fabricated and subjected to impact damage in order to measure the energy required to create both barely and easily visible impact damage. The residual strength of the damaged panels was then measured in both tension and compression. It was noted that panels with very low-energy impact damage showed no reduction in tensile strength but had a significant loss of compressive strength. This work makes no attempt to analytically model the impact event or predict the residual strength of a damaged panel.

3.4 ILLUSTRATIVE SUMMARY.

Based upon previous investigations into the impact damage associated with laminated sandwich structures, some comparison may be drawn among the various variables common to most studies.

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This comparison is meant to provide an illustrative summary as represented in figures 17-19. These figures represent ranges of impact energies, core thicknesses, and facesheet thicknesses which have been used in past investigations. The author is also listed with each range and may be found in the references.

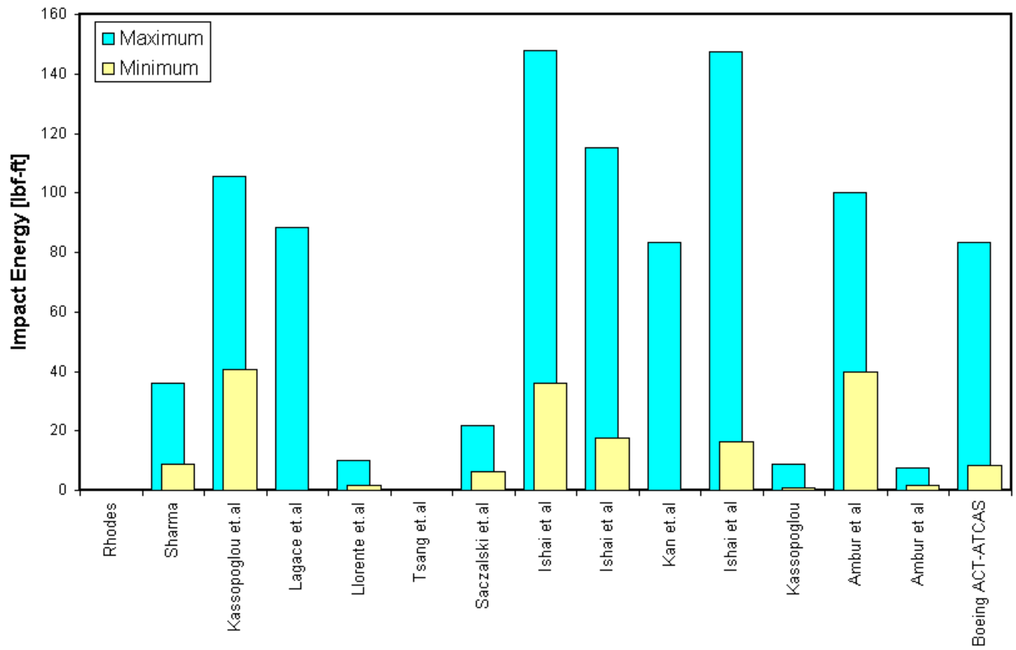

FIGURE 17. RANGES OF IMPACT ENERGY USED IN PREVIOUS INVESTIGATIONS

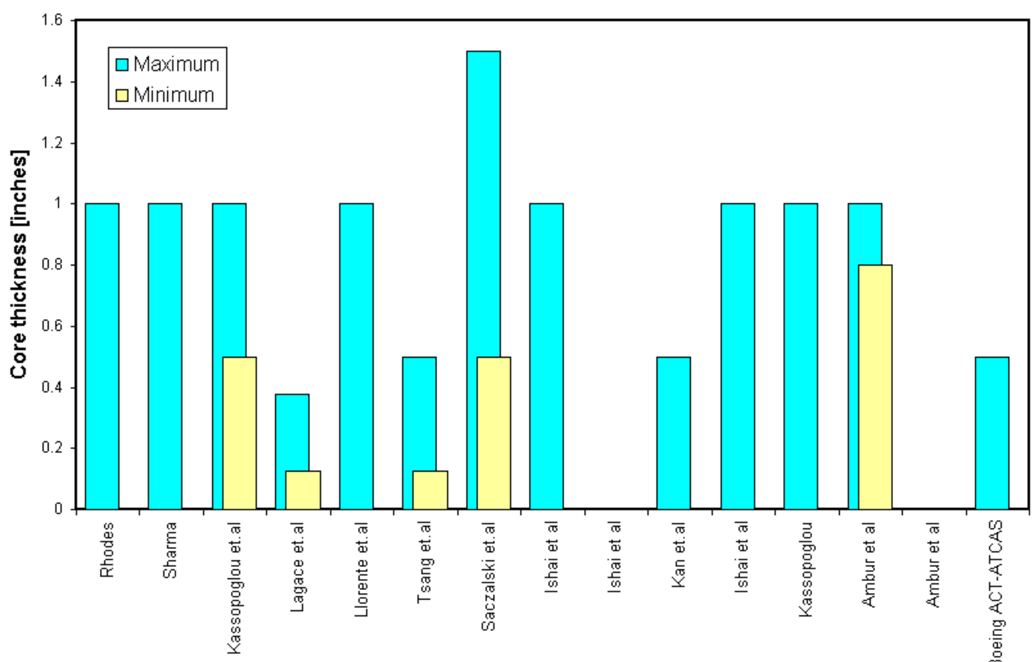

FIGURE 18. RANGES OF CORE THICKNESS USED IN PREVIOUS INVESTIGATIONS

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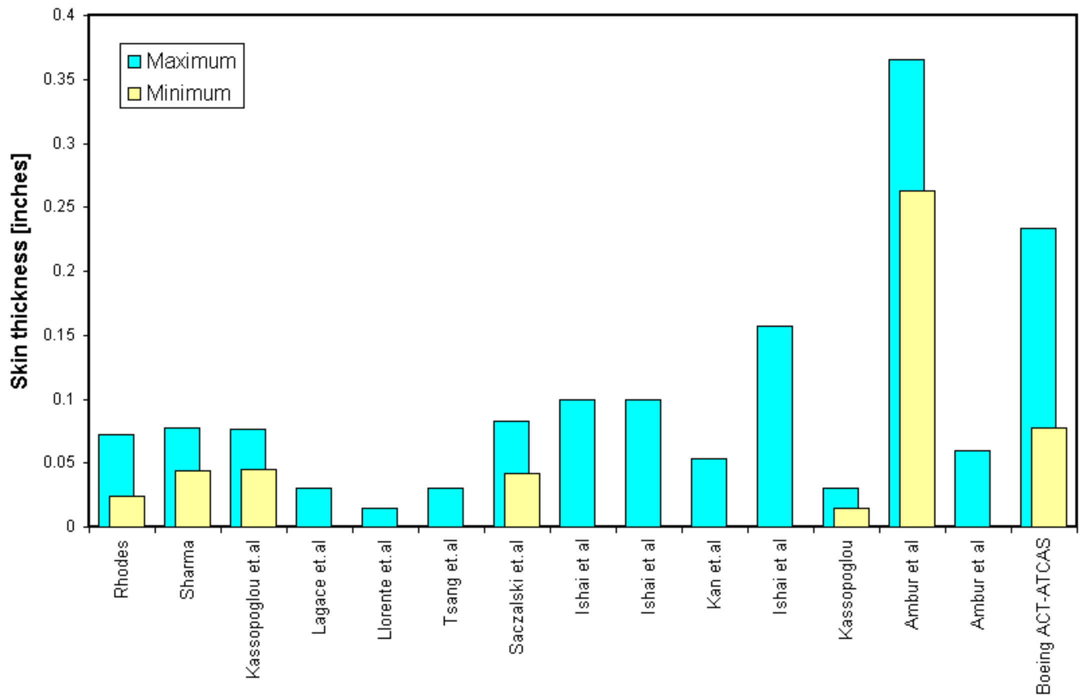

FIGURE 19. RANGES OF SKIN THICKNESS USED IN PREVIOUS INVESTIGATIONS

4. DAMAGE TOLERANCE CERTIFICATION PROCEDURES.

4.1 FEDERAL AVIATION REGULATIONS.

The following excerpts are taken directly from the Federal Aviation Regulations (1996 edition) and describe damage tolerance related issues for both composite and metallic aircraft and rotorcraft. These sections are presented to provide a basic guideline and reference during the proposed damage tolerance investigations.

4.1.1 Part 23 - Airworthiness Standard: Normal, Utility, Acrobatic, and Commuter Category Airplanes.

23.571 Metallic pressurized cabin structures.

For normal, utility, and acrobatic category airplanes, the strength, detail design, and fabrication of the metallic structure of the pressure cabin must be evaluated under one of the following:

(a) A fatigue strength investigation in which the structure is shown by tests, or by analysis supported by test evidence, to be able to withstand the repeated loads of variable magnitude expected in service; or

(b) A fail-safe strength investigation, in which it is shown by analysis, tests, or both that catastrophic failure of the structure is not probable after fatigue failure, or obvious

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partial failure, of a principal structural element, and that the remaining structures are able to withstand a static ultimate load factor of 75 percent of the limit load factor at VC, considering the combined effects of normal operating pressures, expected external aerodynamic pressures, and flight loads. These loads must be multiplied by a factor of 1.15 unless the dynamic effects of failure under static load are otherwise considered.

(c) The damage tolerance evaluation of § 23.573(b).

23.572 Metallic wing, empennage, and associated structures.

(a) For normal, utility, and acrobatic category airplanes, the strength, detail design, and fabrication of those parts of the airframe structure whose failure would be catastrophic must be evaluated under one of the following unless it is shown that the structure, operating stress level, materials, and expected uses are comparable, from a fatigue standpoint, to a similar design that has had extensive satisfactory service experience:

(1) A fatigue strength investigation in which the structure is shown by tests, or by analysis supported by test evidence, to be able to withstand the repeated loads of variable magnitude expected in service; or

(2) A fail-safe strength investigation in which it is shown by analysis, tests, or both, that catastrophic failure of the structure is not probably after fatigue failure, or obvious partial failure, of a principal structural element, and that the remaining structure is able to withstand a static ultimate load factor of 75 percent of the critical limit load factor at VC. These loads must be multiplied by a factor of 1.15 unless the dynamic effects of failure under static load are otherwise considered.

(3) The damage tolerance evaluation of § 23.573(b).

23.573 Damage tolerance and fatigue evaluation of structure.

(a) Composite airframe structure. Composite airframe structure must be evaluated under this paragraph instead of §§ 23.571 and 23.572. The applicant must evaluate the composite airframe structure, the failure of which would result in catastrophic loss of the airplane, in each wing (including canards, tandem wings, and winglets), empennage, their carrythrough and attaching structure, moveable control surfaces and their attaching structure fuselage, and pressure cabin using the damage tolerance criteria prescribed in paragraphs (a)(1) through (a)(4) of this section unless shown to be impractical. If the applicant establishes that damage tolerance criteria is impractical for a particular structure, the structure must be evaluated in accordance with paragraphs (a)(1) and (a)(6) of this section. Where bonded joints are used, the structure must also be evaluated in accordance with paragraph (a)(5) of this section. The effects of material variability and environmental conditions on the strength and durability properties of the composite materials must be accounted for in the evaluations required by this section.

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(1) It must be demonstrated by tests, or by analysis supported by tests, that the structure is capable of carrying ultimate load with damage up to the threshold of detectability considering the inspection procedures employed.

(2) The growth rate or no growth of damage that may occur from fatigue, corrosion, manufacturing flaws or impact damage, under repeated loads expected in service, must be established by tests or analysis supported by tests.

(3) The structure must be shown by residual strength tests, or analysis supported by residual strength tests, to be able to withstand critical limit flight loads, considered as ultimate loads, with the extent of detectable damage consistent with the results of the damage tolerance evaluations. For pressurized cabins, the following loads must be withstood:

(i) Critical limit flight loads with the combined effects of normal operating pressure and expected external aerodynamic pressures.

(ii) The expected external aerodynamic pressures in 1 g flight combined with a cabin differential pressure equal to 1.1 times the normal operating differential pressure without any other load.

(4) The damage growth, between initial detectability and the value selected for residual strength demonstrations, factored to obtain inspection intervals, must allow development of an inspection program suitable for application by operation and maintenance personnel.

(5) For any bonded joint, the failure of which would result in catastrophic loss of the airplane, the limit load capacity must be substantiated by one of the following methods –

(i) The maximum disbonds of each bonded joint consistent with the capability to withstand the loads in paragraph (a)(3) of this section must be determined by analysis, tests, or both. Disbonds of each bonded joint greater than this must be prevented by design features; or

(ii) Proof testing must be conducted on each production article that will apply the critical limit design load to each critical bonded joint; or

(iii) Repeatable and reliable nondestructive inspection techniques must be established that ensure the strength of each joint.

(6) Structural components for which the damage tolerance method is shown to be impractical must be shown by component fatigue tests, or analysis supported by tests, to be able to withstand the repeated loads of variable magnitude expected in service. Sufficient component, subcomponent, element, or coupon tests must be

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done to establish the fatigue scatter factor and the environmental effects. Damage up to the threshold of detectability and ultimate load residual strength capability must be considered in the demonstration.

(b) Metallic airframe structure. If the applicant elects to use § 23.571(a)(3) {There is no § 23.571(a)(3) - Ed.} or § 23.572(a)(3), then the damage tolerance evaluation must include a determination of the probable locations and modes of damage due to fatigue, corrosion, or accidental damage. The determination must be by analysis supported by test evidence and, if available, service experience. Damage at multiple sites due to fatigue must be included where the design is such that this type of damage can be expected to occur. The evaluation must incorporate repeated load and static analyses supported by test evidence. The extent of damage for residual strength evaluation at any time within the operational life of the airplane must be consistent with the initial detectability and subsequent growth under repeated loads. The residual strength evaluation must show that the remaining structure is able to withstand critical limit flight loads, considered as ultimate, with the extent of detectable damage consistent with the results of the damage tolerance evaluations. For pressurized cabins, the following load must be withstood:

(1) The normal operating differential pressure combined with the expected external aerodynamic pressures applied simultaneously with the flight loading conditions specified in this part, and

(2) The expected external aerodynamic pressures in 1 g flight combined with a cabin differential pressure equal to 1.1 times the normal operating differential pressure without any other load.

(c) Inspection. Based on evaluations required by this section, inspections or other procedures must be established as necessary to prevent catastrophic failure and must be included in the Airworthiness Limitations section of the Instructions for Continued Airworthiness required by § 23.1529.

23.574 Metallic damage tolerance and fatigue evaluation of commuter category airplanes.

For commuter category airplanes -

(a) Metallic damage tolerance. An evaluation of the strength, detail design, and fabrication must show that catastrophic failure due to fatigue, corrosion, defects, or damage will be avoided throughout the operational life of the airplane. This evaluation must be conducted in accordance with the provisions of § 23.573, except as specified in paragraph (b) of this section, for each part of the structure that could contribute to a catastrophic failure.

(b) Fatigue (safe-life) evaluation. Compliance with the damage tolerance requirements of paragraph (a) of this section is not required if the applicant establishes that the

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application of those requirements is impractical for a particular structure. This structure must be shown, by analysis supported by test evidence, to be able to withstand the repeated loads of variable magnitude expected during its service life without detectable cracks. Appropriate safe-life scatter factors must be applied.

23.575 Inspections and other procedures.

Each inspection or other procedure, based on an evaluation required by §§ 23.571, 23.572, 23.573 or 23.574, must be established to prevent catastrophic failure and must be included in the Limitations Section of the Instructions for Continued Airworthiness required by § 23.1529.

4.1.2 Part 25 - Airworthiness Standard: Transport Category Airplanes.

25.571 Damage tolerance and fatigue evaluation of structure.

(a) General. An evaluation of the strength, detail design, and fabrication must show that catastrophic failure due to fatigue, corrosion, or accidental damage, will be avoided throughout the operational life of the airplane. This evaluation must be conducted in accordance with the provisions of paragraphs (b) and (e) of this section, except as specified in paragraph (c) of this section, for each part of the structure which could contribute to a catastrophic failure (such as wing, empennage, control surfaces and their systems, the fuselage, engine mounting, landing gear, and their related primary attachments). Advisory Circular AC No. 25.571-1 contains guidance information relating to the requirements of this section (copies of the advisory circular may be obtained from the U.S. Department of Transportation, Publications Section M443.1, Washington, D.C. 20590). For turbojet powered airplanes, those parts which could contribute to a catastrophic failure must also be evaluated under paragraph (d) of this section. In addition, the following apply:

(1) Each evaluation required by this section must include –

(i) The typical loading spectra, temperatures, and humidities expected in service;

(ii) The identification of principal structural elements and detail design points, the failure of which could cause catastrophic failure of the airplane; and

(iii) An analysis, supported by test evidence, of the principal structural elements and detail design points identified in paragraph (a)(1)(ii) of this section.

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(2) The service history of airplanes of similar structural design, taking due account of differences in operating conditions and procedures, may be used in the evaluations required by this section.

(3) Based on the evaluations required by this section, inspections or other procedures must be established as necessary to prevent catastrophic failure, and must be included in the Airworthiness Limitations section of the Instruction for Continued Airworthiness required by § 25.1529.

(b) Damage tolerance evaluation. The evaluation must include a determination of the probable locations and modes of damage due to fatigue, corrosion, or accidental damage. The determination must be by analysis supported by test evidence and (if available) service experience. Damage at multiple sites due to prior fatigue exposure must be included where the design is such that this type of damage can be expected to occur. The evaluation must incorporate repeated load and static analyses supported by test evidence. The extent of damage for residual strength evaluation at any time within the operational life must be consistent with the initial detectability and subsequent growth under repeated loads. The residual strength evaluation must show that the remaining structure is able to withstand loads (considered as static ultimate loads) corresponding to the following conditions:

(1) The limit symmetrical maneuvering conditions specified in § 25.337 at VC and in § 25.345.

(2) The limit gust conditions specified in § 25.341 at the specified speeds up to VC and in § 25.345.

(3) The limit rolling conditions specified in § 25.349 and the limit unsymmetrical conditions specified in §§ 25.367 and 25.427 (a) through (c), at speeds up to VC.

(4) The limit yaw maneuvering conditions specified in § 25.351(a) at the specified speeds up to VC.

(5) For pressurized cabins, the following conditions:

(i) The normal operating differential pressure combined with the expected external aerodynamic pressures applied simultaneously with the flight loading conditions specified in paragraphs (b) (1) through (4) of this section, if they have a significant effect.

(ii) The expected external aerodynamic pressures in 1 g flight combined with a cabin differential pressure equal to 1.1 times the normal operating differential pressure without any other load.

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(6) For landing gear and directly affected airframe structure, the limit ground loading conditions specified in §§ 25.473, 25.491, and 25.493.

If significant changes in structural stiffness or geometry, or both, follow from a structural failure, or partial failure, the effect on damage tolerance must be further investigated.

(c) Fatigue (safe life) evaluation. Compliance with the damage tolerance requirements of paragraph (b) of this section is not required if the applicant establishes that their application for particular structure is impractical. This structure must be shown by analysis, supported by test evidence, to be able to withstand the repeated loads of variable magnitude expected during its service life without detectable cracks. Appropriate safe life scatter factors must be applied.

(d) Sonic fatigue strength. It must be shown by analysis, supported by test evidence, or by the service history of airplanes of similar structural design and sonic excitation environment, that –

(1) Sonic fatigue cracks are not probable in any part of the flight structure subject to sonic excitation; or

(2) Catastrophic failure caused by sonic cracks is not probable assuming that the loads prescribed in paragraph (b) of this section are applied to all areas affected by those cracks.

(e) Damage tolerance (discrete source) evaluation. The airplane must be capable of successfully completing a flight during which likely structural damage occurs as a result of –

(1) Impact with a 4 pound bird at VC at sea level to 8,000 feet;
(2) Uncontained fan blade impact;
(3) Uncontained engine failure; or
(4) Uncontained high energy rotating machinery failure.

The damaged structure must be able to withstand the static loads (considered as ultimate loads) which are reasonably expected to occur on the flight. Dynamic effects on these static loads need not be considered. Corrective action to be taken by the pilot following the incident, such as limiting maneuvers, avoiding turbulence, and reducing speed, must be considered. If significant changes in structural stiffness or geometry, or both, follow from a structural failure or partial failure, the effect on damage tolerance must be further investigated.

4.1.3 Part 27 - Airworthiness Standards: Normal Category Rotorcraft.

27.571 Fatigue evaluation of flight structure.

(a) General. Each portion of the flight structure (the flight structure includes rotors, rotor drive systems between the engines and the rotor hubs, controls, fuselage, landing gear,

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and their related primary attachments), the failure of which could be catastrophic, must be identified and must be evaluated under paragraph (b), (c), (d), or (e) of this section. The following apply to each fatigue evaluation:

(1) The procedure for the evaluation must be approved.
(2) The locations of probable failure must be determined.
(3) In-flight measurement must be included in determining the following:

(i) Loads or stresses in all critical conditions throughout the range of limitations in § 27.309, except that maneuvering load factors need not exceed the maximum values expected in operation.

(ii) The effect of altitude upon these loads or stresses.

(4) The loading spectra must be as severe as those expected in operation including, but not limited to, external cargo operations, if applicable, and ground/air/ground cycles. The loading spectra must be based on loads or stresses determined under paragraph (a)(3) of this section.

(b) Fatigue tolerance evaluation. It must be shown that the fatigue tolerance of the structure ensures that the probability of catastrophic fatigue failure is extremely remote without establishing replacement times, inspection intervals or other procedures under section 27xA.4 of Appendix A.

(c) Replacement time evaluation. it must be shown that the probability of catastrophic fatigue failure is extremely remote within a replacement time furnished under section 27xA.4 of Appendix A.

(d) Fail-safe evaluation. The following apply to fail-safe evaluation:

(1) It must be shown that all partial failures will become readily detectable under inspection procedures furnished under section 27xA.4 of Appendix A.

(2) The interval between the time when any partial failure becomes readily detectable under paragraph (d)(1) of this section, and the time when any such failure is expected to reduce the remaining strength of the structure to limit or maximum attainable loads (whichever is less), must be determined.

(3) It must be shown that the interval determined under paragraph (d)(2) of this section is long enough, in relation to the inspection intervals and related procedures furnished under section 27xA.4 of Appendix A, to provide a probability of detection great enough to ensure that the probability of catastrophic failure is extremely remote.

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(e) *Combination of replacement time and fail-safe evaluations. A component may be evaluated under a combination of paragraphs (c) and (d) of this section. For such component it must be shown that the probability of catastrophic failure is extremely remote with an approved combination of replacement time, inspection intervals, and related procedures furnished under section 27xA.4 of Appendix A.*

4.1.4 Part 29 - Airworthiness Standard: Transport Category Rotorcraft.

29.571 Fatigue evaluation of structure.

(a) *General. An evaluation of the strength of principal elements, detail design points, and fabrication techniques must show that catastrophic failure due to fatigue, considering the effects of environment, intrinsic/discrete flaws, or accidental damage will be avoided. Parts to be evaluated include, but are not limited to, rotors, rotor drive systems between the engines and rotor hubs, controls, fuselage, fixed and movable control surfaces, engine and transmission mountings, landing gear, and their related primary attachments. In addition, the following apply:*

(1) *Each evaluation required by this section must include –*

(i) *The identification of principal structural elements, the failure of which could result in catastrophic failure of the rotorcraft;*

(ii) *In-flight measurement in determining the loads or stresses for items in paragraph (a)(1)(i) of this section in all critical conditions throughout the range of limitations in § 29.309 (including altitude effects), except that maneuvering load factors need not exceed the maximum values expected in operations; and*

(iii) *Loading spectra as severe as those expected in operation based on loads or stresses determined under paragraph (a)(1)(ii) of this section, including external load operations, if applicable, and other high frequency power cycle operations.*

(2) *Based on the evaluations required by this section, inspections, replacement times, combinations thereof, or other procedures must be established as necessary to avoid catastrophic failure. These inspections, replacement times, combinations thereof, or other procedures must be included in the airworthiness limitations section of the Instructions for Continued Airworthiness required by § 29.1529 and section 29xA.4 of Appendix A of this part.*

(b) *Fatigue tolerance evaluation (including tolerance to flaws). The structure must be shown by analysis supported by test evidence and, if available, service experience to be of fatigue tolerant design. The fatigue tolerance evaluation must include the requirements of either paragraph (b) (1), (2), or (3) of this section, or a combination thereof, and also*

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must include a determination of the probable locations and modes of damage caused by fatigue, considering environmental effects, intrinsic/discrete flaws, or accidental damage. Compliance with the flaw tolerance requirements of paragraph (b) (1) or (2) of this section is required unless the applicant establishes that these fatigue flaw tolerant methods for a particular structure cannot be achieved within the limitations of geometry, inspectability, or good design practice. Under these circumstances, the safe-life evaluation of paragraph (b)(3) of this section is required.

(1) Flaw tolerant safe-life evaluation. It must be shown that the structure, with flaws present, is able to withstand repeated loads of variable magnitude without detectable flaw growth for the following time intervals –

(i) Life of the rotorcraft; or

(ii) Within a replacement time furnished under section 29xA.4 of appendix A to this part.

(2) Fail-safe (residual strength after flaw growth) evaluation. It must be shown that the structure remaining after a partial failure is able to withstand design limit loads without failure within an inspection period furnished under section 29xA.4 of appendix A to this part. Limit loads are defined in § 29.301(a).

(i) The residual strength evaluation must show that the remaining structure after flaw growth is able to withstand design limit loads without failure within its operational life.

(ii) Inspection intervals and methods must be established as necessary to ensure that failures are detected prior to residual strength conditions being reached.

(iii) If significant changes in structural stiffness or geometry, or both, follow from a structural failure or partial failure, the effect on flaw tolerance must be further investigated.

(3) Safe-life evaluation. It must be shown that the structure is able to withstand repeated loads of variable magnitude without detectable cracks for the following time intervals –

(i) Life of the rotorcraft; or

(ii) Within a replacement time furnished under section 29xA.4 of appendix A to this part.

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4.2 ADVISORY CIRCULAR RECOMMENDATIONS.

The following excerpt is taken directly from FAA Advisory Circular (AC) 20-107A which provides special guidelines for composite airframes as a means to show compliance with applicable FAA regulations. The portion of AC 20-107A presented below specifically addresses issues regarding damage tolerance.

The evaluation of composite structure should be based on the applicable requirements of FAR 23.571, 23.572, 25.571, 27.571, and 29.571. The nature and extent of analysis or tests on complete structures and/or portions of the primary structure will depend upon applicable previous fatigue/damage tolerant designs, construction, tests, and service experience on similar structures. In the absence of experience with similar designs, FAA-approved structural development tests of components, subcomponents, and elements should be performed. The following considerations are unique to the use of composite material systems and should be observed for the method of substantiation selected by the applicant. When selecting the damage tolerance or safe life approach, attention should be given to geometry, inspectability, good design practice, and the type of damage/ degradation of the structure under consideration.

Damage Tolerance (Fail-Safe) Evaluation

(1) Structural details, elements, and subcomponents of critical structural areas should be tested under repeated loads to define the sensitivity of the structure to damage growth. This testing can form the basis for validating a no growth approach to the damage tolerance requirements. The testing should assess the effect of the environment on the flaw growth characteristics and the no growth validation. The environment used should be appropriate to the expected service usage. The repeated loading should be representative of anticipated service usage. The repeated load testing should include damage levels (including impact damage) typical of those that may occur during fabrication, assembly, and in service, consistent with the inspection techniques employed. The damage tolerance test articles should be fabricated and assembled in accordance with production specifications and processes so that the test articles are representative of production structure.

(2) The extent of initially detectable damage should be established and be consistent with the inspection techniques employed during manufacture and in service. Flaw/damage growth data should be obtained by repeated load cycling of intrinsic flaws or mechanically introduced damage. The number of cycles applied to validate a no growth concept should be statistically significant, and may be determined by load and/or life considerations. The growth or no growth evaluation should be performed by analysis supported by test evidence or by tests at the coupon, element, or subcomponent level.

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(3) The extent of damage for residual strength assessments should be established. Residual strength evaluation by component or subcomponent testing or by analysis supported by test evidence should be performed considering that damage. The evaluation should demonstrate that the residual strength of the structure is equal to or greater than the strength required for the specified design loads (considered as ultimate). It should be shown that stiffness properties have not changed beyond acceptable levels. For the no growth concept residual strength testing should be performed after repeated load cycling.

(4) An inspection program should be developed consisting of frequency, extent, and methods of inspection for inclusion in the maintenance plan. Inspection intervals should be established such that the damage will be detected between the time it initially becomes detectable and the time at which the extent of damage reaches the limits for required residual strength capability. For the case of no growth design concept, inspection intervals should be established as part of the maintenance program. In selecting such intervals the residual strength level associated with the assumed damages should be considered.

(5) The structure should be able to withstand static loads (considered as ultimate loads) which are reasonably expected during a completion of the flight on which damage resulting from obvious discrete sources occur (i.e., uncontained engine failures, etc.). The extent of damage should be based on a rational assessment of service mission and potential damage relating to each discrete source.

(6) The effects of temperature, humidity, and other environmental factors which may result in material property degradation should be addressed in the damage tolerance evaluation.

Fatigue (Safe-Life) Evaluation

Fatigue substantiation should be accomplished by component fatigue tests or by analysis supported by test evidence, accounting for the effects of the appropriate environment. The test articles should be fabricated and assembled in accordance with production specifications and processes so that the test articles are representative of production structure. Sufficient component, subcomponent, element or coupon tests should be performed to establish the fatigue scatter and the environmental effects. Component, subcomponent, and/or element tests may be used to evaluate the fatigue response of structure with impact damage levels typical of those that may occur during fabrication, assembly, and in service, consistent with the inspection procedures employed. The component fatigue test may be performed with an as-manufactured test article if the effects of impact damage are reliably predicted by subcomponent and/or element tests and are

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accounted for in the fatigue test or in analysis of the results of the fatigue test. It should be demonstrated during the fatigue tests that the stiffness properties have not changed beyond acceptable levels. Replacement lives should be established based on the test results. An appropriate inspection program should be provided.

4.3 PREVIOUS CERTIFICATION APPROACHES, DAMAGE TOLERANCE – BEECH STARSHIP.

4.3.1 Element (Coupon) Static Tests (For Design).

Test specimens ranged in width between 4 and 8 inches, depending upon the particular mode of loading. The barely visible impact damage was done with a 3/4-inch-diameter spherical indentor, and the detectable puncture was done through one face sheet with a 1/2-inch by 1/2-inch crucifix punch.

Longitudinal Tension–top of fuselage:

- Flat test specimens with layup configurations representative of the fuselage and with barely visible impact damage were required to withstand ultimate load.
- Flat test specimens with layup configurations representative of the fuselage and with detectable puncture damage were required to withstand limit load.

Shear–side of fuselage:

- Flat test specimens with layup configurations representative of the fuselage and with barely visible impact damage were required to withstand ultimate load.
- Flat test specimens with layup configurations representative of the fuselage and with detectable puncture damage were required to withstand limit load.

Compression–bottom of fuselage:

- Flat test specimens with layup configurations representative of the fuselage and with barely visible impact damage were required to withstand ultimate load.
- Flat test specimens with layup configurations representative of the fuselage and with detectable puncture damage were required to withstand limit load.

Hoop Tension–top of fuselage:

- Cylindrical (about 1/3 scale: 24″ diameter) test specimens with layup configurations representative of the fuselage and with barely visible impact damage were required to withstand ultimate cabin pressure.

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- Cylindrical (about 1/3 scale: 24″ diameter) test specimens with layup configurations representative of the fuselage and with detectable puncture damage were required to withstand approximately 60% of ultimate cabin pressure [1.1(operating pressure) + aerodynamic vacuum].

Water Ingression Tests:

Twelve-inch-square panels with inflicted punctures of one face sheet were immersed in water to allow water into the core in the punctured regions. They were then subjected to freeze/thaw cycles with vacuum applied during freeze to simulate high altitude flight and then inspected to ensure that water did not propagate beyond the punctured regions.

Lightning Strike Tests:

Twenty-four-inch square panels were subjected to electrically simulated lightning strike to determine the degree of damage. This degree of damage (considered as larger damage that only occasionally happens) was then simulated mechanically in large components for full-scale testing described later. This same kind of testing was also done with 24-inch panels with joints representative of actual joints in the aircraft.

<u>4.3.2 Element Cyclic Tests (For Verification)</u>.

Longitudinal tension–top of fuselage:

- Flat test specimens with layup configurations representative of the fuselage and with barely visible impact damage were subjected to constant-amplitude cyclic loading to establish sensitivity to flaw growth.

- Flat test specimens with layup configurations representative of the fuselage and with detectable puncture damage were subjected to constant-amplitude cyclic loading to establish sensitivity to flaw growth.

Compression–bottom of fuselage:

- Flat test specimens with layup configurations representative of the fuselage and with barely visible impact damage were subjected to constant-amplitude cyclic loading (at three different stress levels) to establish sensitivity to flaw growth.

- Flat test specimens with layup configurations representative of the fuselage and with barely visible impact damage were subjected to spectrum loading (representing lifetime varying amplitude loads) to establish sensitivity to flaw growth.

- Flat test specimens with layup configurations representative of the fuselage and with detectable puncture damage were subjected to constant-amplitude cyclic loading (at three different stress levels) to establish sensitivity to flaw growth.

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- Flat test specimens with layup configurations representative of the fuselage and with detectable puncture damage were subjected to spectrum loading (representing lifetime varying amplitude loads) to establish sensitivity to flaw growth.

Shear–side of fuselage:

- Flat test specimens with layup configurations representative of the fuselage and with barely visible impact damage were subjected to constant-amplitude cyclic loading (at three different stress levels) to establish sensitivity to flaw growth.

- Flat test specimens with layup configurations representative of the fuselage and with detectable puncture damage were subjected to constant-amplitude cyclic loading (at three different stress levels) to establish sensitivity to flaw growth.

4.3.3 Large Subcomponent Static Tests (For Risk Reduction).

Forward fuselage and windshield (undamaged) was required to meet the following conditions (in hot/wet environment):

- Carry limit bending load, followed by ultimate bending load.

- Carry the combination of limit bending load and maximum operating cabin pressure, followed by the combination of ultimate bending load and 1.5 times maximum operating cabin pressure.

- Carry 1.33 times maximum operating pressure, followed by 1.5(1.33) or 2 times maximum operating pressure.

Wing box (undamaged) was required to carry limit bending load, followed by ultimate bending load (in hot/wet environment).

Wing box and fuselage intersection (undamaged) was required to carry limit bending load, followed by ultimate bending load (in room temperature dry environment).

Windshield: The undamaged windshield was subjected to cyclic pressure testing for the equivalent of one lifetime. Mechanical damage was then inflicted and the windshield was subjected to an additional lifetime of cyclic pressure testing. Finally, the windshield was statically tested and required to carry fail-safe pressure.

Windshield: The windshield with electrically simulated lightning strike was tested statically and required to carry fail-safe pressure.

4.3.4 Full-Scale Cyclic Test (For Verification).

Components (fuselage, wing, tail) were subjected to spectrum cyclic loading representing two lifetimes of expected missions. During these tests, damage was mechanically inflicted to

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simulate in-service damage including lightning strike, hail damage, runway damage, and tool impacts. These damage modes were tested through at least one lifetime of testing to verify that damage would not grow in an unpredictable manner and would always be detected by the specified inspections.

Larger damage was inflicted to simulate impacts with ground service equipment, hanger doors, and other aircraft and poor maintenance practices and lightning strike. These larger modes of damage (that only occasionally happen) will be detected before the next flight; thus the demonstration for these modes consisted of relatively few cycles and were also included in the residual strength tests.

4.3.5 Full-Scale Static Testing (For Verification).

After the completion of the full-scale cyclic testing, the major components were subjected to static load tests to verify that the structure would still carry the required residual strength loads (flight loads and or pressure loads expected to be encountered during the service life of the aircraft) in spite of all the cyclic loading and the inflicted damage. The larger damage modes had different residual strength requirements than the other damage modes.

4.3.6 Pre In-Service Inspection (For Initial Quality).

Acceptance criteria were established for structure with porosity, voids, and disbonds to account for initial quality (flaws) developed during the manufacturing process. Damage modes such as porosity, voids, and disbonds were subjected to specified acceptance criteria. This initial quality is intrinsic to the manufacturing process and the inspection standards and represent the as-delivered state, and therefore, the structure must be capable of meeting all requirements of strength, stiffness, safety, and longevity with this initial quality.

4.3.7 Post In-Service Inspection (For Continued Airworthiness).

Based on interpretation of the test results, (1) inspection procedures, (2) threshold time, and (3) frequency of inspections were established and published in the airplane manuals. A factor was applied so that allowance was made for the damage to exist over several inspection intervals, depending on the criticality of the structure. (A further factor may be needed to account for scatter revealed in the flaw growth test results.)

5. SURVEY OF CURRENT COMPOSITE SANDWICH CONSTRUCTION.

The survey described is this section is meant to provide a listing of companies utilizing composite sandwich construction techniques for primary load-bearing structure. The listing for each company is the future, current, or past production model designation; aircraft type; general specifications; FAR certification category; and details regarding the manufacturing and layup configuration. This section is meant to provide a guideline for further studies into damage tolerance evaluations which would be reflective of current sandwich construction techniques and applications utilized in industry. This listing is restricted to aircraft companies using composite sandwich construction techniques in the fuselage only and does not include other applications such as control surfaces and horizontal/vertical stabilizers.

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5.1 RAYTHEON AIRCRAFT COMPANY.

5.1.1 Beech Starship.

Engine Type:	Turboprop
Maximum Cruise Speed:	386 mph
Certified Ceiling:	41,000 ft
Maximum Range:	1,576 nm
Empty Weight:	10,320 lb
Gross Weight:	14,900 lb
Wingspan:	54.4 ft
Length:	46.1 ft
Height:	12.9 ft
Wing Area:	280.8 ft^2
Certification Category:	FAR Part 23
Number of Seats (typical):	10 (including pilots)
Pressurized:	Yes
Facesheet Material Type:	Graphite Epoxy
Nominal Facesheet Thickness:	0.030 in (symmetric)
Core Type:	Aramid Fiber Honeycomb (Nomex®) - 3/16″ cell size
Nominal Core Thickness:	3/4″
Core Density:	3 lb/ft^3
Film Adhesive:	Yes
Manufacturing Method:	hand layup–autoclave cure
Layup Configuration:	Reference figure 20

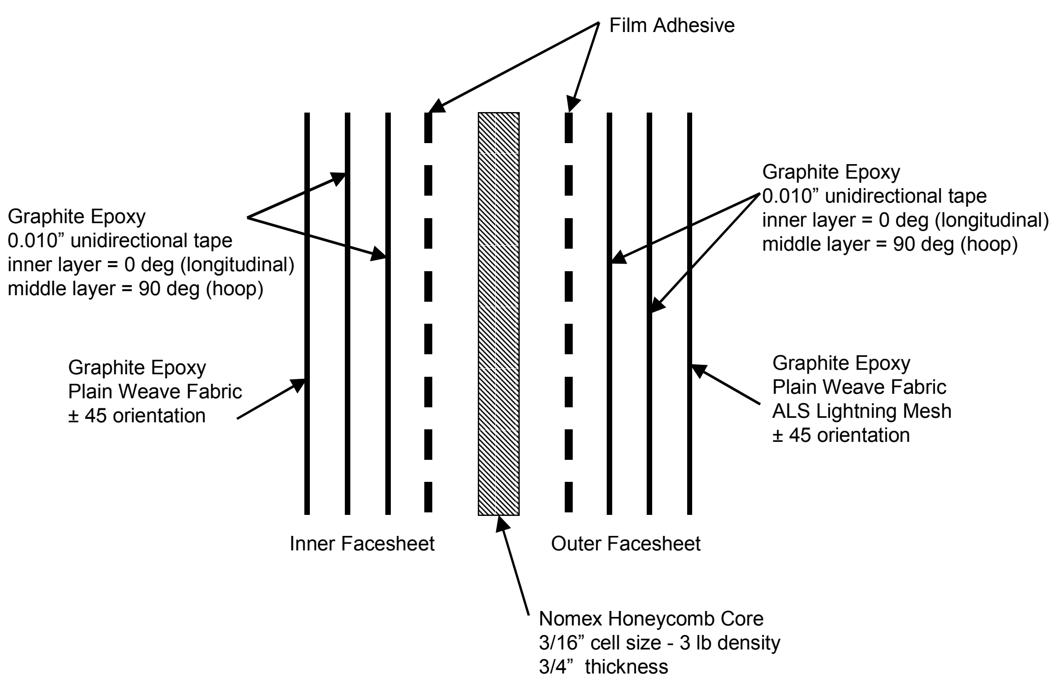

FIGURE 20. TYPICAL LAYUP CONFIGURATION FOR BEECH STARSHIP

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5.1.2 Premier 1.

Engine Type:	Turbofan
Maximum Cruise Speed:	531 mph
Certified Ceiling:	41,000 ft
Maximum Range:	1,500 nm
Empty Weight:	7,300 lb
Gross Weight:	12,500 lb
Wingspan:	44.5 ft
Length:	45.3 ft
Height:	15.0 ft
Wing Area:	212 ft^2
Certification Category:	FAR Part 23
Number of Seats (typical):	7 (including pilots)
Pressurized:	Yes
Facesheet Material Type:	Graphite Epoxy
Nominal Facesheet Thickness:	0.025 in (symmetric)
Core Type:	Aramid Fiber Honeycomb (Nomex®) - 3/16″ cell size
Nominal Core Thickness:	3/4″
Core Density:	3 lb/ft^3
Film Adhesive:	Yes
Manufacturing Method:	automated fiber placement–autoclave cure
Layup Configuration:	Reference figure 21

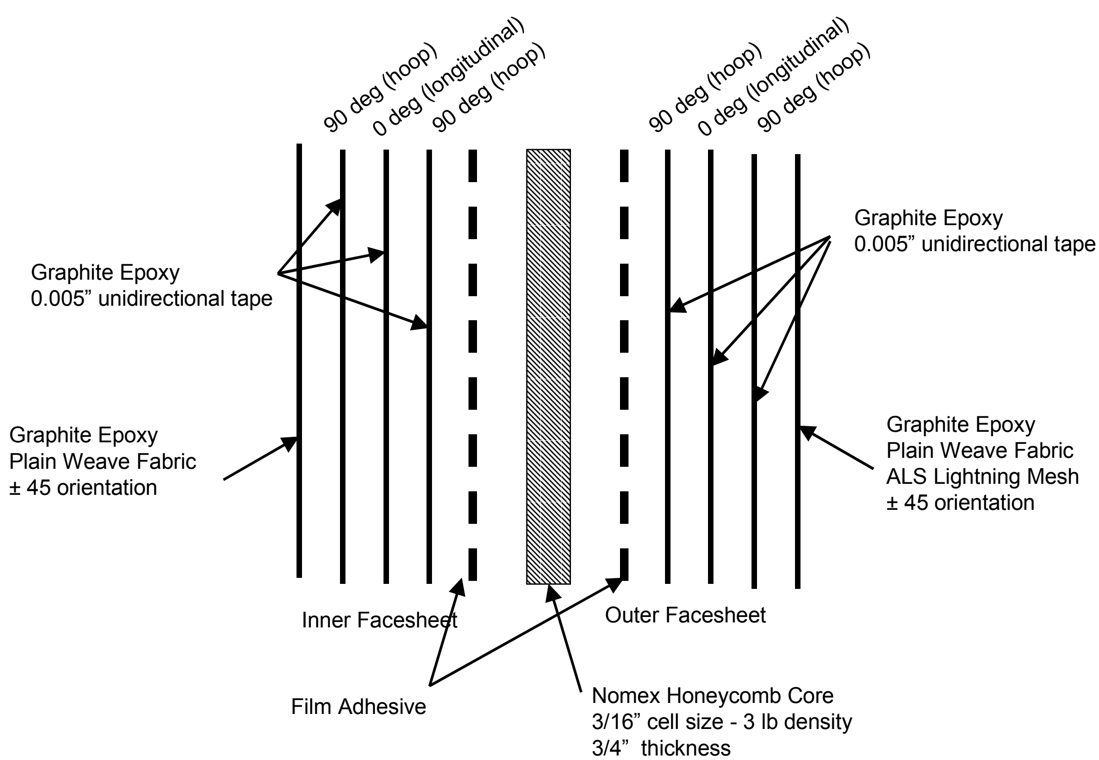

FIGURE 21. TYPICAL LAYUP CONFIGURATION FOR PREMIER I

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5.1.3 Hawker Horizon.

Engine Type:	Turbofan
Maximum Cruise Speed:	557 mph
Certified Ceiling:	45,000 ft
Maximum Range:	3,400 nm
Empty Weight:	20,930 lbs
Gross Weight:	36,200 lbs
Wingspan:	61.8 ft
Length:	69.2 ft
Height:	19.7 ft
Wing Area:	531 ft^2
Certification Category:	FAR Part 25
Number of Seats (typical):	10-16 (including pilots)
Pressurized:	Yes
Facesheet Material Type:	Graphite Epoxy
Nominal Facesheet Thickness:	TBD
Core Type:	Aramid Fiber Honeycomb (Nomex®) - 3/16″ cell size
Nominal Core Thickness:	1″
Core Density:	3 lb/ft^3
Film Adhesive:	Yes
Manufacturing Method:	automated fiber placement–autoclave cure
Layup Configuration:	To be determined

5.2 CIRRUS DESIGN.

Designation:	SR20
Engine Type:	Reciprocating Engine-Powered
Maximum Cruise Speed:	184 mph
Maximum Range:	800 nm
Empty Weight:	1,800 lbs
Gross Weight:	2,900 lbs
Wingspan:	35.6 ft
Length:	26.3 ft
Height:	9.3 ft
Wing Area:	135 ft^2
Certification Category:	FAR Part 23
Number of Seats (typical):	4 (including pilots)
Pressurized:	No
Facesheet Material Type:	E-Glass Epoxy (7781 Fabric)
Nominal Facesheet Thickness:	0.020 in (symmetric)
Core Type:	Divinycell® Foam
Nominal Core Thickness:	3/8″
Core Density:	4.4 lb/ft^3
Film Adhesive:	No

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Manufacturing Method:	hand layup–oven cure
Layup Configuration:	Reference figure 22

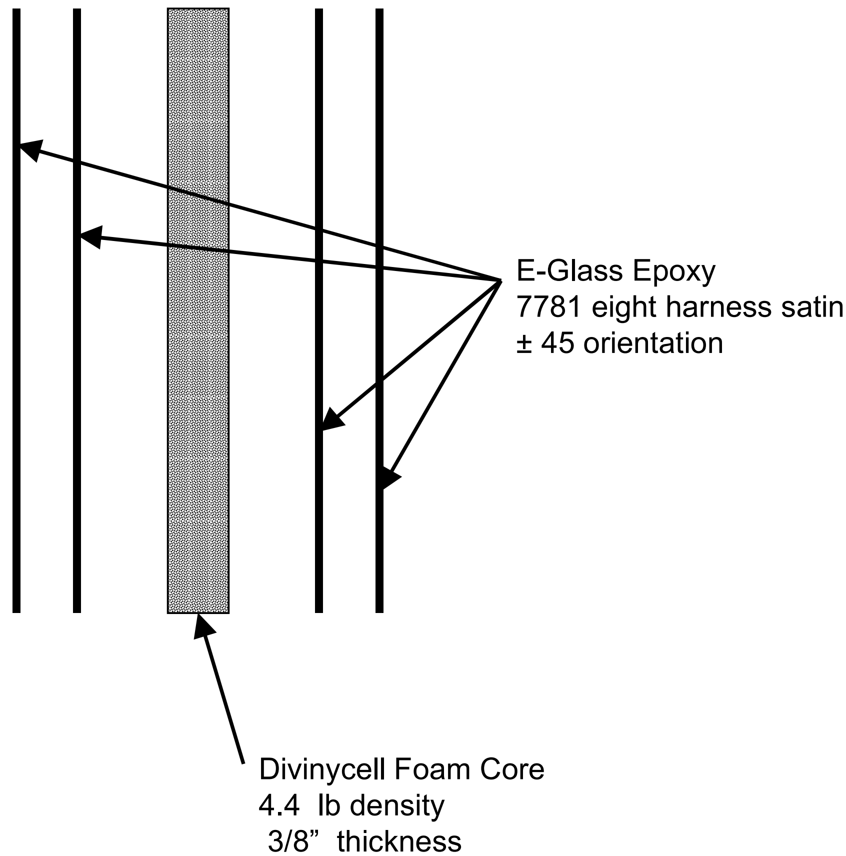

FIGURE 22. TYPICAL LAYUP CONFIGURATION FOR SR20

5.3 LANCAIR PAC/USA.

Designation:	Columbia 300
Engine Type:	Reciprocating Engine Powered
Maximum Cruise Speed:	220 mph
Certified Ceiling:	18,000 ft
Maximum Range:	800 nm
Empty Weight:	2,045 lbs
Gross Weight:	3,400 lbs
Wingspan:	36.1 ft
Length:	25.1 ft
Height:	9 ft
Wing Area:	142 ft^2
Certification Category:	FAR Part 23
Number of Seats (typical):	4 (including pilots)
Pressurized:	No
Facesheet Material Type:	E-Glass Epoxy (7781 Fabric)

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Nominal Facesheet Thickness:	0.020 in
Core Type:	Aramid Fiber Honeycomb (Nomex®) - 3/16″ cell size
Nominal Core Thickness:	1/4″, 3/8″ (majority), 1/2″
Core Density:	3 lb/ft^3
Film Adhesive:	No
Manufacturing Method:	hand layup–oven cure
Layup Configuration:	Reference figure 23

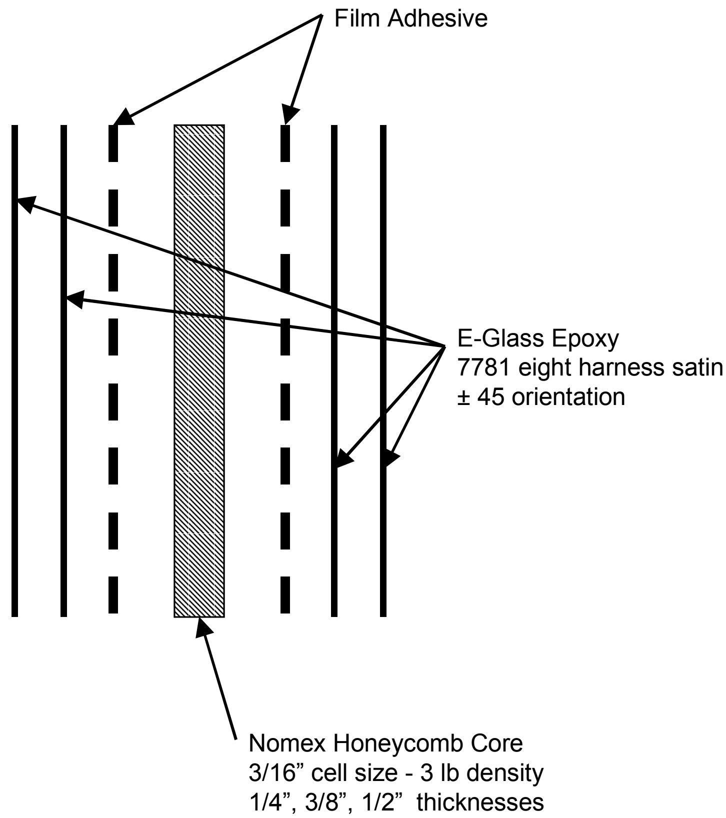

FIGURE 23. TYPICAL LAYUP CONFIGURATION FOR COLUMBIA 300

5.4 BOEING PHILADELPHIA.

The design of sandwich structures at Boeing Philadelphia is empirically based. Compression has been determined as the dominant loading condition. Although in-plane shear can be the dominant critical loading, the resulting compressive load initiates failure. Under flexural loads the strength of the facesheet in compression is determined from in-plane compression allowables

Boeing's standard specimen is 3.25″ wide and approximately 14″ long with cocured facesheets. A three-part 1″ thick honeycomb structure is used with high-density aluminum honeycomb at both ends to enable gripping of the specimen. The specimen is then loaded in compression until failure. Initially specimens approximately 8″ in width were manufactured and impacted. After impact the specimens were machined to the 3.25″ width. Subsequently larger width panels were

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manufactured and impacted at five locations across the width. This larger panel was then machined into five specimens each 3.25″ in width. In all cases impact is done with clamped ends and free edges. Barely visible impact damage is characterized often by the depth of the resulting indentation.

Fatigue of sandwich structures with damage is not a key concern because of design allowable at 80% of mean less 3σ. In all cases, specimens reached the runout limit of typically 2.5 million cycles in a hot-wet environment thought to be conservative. The ability to use compressive strength for shear requirements was confirmed by testing 24″ by 24″ sandwich panels in a picture frame fixture. The resulting failure mode and failure load correlated well with compression results.

Key areas of concern are scaling effects (width and curvature) and multiple or repeated impacts (e.g., hail storms)

5.5 SIKORSKY AIRCRAFT.

Again the design of sandwich structures is empirically based. Compression after impact provides a design allowable.

Sikorsky's standard specimen is 6″ by 6″ with a 1″ core. The core is potted with resin approximately 1″ from both ends and then the facesheets are laid up and cocured. The specimen is supported around all ends and edges allowing a circular impact area to be revealed. The specimen is then impacted and loaded in compression. Clamps are applied to both ends to prevent brooming and to distribute the load.

Facesheet wrinkling is seen as the controlling mechanism in thin-gauge sandwich structures. Strength of undamaged sandwich structures is likened to plates with out-of-plane wrinkles on an elastic foundation by extending classical formulas. Some aspects of the classical formulas are not well understood when applied to advanced materials. For example, an increase in the rigidity of the core from Nomex to aluminum ought to have resulted in a greater strength. However, the strength was reduced and unpredicted core failure occurred.

Fatigue is handled by not exceeding 36% of the damaged strength for a lifetime of 10^6 cycles and by not exceeding 31% for an infinite lifetime (>10^8 cycles). This is based on an extrapolation of S-N curves out from 10^7 cycles assuming an endurance limit. Sikorsky feels that fatigue is not a laminate level concern in Comanche. The question remains, however, whether or not the above truncation limits apply to the core and adhesive.

Key areas of concern are the isolation and prediction of individual failure modes. The ability to redirect the load around damage by fiber steering is of interest. The exact redistribution of the load in-plane and through the back facesheet is unexplored.

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6. SUGGESTED APPROACH AND METHODOLOGY.

The long-term objectives of any program should clarify key safety issues pertaining to the damage tolerance of composite sandwich airframe structures including residual strength and damage propagation. Methodology should be developed to assess the adequacy of sandwich designs to meet FAA safety requirements. Some special attention on pressurized fuselage structure may be warranted.

The determination of the damage tolerance characteristics of sandwich panels has been limited in previous investigations to relatively few sandwich configurations and damage states. The purpose of any future investigations should determine the effect of a wide variety of impact damage states on the damage tolerance characteristics of composite sandwich panels over a significant range of panel configurations commonly used in aerospace applications. These efforts should provide results that will be beneficial in achieving damage tolerance for the safe application of composite sandwich structure. They should also be useful in engineering design to satisfy relevant FAA certification requirements and in support of safe maintenance practices.

The general philosophy applied during damage tolerance certification, shown schematically in figure 24, relates representative damage size to design load requirements. As in the case of metal aircraft, ultimate strength and damage tolerance philosophies are used to maintain a reliable and safe operation of composite structures. As shown in figure 24, this philosophy may typically be described using three distinct regions and may be summarized as follows:

a. Nonvisible or Barely Visible Impact Damage (BVID) or defects that are not detectable during manufacturing inspections and service inspections must withstand ultimate load and not impair operation of the aircraft for its lifetime. In this region, it is assumed that the damage may never be discovered during the aircraft's lifetime and must support ultimate design load.

b. Once the damage is observed, which is larger than the Allowable Damage Limit (ADL), the damage must be repaired when discovered. This damage, which is detectable using selected service inspections, must survive repeated loads for the specified inspection interval (with high confidence) and must withstand a once per lifetime load (Limit Load). It is necessary in a damage tolerant design that service damage falling in this region be found and characterized using practical inspection techniques. Design requirements and objectives for immediately obvious damage are also common. Consideration of this type of damage promotes fail safety by demonstrating Limit Load capability, but has no need to demonstrate tolerance to repeated loads (i.e., immediately obvious damage should be found within a small number of flights).

c. The last region covers damage associated with a discrete source event. Usually, this damage occurs in flight and is apparent to the operator. Under this condition, the Critical Damage Threshold (CDT) may be exceeded and the aircraft must withstand limited maneuvers, with and without pressure, as necessary for safe flight under Federal Regulations. Depending on the category and configuration of aircraft, there are specific discrete source damage requirements for birdstrike and engine burst.

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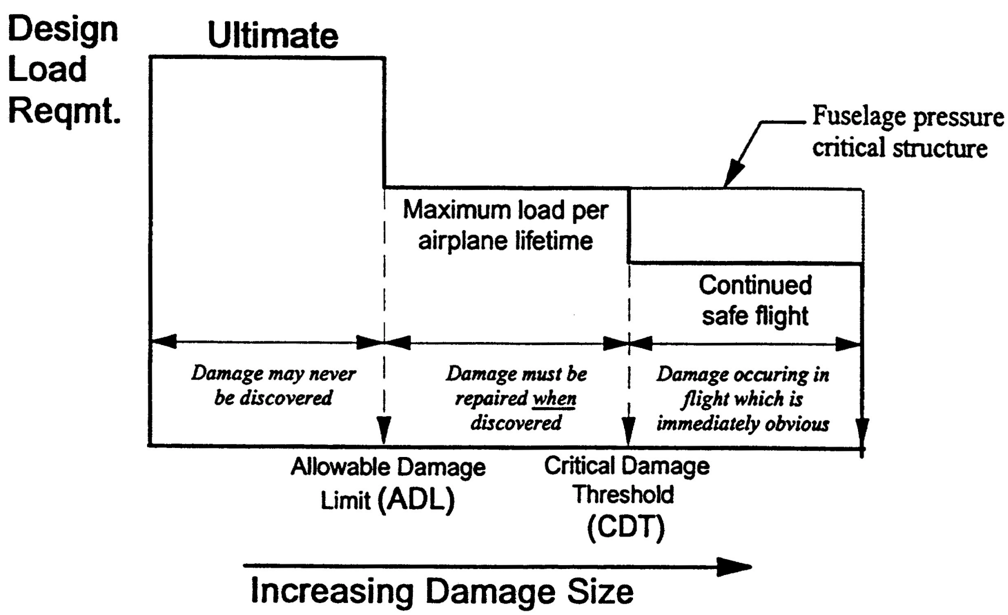

FIGURE 24. DAMAGE TOLERANT DESIGN PHILOSOPHY

6.1 OBJECTIVES.

The investigation of the damage resistance and tolerance characteristics of sandwich structures is summarized in five major proposed tasks. The experimental observations should be used for developing and validating a semiempirical model to predict the damage resistance and tolerance capabilities for a given sandwich panel configuration.

6.2 TASK 1: DAMAGE FORMATION IN SANDWICH STRUCTURES SUBJECTED TO LOW-VELOCITY IMPACT.

In this task, impact damage development in sandwich composites should be investigated experimentally. The effects of a number of variables should be studied during this task. Although the focus of this task is primarily associated with mechanically inflicted damage, the overall goal of the proposed investigation should be damage tolerance. Characteristic damage sizes associated with a given impact event and panel configuration will serve as one basis for assessing residual strength degradation and damage growth in task 2 and task 3, respectively. The investigation range for extrinsic variables should be chosen based on the existing literature and along the guidelines of NASA-RP-1092 [50].

The impact energy and velocity, impactor material, and impactor diameter ranges summarized previously should form the basis for preliminary tests that will be used for the detailed damage tolerance tests on candidate materials. The damage induced in sandwich panels impacted at various impact energies and velocities should be characterized using conventional non-destructive inspection methods such as Ultrasonic C-scan, Laser Shearography, and Coordinate Measuring. Typical damage characteristics that should be quantified include the damage area, shape, and depth of indentation. Any correlation between relevant impact parameters, panel

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configurations, and induced damage may serve as the basis for a semiempirical model characterizing impact damage development in sandwich laminates.

In this task, the damage size and shape from a given impact environment should be quantified for a given specimen configuration. The effects of impact energy, impact velocity, and impactor diameter should be summarized in carpet plots based upon the experimental test data. A typical plot of damage size versus impact energy data is shown schematically in figure 25. Damage detection thresholds (detectable by eyesight or popular NDI equipment) may also be established from the experimental data. This type of data may be beneficial to certification officials in the near term as confidence is gained from evolving sandwich designs.

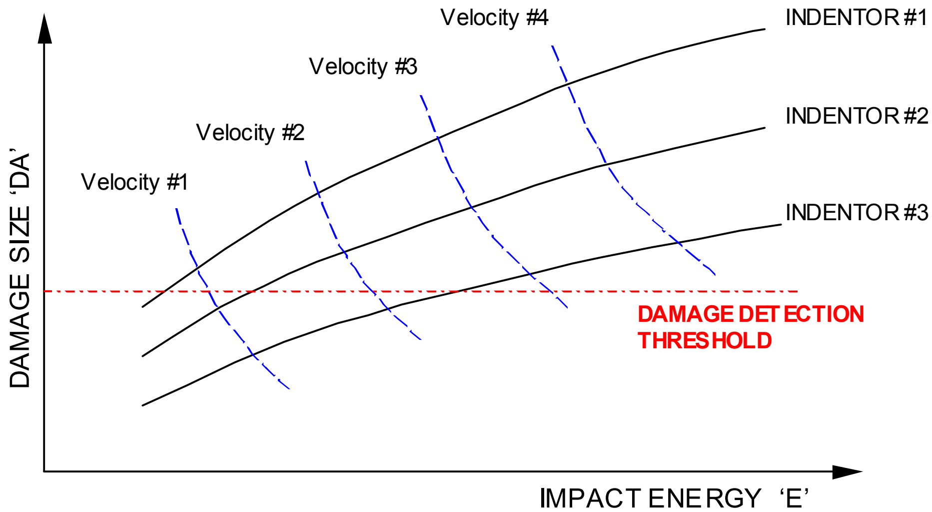

FIGURE 25. TYPICAL PLOTS EXPECTED FROM THE EXPERIMENTAL PROGRAM

6.3 TASK 2: RESIDUAL STRENGTH TESTING OF SANDWICH PANELS.

This section of the suggested investigation addresses the damage tolerance aspects of sandwich panels. The main objective of this task should be to correlate a characteristic damage dimension with the degradation in residual strength. The residual strength of damaged panels should be evaluated using compression after impact tests. These tests ostensibly represent the most critical form of loading the sandwich structure might undergo during its service life as shown previously.

For a given characteristic damage dimension (e.g., damage area and dent depth), the residual strength under compressive loading should be determined based on the experimental observations. Residual strength versus damage size plots such as shown in figure 26 should be generated from the experimental data. The residual strength for a given sandwich panel configuration may also be characterized in terms of the impact energy level. The residual strength and damage size data should help clarify the definition of Barely Visible Impact Damage (BVID).

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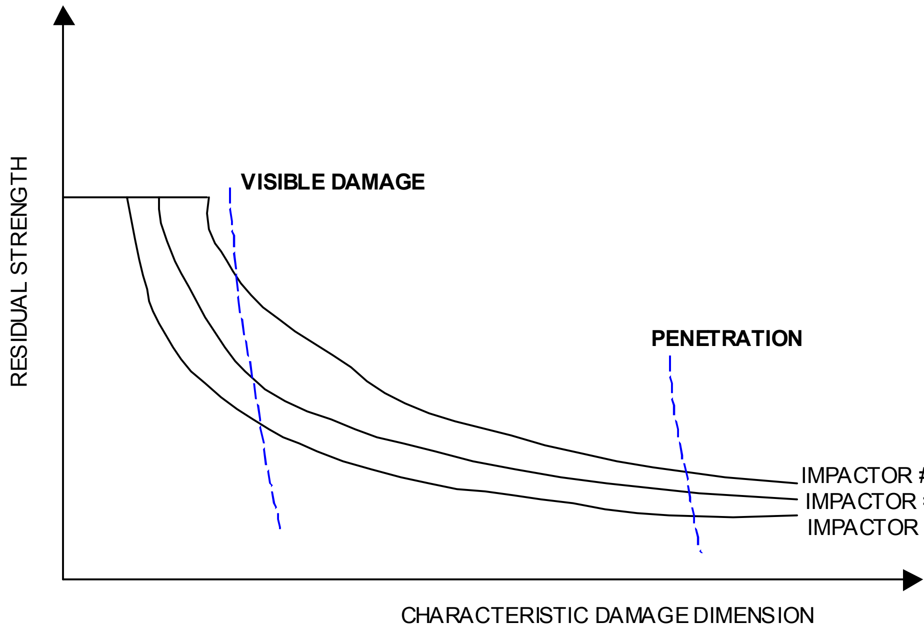

FIGURE 26. TYPICAL PLOT TO BE GENERATED USING THE RESIDUAL STRENGTH DATA

6.4 TASK 3: FLAW GROWTH THRESHOLDS AND DAMAGE EVOLUTION UNDER VARIABLE-AMPLITUDE CYCLIC LOADING.

The growth of impact-induced damage in sandwich panels subjected to variable-amplitude cyclic loading is a major concern in the aviation industry. Based upon the results of static compression-after-impact testing, a limited number of undamaged and damaged sandwich panel configurations should be selected for constant-amplitude cyclic load or spectrum load testing. Flaw growth thresholds, damage evolution, and residual strength degradation under uniaxial cyclic loading will be investigated for a number of sandwich panel configurations and initial-damage states.

While impact-damaged sandwich panels are generally critical in compression, the effect of cyclic compression and cyclic tension loading on damage initiation/growth remains unclear. Both constant-amplitude cyclic loading and spectrum loading should be considered. The maximum and minimum stress levels in a given constant-amplitude cyclic test should be based on some suitable fraction of the compressive residual strengths for undamaged panels.

Sandwich panels with varying degrees of impact damage should also be subjected to cyclic loading. For a given panel configuration, initial damage state, and cyclic load case, damage growth should be determined as a function of the number of loading cycles. Hence, a damage growth threshold may be established in terms of panel configuration, initial damage state, and cyclic stress level. In addition, at predetermined numbers of load cycles, static compressive residual strength tests should be performed on select test panels. The degradation in residual strength due to cyclic loading should be determined as a function of panel configuration, initial damage state, and cyclic stress level. This is shown schematically in figure 27. The cyclic flaw

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growth data may aid in the establishment of initial inspection thresholds and repeat inspection intervals for sandwich panels with varying degrees of initial impact damage.

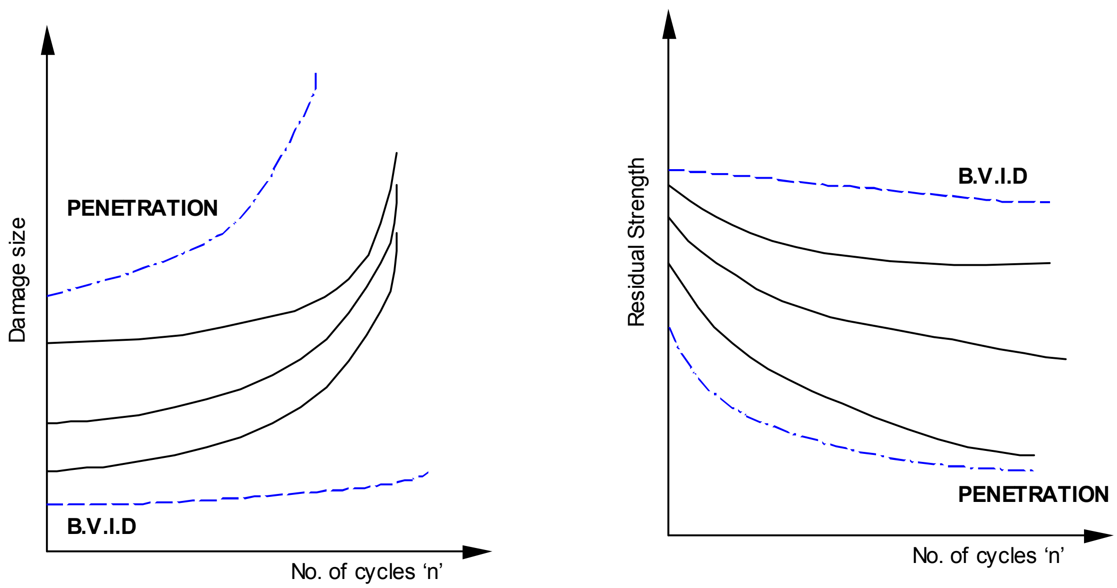

FIGURE 27. TYPICAL DAMAGE GROWTH AND RESIDUAL STRENGTH CHARACTERISTICS FOR SANDWICH PANELS WITH VARIOUS INITIAL FLAW SIZES/TYPES

6.5 TASK 4: ANALYTICAL MODEL DEVELOPMENT.

A number of analytic, numerical, and semiempirical models for predicting impact damage development and residual strength degradation in sandwich composites should be considered as part of any study. An attempt should be made to develop and validate a reliable micromechanically based analytical model for characterizing the impact damage development and residual strength degradation in idealized sandwich panels that is valid over a suitable range of material systems and impact parameters and that accounts for fundamental failure modes/mechanisms. Given the wide range of potential material systems and impact parameters, however, it is likely that robust closed-form analytic models will be difficult to obtain for the general case involving large-scale damage and/or projectile penetration. Various semiempirical modeling techniques or numerical approaches (e.g., nonlinear finite elements, boundary elements, or a hybrid approach) may prove more fruitful when considering a broad range of impact and material parameters. One particularly attractive and elegant approach is to use a factorial-based design of experiments technique to determine the effect of a wide range of governing material system and impact parameters on the damage induced and the damage characteristics of sandwich composites. Based on a number of carefully selected experiments, statistically reliable polynomial expressions characterizing the impact damage induced and as well as the damaged thermomechanical response may be determined as a function of selected test parameters. The nonlinear interaction effects between critical design and impact parameters (residual strength, impact energy, panel configuration, etc.) may then be readily inferred.

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6.6 TASK 5: FULL-SCALE/COMPONENT TESTING AND VERIFICATION.

The final task of a damage tolerance program should be focused towards obtaining full-scale or large-scale component information on the effects of large-scale damage on realistic aircraft structure. The investigation should primarily be focused towards fuselage applications which undergo internal pressure loads as well as bending loads. This proposed effort may be accomplished using testing facilities at the FAA. Several impact states should be chosen based upon tasks 1 through 4 and mechanically inflicted upon the test article. Cyclic loading to obtain flaw growth information and residual strength tests should be performed and compared with the data and model predictions from tasks 1 through 4.

6.7 SUMMARY OF SUGGESTED TASKS.

The first four tasks of this investigation may be summarized in the flowchart shown in figure 28.

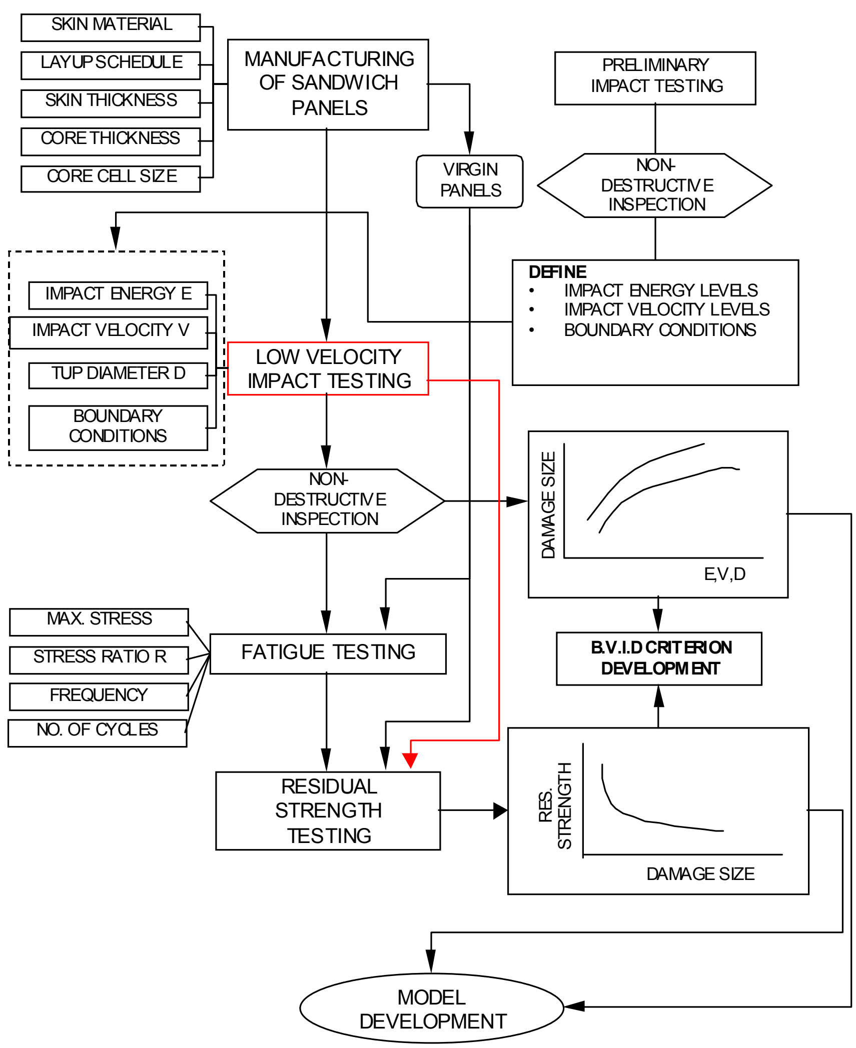

FIGURE 28. SUMMARY OF THE DAMAGE RESISTANCE AND TOLERANCE INVESTIGATION OF SANDWICH PANELS

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