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Aging of Composite Aircraft Structures

Principal Investigators: Dr. John Tomblin, Executive Director, NIAR and
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Problem Statement

As more commercial and military airplanes are required to maintain operational capability beyond their original design life objectives, it has become necessary to assess the structural health of aging aircraft to ensure their airworthiness and structural integrity for continued service in commercial and military applications. Most of the aging aircraft studies have focused so far on metallic structures; however, as more composite components are being certified and used on primary and secondary aircraft structures, it is crucial to address this aging concern for composite components as well. With these concerns in mind, the primary objective of this research program is to evaluate the aging effects on a B737 composite stabilizer and a Beechcraft Starship.

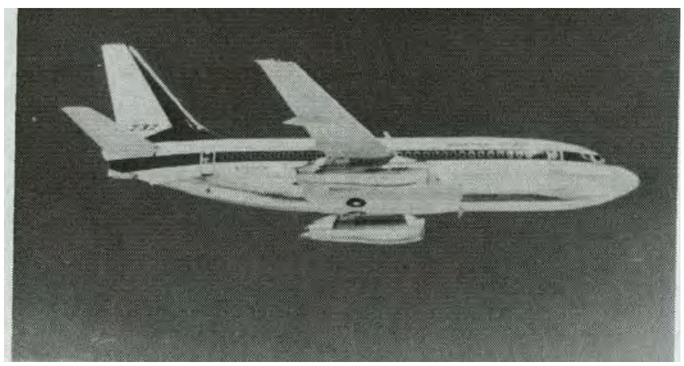


Figure 1. B737 Composite Horizontal Stabilizer

Figure 2. Beechcraft Starship

Background

B737 Horizontal Satblizer, Aircraft Energy Efficiency Program (ACEE)

In 1975, NASA initiated the **A**ir**C**raft **E**nergy **E**fficiency (ACEE) Program. With this effort, all three US OEMs: Boeing, Mc Donnell Douglas and Lockheed Martin designed, built, certified, and deployed a limited production run of both secondary and primary structural components as shown in Table 1 below.

Table 1. *NASA ACEE Components*

Manufacturer	Secondary Component	Primary Component
Boeing	727 Elevator	737 Horizontal Stabilizer
Douglas	DC-10 Upper Aft Rudder	DC-10 Vertical Stabilizer
Lockheed	L-1011 Inboard Aileron	L-1011 Vertical Stabilizer

As part of the program, Boeing re-designed and manufactured five shipsets of the B737 horizontal stabilizer using graphite-epoxy composites. The OEM adopted a fail safe damage tolerant approach to certify the structure where the structure's capability to sustain ultimate loads was validated by analysis supported by appropriate test evidence. This complies with the requirements of FAR Part 25 certification requirements for commercial transport aircraft.

The aircraft that these early components were deployed on are now nearing or at the end of their useful economic life. As such, they are being removed from service enabling post-service teardown inspection activities. The B737 R/H horizontal stabilizer which is the subject of the current investigation is one of the oldest medium primary structures built using composite materials. The structure was built as part of the NASA ACEE program using carbon reinforced graphite epoxy composites, was certified by the FAA and entered service in August 1984. The structure was in service for the following 18 years and was retired in 2002 after accumulating 48000 flights.

As an attempt to characterize the structural health of the horizontal stabilizer after 18 years of service, the NIAR (National Institute for Aviation Research) acquired the aged structure in 2005 and conducted several non-destructive and destructive tests to assess its structural health. Data generated will be used to understand aging mechanisms on composite parts currently in service, and to shed some light on main differences between damage mechanisms and damage accumulation in metallic components versus composite components. This could be used to establish more accurate inspection and maintenance plans for composite structures to ensure their continued airworthiness and safety.

Beechcraft Starship Teardown:

The starship program was officially launched in 1982. Objectives were to produce the most advanced turboprop business airplane feasible at the time and to promote the use of composites in a business aircraft. The first beechcraft starship was flown on February 15th, 1986. The second joined the test flight program in June 1986, and the third was ready for flight in the early spring of 1987. In the course of a two-year flight test program, they flew almost 2,000 hours, and on June 14th the Starship was certified by the FAA.

Objectives:

The primary objective of this project is to evaluate effects of aging on composite aircraft structures. It involves the investigation of two aircraft structures, a decommissioned Boeing 737 stabilizer that had a commercial service history of 18 years and a Beechcraft Starship, after 12 years of service. The proposed research is sub-divided into small sub-tasks to understand the aging mechanism of the structures which includes the following:

- Conduct detailed visual inspections prior to teardown to identify external damage to the structure or possible manufacturing/ in service defects.
- Conduct detailed non-destructive inspection to identify defects induced during manufacture or service
- Perform mechanical tests at the coupon and the element level to investigate any changes in the material properties
- Quantify the physical properties of the composite structures and determine porosity and moisture levels.
- Investigate the structures for microcracks, delaminations, damage, repair and bond integrity if applicable.
- Conduct detailed thermal analysis to identify any changes in the glass transition temperature of the material and investigate the degree of cure of the structure
- Evaluate effectiveness of repairs through destructive evaluation

The B737 composite horizontal stabilizer as shown in figures 3 and 4 consists of a co-cured skin and stiffener piece, 191 inches long and 50.5 inches wide at the root with stringers spaced 3.85 inches apart. Bolted Titanium spar lugs were used to fasten the stabilizer to the fuselage center section: this concept used two titanium plates bonded and bolted externally to a pre-cured graphite-epoxy spar. Honeycomb ribs were used because of the simplicity of the concept in terms of tooling, fabrication and cost. The composite design yielded average weight savings of approximately 21.6% with respect to the metal configuration or a final weight of approximately 206 lbs.

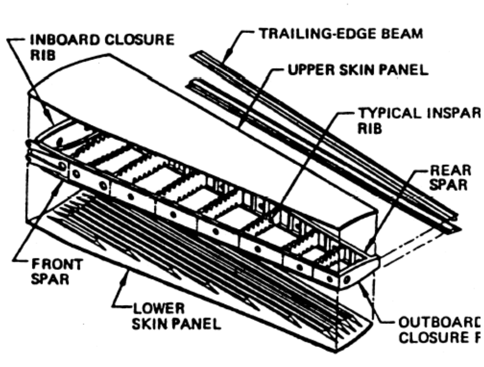
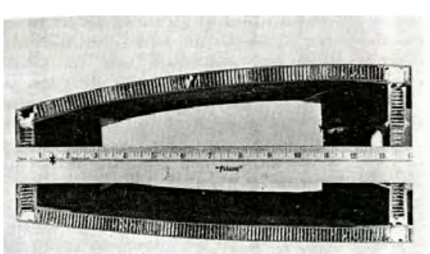
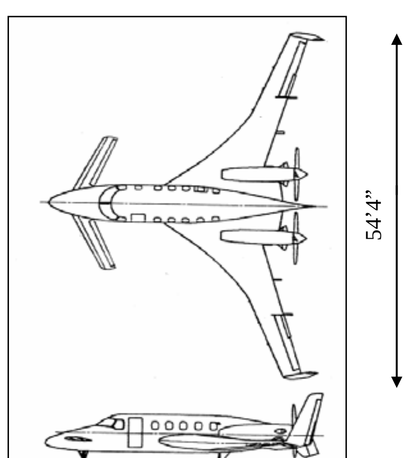


Figure (3). B-737 stabilizer configuration. Figure (4). B-737 composite stabilizer after 18 years of service.

The Beechcraft Pressure Cabin is a sandwich construction consisting of only two parts, a right and a left hand side bonded and riveted at the center section along the top and the bottom centerlines. The main wing is a sandwich construction as well with no spanwise stringers, three spars and five ribs as shown in figures 5 and 6.





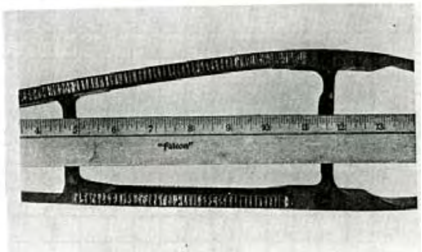


Figure (5). Beechcraft Starship aft and fwd wing cross sections. Figure (6). Beechcraft Starship Overall Dimensions (Courtesy of Beechcraft)

Expected Outcomes

The ultimate goal of this research program is to understand the aging mechanisms, characterize their effects on the composite structure, and to give recommendations pertaining to characterizing composite aging. The data generated will be used by the FAA to assess the efficacy of the current/ emerging certification methods; it will also be used to issue policy pertaining to composite maintenance, inspection, repair and long term airworthiness assurance.

Crashworthiness of Composite Airframe Structures

Principal Investigator: Suresh (Raju) Keshavanarayana
Department of Aerospace Engineering
Wichita State University

The crashworthiness of an airframe structure is measured in terms of its ability to maintain a survivable volume for the occupants and alleviate the loads transmitted to the occupants during potentially survivable accident scenarios per FAR §25.561 and §25.562. The occupant loads are minimized by dissipating the kinetic energy using an energy absorption device, while the structural integrity is maintained by accounting for the dynamic loads during the sizing of structural elements. The performance of an airframe under crash loads is dictated primarily by its geometry, structural arrangements, materials and energy absorption devices used to dissipate the energy, and the interaction of these variables. The energy dissipation in metallic airframes is primarily due to plastic deformation while in composite airframes is due to synergistic sequence of failure mechanisms. The limited number of dynamic and drop tests performed on fully composite fuselage structures have indicated differences in the crush patterns/failure modes, stiffness and other structural properties, compared with the traditional metallic fuselage structures. The test results are useful in the appraisal of the structural performance under crash loading, but do not reveal the various mechanisms that do and do not contribute to the overall performance of the structure. Further, the performances of individual components may be influenced by the overall structural assembly. To investigate the effectiveness of various components, numerical modeling would be more appropriate and less expensive. However, the predictions of the numerical models are dependent on the geometric definition of the structure, the material models, failure criteria, etc. The description of material behavior under dynamic loading is a key aspect of the numerical modeling of the crash scenarios.

In this study, a building block approach illustrated in figure (1) has been embraced to study the rate effects on the behavior of composite airframe structures. To begin with, the rate effects (i.e., strain rate effects), will be characterized at the fundamental ply level, since for most numerical analysis, the material properties are specified at this level. The material testing at this level includes the characterization along the primary ply directions and off-axis specimen testing. The off-axis specimen tests are intended to simulate a combined stress state in the plies and not to characterize material property. This combined stress state will be used for benchmarking material models in the future. The next level of testing will include cases where *strain* and *strain rate* gradients exist, e.g., open-hole tension, which will serve as benchmark data for the material failure models. Moving up the building blocks, small components and assembly of components will be characterized under dynamic loading, culminating in characterization of scaled aircraft structures, which would be the most expensive. The understanding of rate effects at the ply level will help identify the contributions of geometric effects and material rate sensitivity to the observed rate effects in structural components and their assemblies subjected to dynamic loading.

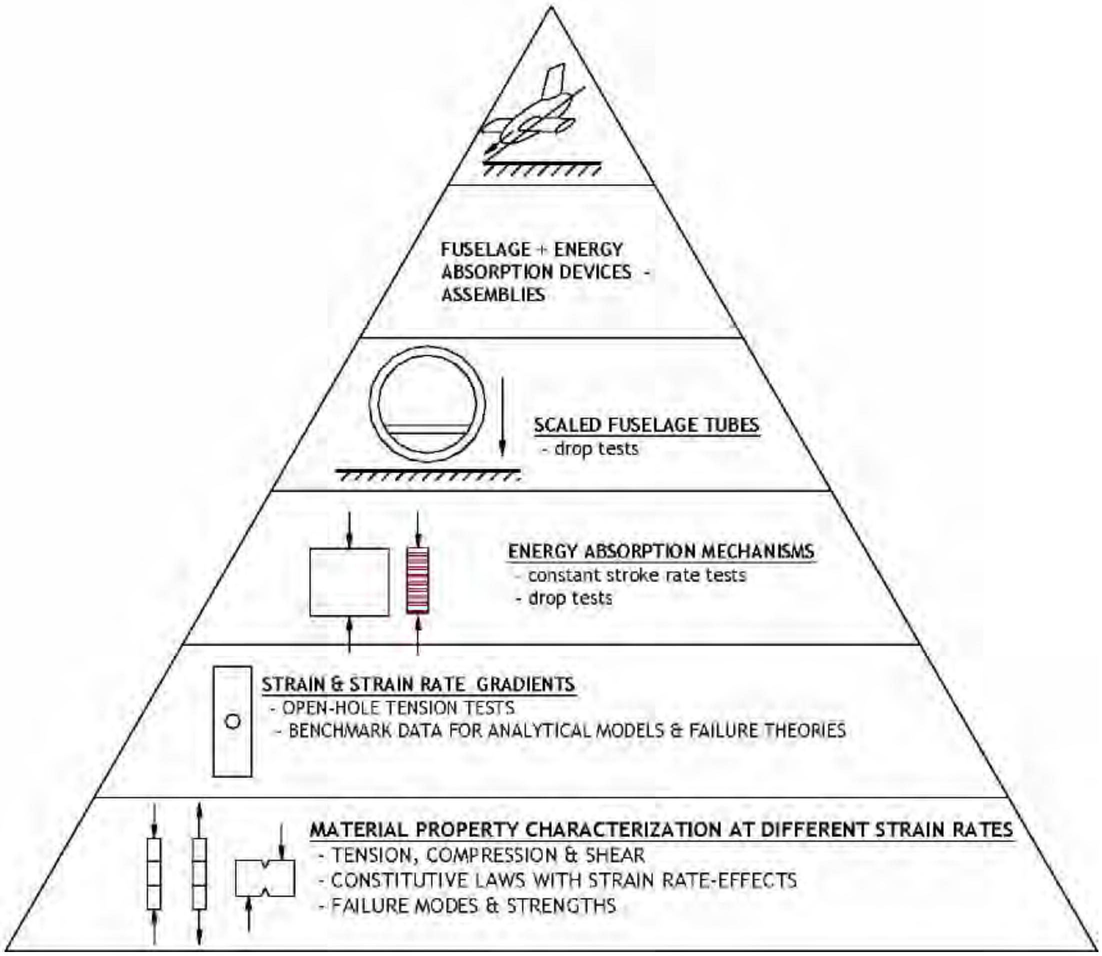


Figure 1: Building block approach for characterizing rate effects on composite airframe structures.

In the first phase of this investigation, the effects of strain rate on the in-plane tension, compression and shear properties of selected composite material systems were characterized experimentally. A limited number of off-axis and open-hole tension tests were conducted to investigate the rate effects under combined stress states and in the presence of strain and strain rate gradients. A considerable effort was focused on researching the appropriate method for high rate tests and the design, fabrication and assembly of apparatus for the current program.

In the ongoing phase of the program, the effects of test speed on interlaminar fracture toughness of the composite material systems and energy absorption characteristics of corrugated laminates are being studied. The mode-I interlaminar fracture toughness is being investigated using the double cantilever beam (DCB) specimen configuration. The inertial effects associated with the hinges and the load train between the load cell and the specimen has been minimized. The effects of displacement rate on the fracture initiation load and crack propagation load has been of particular interest. The crack tip propagation speed has been obtained approximately using the actuator stroke rate and the recorded load in conjunction with Euler-Bernoulli beam theory.

Standardization of Numerical and Experimental Methods for Crashworthiness Energy Absorption of Composite Materials

Principal Investigator: Dr. Paolo Feraboli

Introduction

Recent studies [1] identified the key factors preventing the introduction of composites in primary crash-resistant structures in aircraft and automotive as the absence of:

- available design guidelines,
- accurate and inexpensive simulation tools,
- specialized test methods for the characterization of energy-absorption,
- accessible and adequate composite material property database.

A coordinated cross-organizational effort can lead to substantial advances in all these fronts through systematic investigations of current practices. Such effort has been undertaken by the Crashworthiness Working Group of the CMH-17 (Composite Materials Handbook), which comprises representatives from the aerospace and automotive industries, academia, and government laboratories and operates in parallel with ASTM Committee D-30 on Composite Materials.

The Crashworthiness WG is currently developing a standardized test method for measuring the specific energy absorption (SEA) capability of composite materials, and developing guidelines for analytical methods to effectively simulate composite structures under crash loading.

Development of test standards

The vast majority of the work in the area of crashworthiness energy absorption has focused on thin-wall tubular specimens [2,3]. Yet, almost every investigator around the world has used different specimen geometries, setups, and loading rates to assess the energy absorption (EA) of their designs. Comparisons across different materials and stacking sequences are still very limited and only in a few occasions they have been performed systematically. The lack of an agreed upon test method makes the comparison of test results and research findings across the literature virtually impossible.

Limited work has been done with test geometries other than tubes, and it evolved in two directions [3-5]. The first approach features simple specimens, such as flat plates with or without built-in crush initiators, and very complex and costly anti-buckling support fixtures. The second approach features self-supporting specimen geometries, which require dedicated molding tool for manufacturing, but don't require the use of specific test fixtures.

Experimental work conducted to date has shown that composite energy absorption measurements are inherently related to test set-up characteristics through complex relationships. This applies to all specimen shapes, whether tubes, flat plates, or corrugated webs. Tubes present complex interactions due to the hoop constraint that off-axis fibers provide; flat plates present unknown degrees of interaction and constraint between the coupon and the support fixtures; corrugated webs have shown a moderate influence of corrugation geometry on measured SEA. However, it seems that self-stabilizing, open-section specimens such as corrugated webs may present a more acceptable means by which to measure a material's SEA and with which to validate material models in numerical simulations. This is in large part attributable to the fact that like tubes they do not require support fixtures, and like flat plates they do not have fiber in the hoop direction.

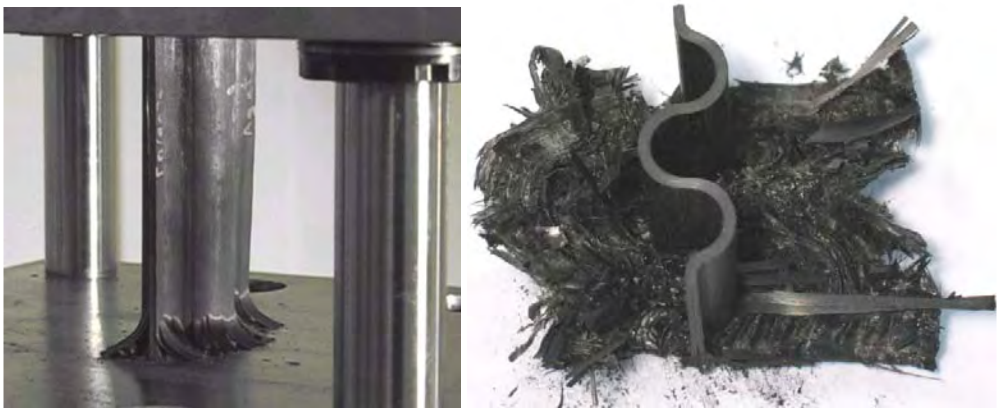


Figure 1. Corrugated plate specimen ready for testing in the test fixture, and close-up after crushing.

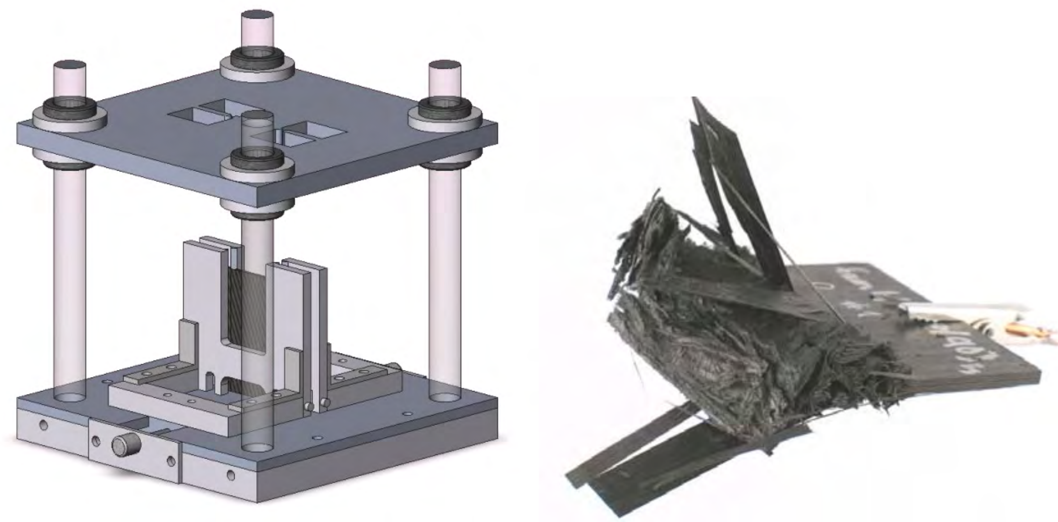


Figure 2. Flat plate specimen in the specialized test fixture, and close-up after crushing.

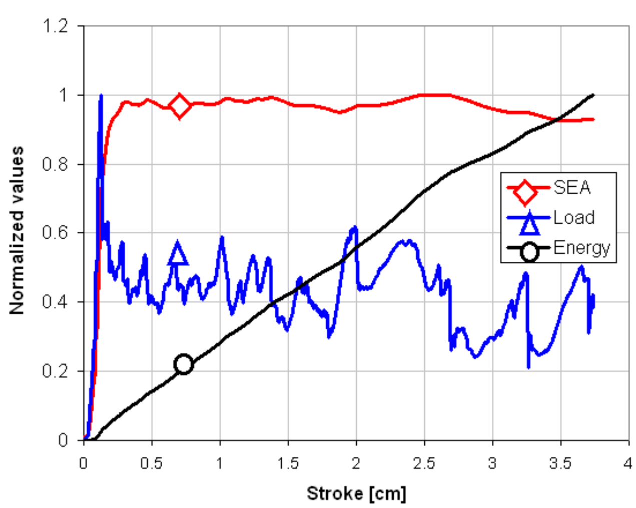


Figure 3. Exemplar plots of load, SEA, and total energy absorbed traces vs. stroke.

Tues., June 17 presentation

Validation of modeling strategies

Composite structures used in energy absorption applications are capable of sustaining large deformation with substantial load carrying capability. To predict the energy absorption, the constitutive models should be able to describe the response of substantially damaged material. The work by the members of CMH-17 Crashworthiness WG indicated that the capability of describing such responses in different codes with current composite models varies, but no one model can capture the complicated fracture morphologies. The group is coordinating a numerical round robin to compare and benchmark composite constitutive models in different codes (LS-DYNA, ABAQUS Explicit, PAM-CRASH, RADIOSS).

Composite constitutive models implemented in commercial finite element (FE) programs are continuum mechanics models. Composites are modeled as orthotropic linear elastic materials within a failure surface. The exact shape of the failure surface depends on the failure criterion adopted in the model. The failure criteria for laminated composites are typically strength based. Beyond the failure surface, the appropriate elastic properties are degraded according to the assigned degradation laws. By means of stiffness degradation, the nonlinear and softening responses of a composite material after its initial failure are described mathematically.

Depending upon the specific degradation law used in a model, the continuum mechanics models can be further divided into two broad categories. They are either progressive failure models or continuum damage mechanics (CDM) models (see Table 1). Progressive failure models use a ply discount method to degrade material properties. At the failure surface, the values of the appropriate elastic properties of the ply at the material direction are degraded from its undamaged state to a fully damaged state, which is often considered a complete loss and assigned a value of zero. The so-called progressive failure is realized through ply-by-ply failure in the perspective of a laminate. Progressive failure models have shown success [6] in axial crush simulations of composites exhibits brittle fracture, which are often observed in composite reinforced with unidirectional prepreg and fine woven fabric. Composites reinforced with fabrics of large tow size such as braid, continuous strand glass mat (CSM), and pultruded fiber glass tend to form continuous fronds at the crushing front instead of being crushed into debris. Our efforts using progressive models so far have not led to stable simulations for composites exhibit this type of failure [7]. Composite models based on continuum damage mechanics appear to be a better choice to model composites that form continuous fronds [7,8].

Table 1. Summary of composite material models available in the commercial explicit FE code LS-DYNA.

MAT	Title	Brick	Shell	T-shell	Degradation Law
22	Composite Damage	y	y	y	Progressive failure
54/55	Enhanced Composite damage		y		Progressive failure
58	Laminated Composite Fabric		y		Damage Mechanics
59	Composite Failure	y	y		Progressive failure
161	Composite MSC	y			Damage Mechanics
162	Composite MSC	y			Damage Mechanics

LS-DYNA simulation of a sine-panel

LS-DYNA explicit finite element analysis code is used to simulate the crushing of sine-wave composite panel structure enabling efficient characterization of energy absorption of composite. Figure 4 shows the geometry definition of the sine-wave panel with 12-ply lay-up $[0/90]_{3s}$ for a thickness of 0.075 in. (1.9 mm). Specimen length is 3.0 in. (76 mm). The material properties used in the model have been previously characterized for the highly toughened material system used in the sine crushing experiments [4]. The specimen is standing freely and in contact with the crush plates, both the movable and the fixed ones. The contact surfaces are dry, rigid, as-milled steel plates. A 45-degree chamfer is used as trigger mechanism to initiate stable crushing.

The model uses MAT54 shell elements, while the trigger is modeled as a row of element at 1/3rd of the actual thickness. Imposed test velocity is 1.0 in/ sec (25 mm/ sec), which is noticeably below the observed dynamic thresholds [4], and has shown to give identical SEA measurements as the 0.05 and 0.5 in/ min tests (1.3 and 13 mm/min respectively) [5]. The model is shown in Figure 5, left.

The force-deflection curve obtained from the modeled is compared to the experimental one in Figure 5, right. The simulated curve is shown both as raw (unfiltered), as well as filtered using a 100 Hz SAE filter. The key parameters to facilitate crushing are the use of the “crashfront” elements available with LS-Dyna’s composite crush model, with “SOFT” parameter between the web and the plate. Through this feature, when a row of elements is deleted during the crush the remaining nodes are “pre-loaded” so that the element deletion will not lead to large dynamic oscillations providing a stable crush behavior. In the present study the most effective value for the soft contact parameter is has been found to be 0.6.

Using MAT54 for strain-base material failure model allows elements to erode when strain components fail at all element integration points. This technique deletes rows of elements as they come in contact with the loading plate, hence element debris or fronds are not expected to appear in the simulated failure morphology. Other investigators have shown [8,9] the formation of frond using material model MAT58.

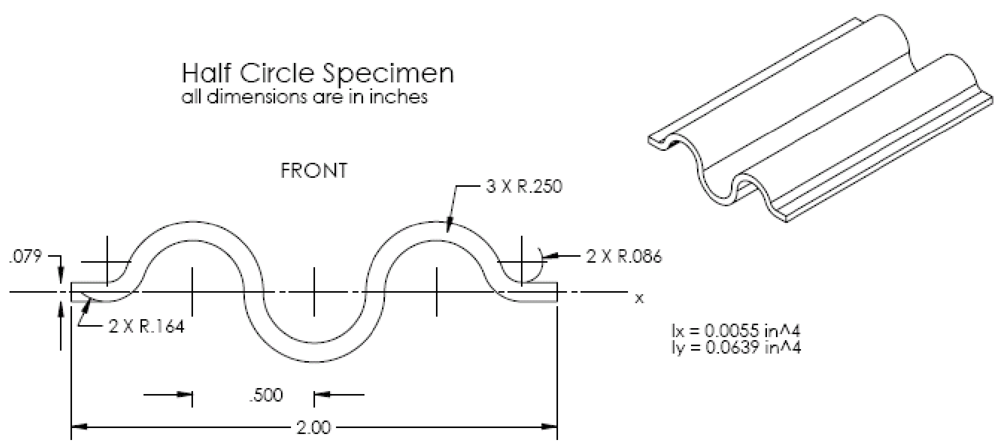


Figure 4. Semicircular sine-wave panel geometry

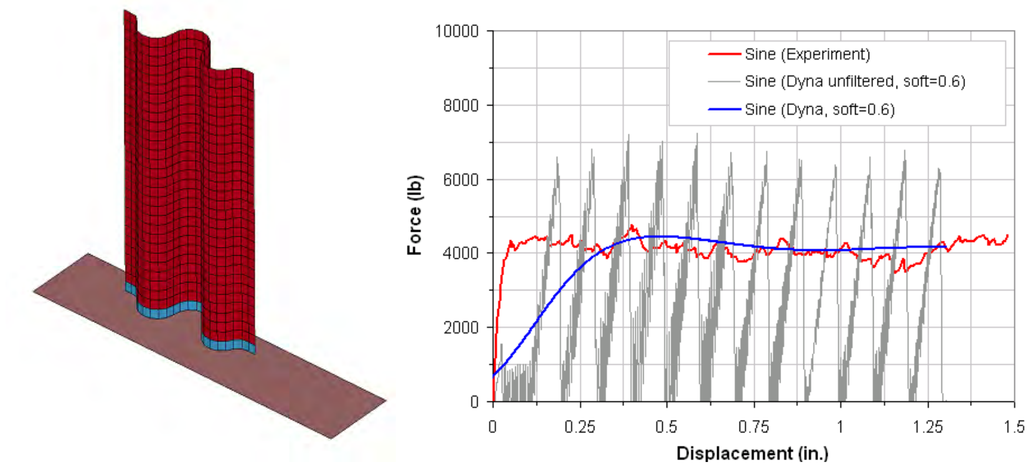


Figure 5. LS-DYNA Finite Element Model and comparison with experiment

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Damage Tolerance Testing and Analysis Protocols for Full-Scale Composite Airframe Structures under Repeated Loading

Principal Investigator(s): John S. Tomblin, PhD and Waruna Seneviratne (WSU/NIAR)

Over the years, composite and hybrid structural certification programs adopted methodologies utilized for metal structures that are based on several decades of experience in full-scale structural certification and service. Despite the advantages such as high specific weight, tailorability, and fatigue resistance, composite structural certification becomes challenging due to the lack of experience in large scale structures, complex interactive failure mechanisms, sensitivity to temperature and moisture, scatter in data, especially in fatigue. Most of the current fatigue life assessment methodologies for advanced composite structures rely on empirical S-N data in lower levels of building block of testing. Variation of material characteristics for different fiber-resin systems, layup configurations, environments, loading conditions, etc. often make the analysis and testing of composites challenging. The anisotropic heterogeneous characteristics and change in failure modes over the fatigue life and multiple failure mechanisms that interact with each other make it challenging to predict the damage growth in composite structures. Consequently, most of the damage mechanisms and wearout approaches discussed in the literature are also depend on empirical data for refinement or calibration. Some approaches only discuss failure progression under certain loading configuration and often specific to a material system. As a part of the F/A-18 certification, a probabilistic methodology was developed to certify composite structures with the same level of confidence as metallic structures [1]. This methodology is formulated to account for the uncertainties of applied loads as well as the scatter in static strength and fatigue life related to composite structures. Over the years, several composite structural certification programs employed this certification methodology, which is developed for materials and test methods that were then considered current. Since then, materials and process techniques as well as test methods for evaluating composites have evolved significantly. Consequently, test data often display significantly less scatter with high reliability. Thus, NAVY probabilistic approach can be reevaluated with robust analytically techniques for newer material forms and to represent structural details of current aircraft structures to obtained improved life factor and load enhancement factors.

Primary objective of this research is to develop a probabilistic approach to synthesize life factor, load enhancement factor and damage in composite structure to determine fatigue life of a damage tolerant aircraft. This methodology will extend the current certification approach to explore extremely improbable high energy impact threats, i.e. damages that reduce residual strength of aircraft to limit load capability, and allow incorporating certain design changes into full-scale substantiation without the burden of additional time-consuming and costly tests. Research is conducted in three phases (Figure 1):

1. Load-life combined approach
2. Damage tolerance and flaw growth testing
3. Load-life-damage (LLD) hybrid approach

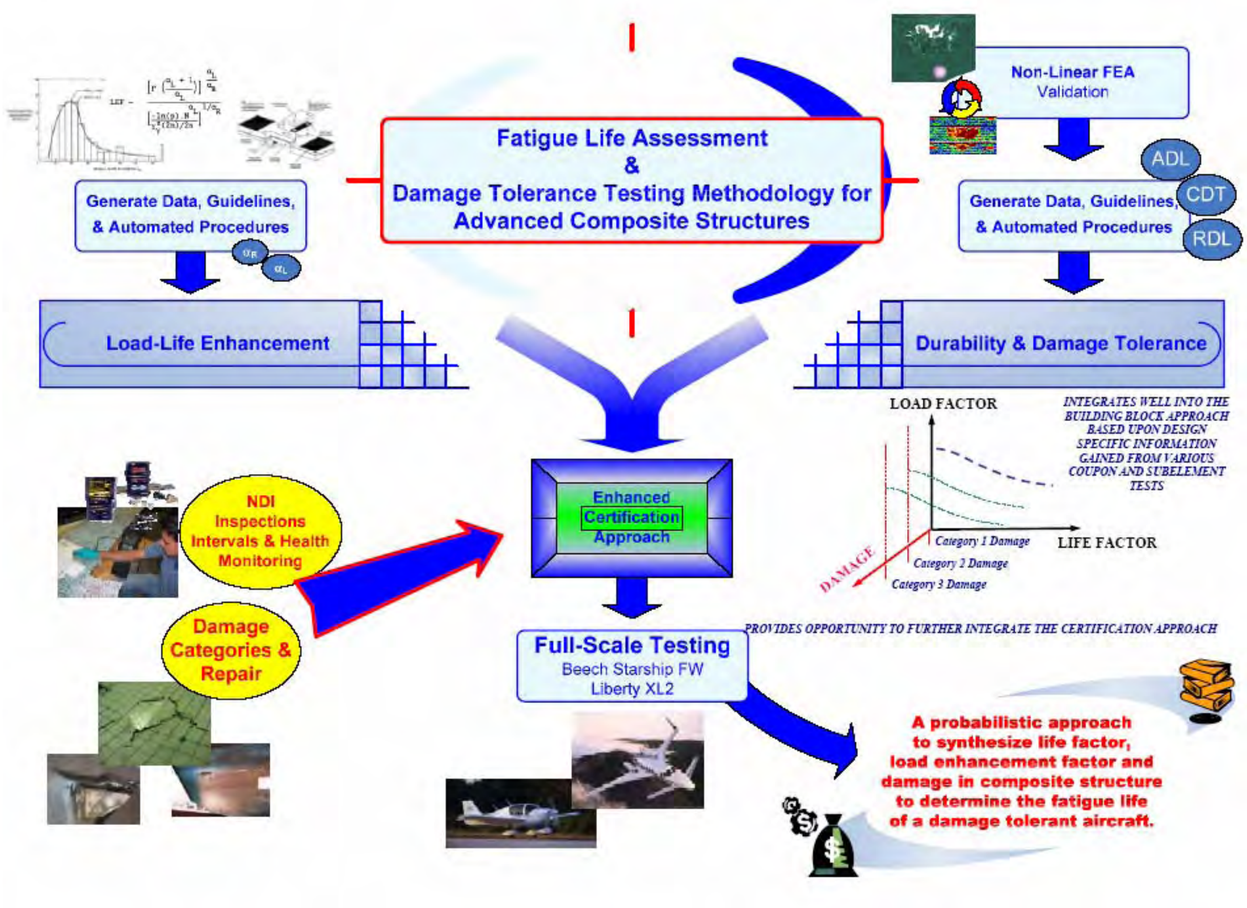

Figure 1 - Overview of research

First subtask of phase 1 is intended to generate a database of fatigue life data for several composite material systems that are commonly used in general aviation. Second subtask in this phase is to add static strength scatter factors to the database and generate improved load enhancement factors for several example materials. These data can then be used to generate necessary load enhancement factors for the full-scale demonstrations included in the final stage of the research. The improvements in materials and processes, and test methods produce life and load factors lower than the values commonly used in most certification programs based on NAVY research. Data gather in this phase will be used to provide guidance for generating safe and reliable load and life factors pertaining to a specific structure. In addition, a user-friendly computer code that can be used for scatter analysis of composite will be developed. This code will alleviate misinterpretation of any statistical or mathematical processes during the analysis and provide guidance for selecting different techniques appropriate for a particular application or a structural detail.

During damage tolerance phase of the research, fatigue characteristics of different categories of damages will be studied. Combined data from Phase 1 and 2 will be used in the methodology development and full-scale validation in the final phase. Nonlinear finite element analysis along with empirical validations will be used for scaling element-level damages to the damages required in full-scale durability and damage tolerance articles.

Final phase is intended to develop the Load-Life-Damage (LLD) hybrid approach for damage tolerance certification of composite structures. This methodology will be validated with several full-scale examples, Beechcraft starship forward wings (Figure 2). Category 2 (CAT2) and 3 (CAT3) damages will be inflicted onto full-scale articles to demonstrate LLD hybrid approach. Three static articles are tested to determine the ultimate load capabilities of the structure to define NIAR research limit and ultimate loads (NRLL and NRUL, respectively). Further, two additional static articles will be allocated for CAT2 and CAT3 baseline, i.e. post-impact residual strength. CAT2 static test is already completed and CAT3 static article will be tested following the damage tolerance tests in Phase 2. CAT2 fatigue test will also be conducted to demonstrate the damage tolerance and repair durability of this structure. Then, CAT3 fatigue articles will be tested until failure to validate proposed hybrid approach and to evaluate the fail safety of the structure with CAT3 damage.

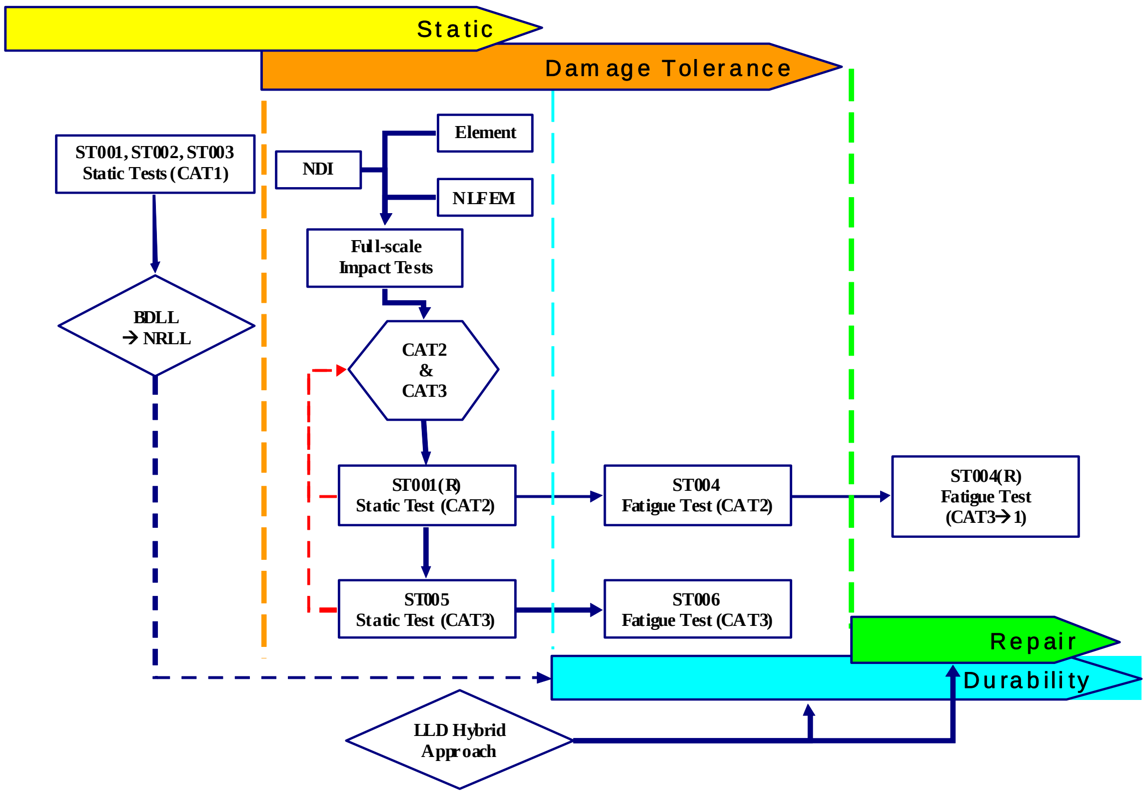

Figure 2. Outline of current status of full-scale testing

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Fluid Ingression Damage Mechanism in Composite Sandwich Structures

Allison Crockett, NIAR

Honeycomb sandwich construction techniques have been and are currently being used widely throughout the aerospace industry. These types of sandwich construction have been shown to be ideally suited for fuselage structures as well as control surfaces throughout the industry. A critical issue for the sandwich construction concept is the environmental durability. Skin panels may be exposed to water and other fluids both interior and exterior of the aircraft in service. Exposure to these fluids in combination with thermal and mechanical loads may result in degradation of these structures over time.

Sandwich structures have been observed to absorb moisture resulting in increased weight gain, degradation of the core and facesheet materials and degradation of the core-to-facesheet interface bond [1-6]. Figure (1) shows a basic schematic of the elements of sandwich construction in which the facesheets take the inplane loads (tension/compression) and the core takes the shear loads. It is important to understand the fluid ingression paths and damage mechanisms that result in decreased structural performance as related to continued airworthiness of the aircraft.

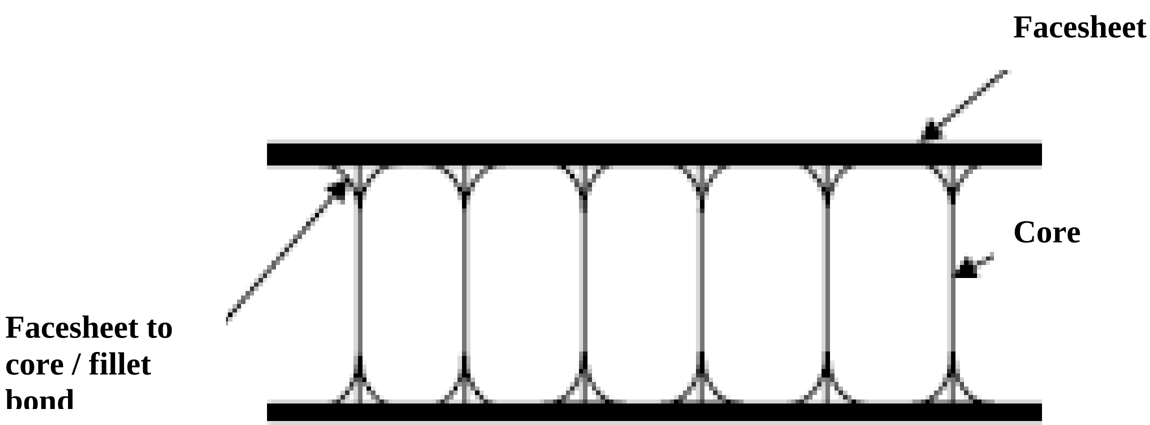

Figure (1). Basic schematic of honeycomb sandwich structure.

Historically, fluid ingression has been shown to enter the structure through several channels. Free-water or fluid can ingress easily through damaged facesheets /skins, near edges or penetrations through panels where seals are degraded or damaged, wicking may occur through rivet holes or impact damage and by diffusion through the epoxy matrix on the composite skin, particularly in the case of thin facesheets. A number of reports have documented service problems due to water ingression and have described ways to avoid these types of problems in future designs [6].

The objective for this research program is to characterize the fluid ingression phenomenon in composite sandwich structures as well as to document the damage mechanisms which allow the fluid ingression to propagate and potentially degrade the structural performance of the sandwich structure.

The technical approach proposed for the program will develop a detailed test plan to address these variables and their relationships in structural performance degradation due to fluid ingression.

Different damage mechanisms will contribute differently to the rate of moisture ingression. This research will attempt to isolate each mechanism independently to study the effects on the rate of moisture ingression. The act of isolating each damage mechanism and its effects on the rate of moisture ingression will provide a key guideline for where additional inspection intervals may need to be implemented.

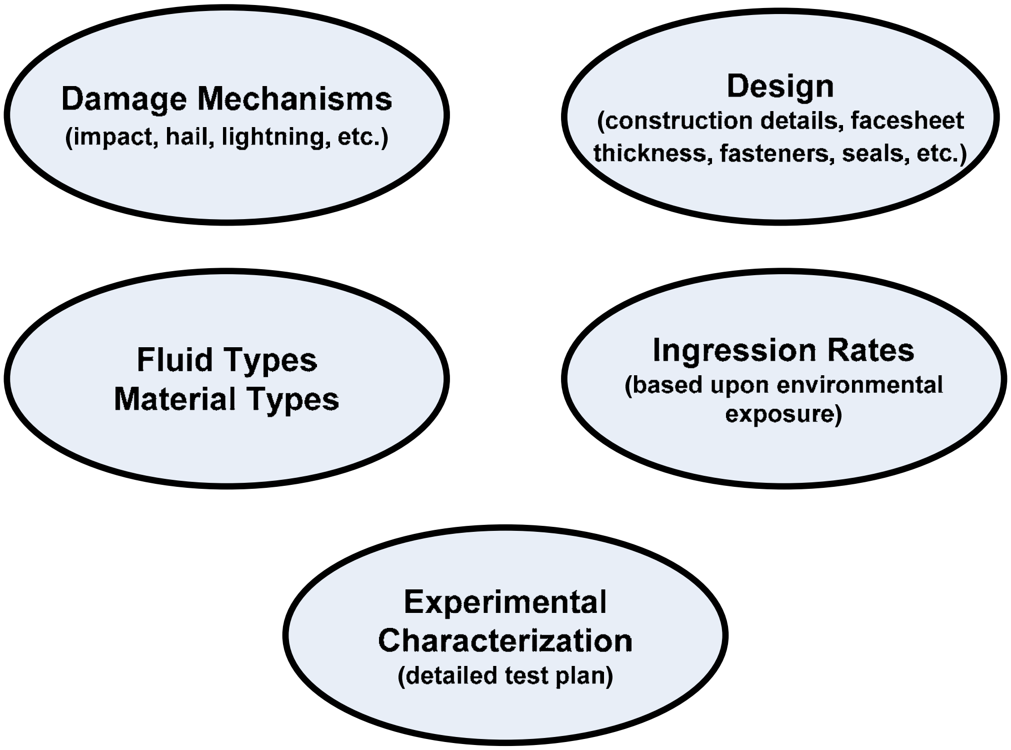

Figure (2). Fluid Ingression research variables.

As this project progresses industry can expect to see documentation for what has happened in terms of research, but more importantly what was learned from research. This will allow for different companies not to make the same mistakes. Furthermore answering some key concerns for fluid ingression will provide reassurance to the aerospace community that fluid ingression is no longer a problem that we do not know how to resolve.

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Certification by Analysis I and II

Principal Investigator: Dr. G. Olivares, Research Scientist, NIAR Crashworthiness Laboratory, WSU

Overview:

Physical testing is increasingly being replaced by numerical simulation models because it provides a more rapid and less expensive way to evaluate design concepts and design details. In the aerospace industry, crashworthiness numerical simulation methods are primarily used at the very end of the product development process. Often they are applied to confirm the reliability of an already existing design, or sometimes for further design improvements by means of optimization methods. There are a number of CAE (Computer Aided Engineering) tools that could be used for solving aircraft crashworthiness problems. These are best utilized by using a systems approach that uses a combination of CAE analyses, component tests, sled and/or full-scale tests.

Advisory Circular (AC) 20-146 sets forth an acceptable means, but not the only means, for demonstrating compliance to the following by computer modeling analysis techniques validated by dynamic tests:

- Title 14 Code of Federal Regulation (14 CFR) parts 23, 25, 27, and 29, sections 23.562, 25.562, 27.562, and 29.562

- The Technical Standard Order (TSO) associated with the above regulation, TSO C127/C127a.

Computer modeling analytical techniques may be used to do the following, provided all pass/fail criteria identified in §§ 23.562, 25.562, 27.562, or 29.562 are satisfied:

- Establish the critical seat installation/configuration in preparation for dynamic testing.
- Demonstrate compliance to §§ 23.562, 25.562, 27.562, or 29.562 for changes to a baseline seat design, where the baseline seat design has demonstrated compliance to these rules by dynamic tests. Changes may include geometric or material changes to primary and non-primary structure.

This AC provides generic guidance on how to validate the numerical model and under what conditions the model may be used in support of certification or TSO approval/authorization. AC 20-146 relies on the engineering judgment of the applicant and the FAA ACO to determine compliance. This AC could be enhanced if more specific data pertaining the modeling and validation procedures were defined, for example:

- Numerical Anthropometric Test Dummies (ATD) database validation criteria
- Structural seat numerical model / Testing validation criteria
- Material library with strain rate dependant mechanical properties for typical seat components
- Part joining modeling and failure criterion guidelines (rivets, welds, bolted joints, etc.)

Objective:

The intent of this research is to provide an overview of numerical modeling practices so that engineers can gain an understanding of the fundamental modeling methods, a feeling for the comparative usefulness of different modeling approaches, develop an appreciation of the modeling problem areas, and limitations of current numerical models.

The research has been concentrated initially in the validation of Hybrid II and Hybrid III FAA numerical models. This is being accomplished by establishing partnerships with numerical ATD model developers. NIAR has conducted a series of dynamic 14CFR 23.562 and 25.562 sled tests for typical aircraft configurations as shown in figure one. The results from these sled tests in conjunction with component calibration tests are used to evaluate the performance of the numerical ATDs.

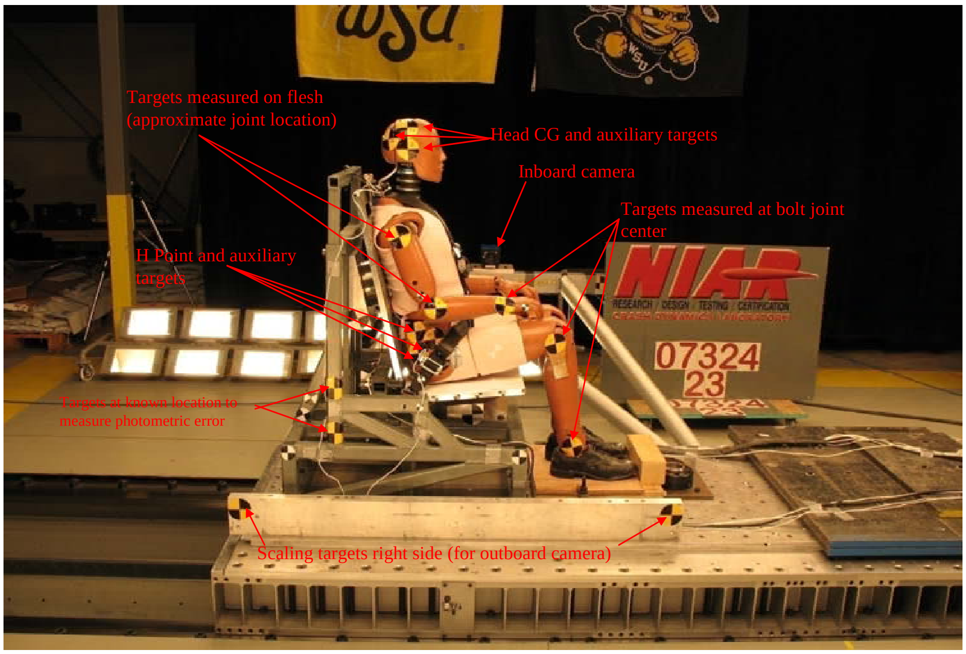

Figure 1. FAR25.562 Sample Rigid Seat Sled Test Setup HIII FAA.

Various numerical model validations that can quantitatively compare experimental and computational results over a series of parameters are being evaluated to objectively assess computational accuracy over the traditional qualitative graphical comparison.

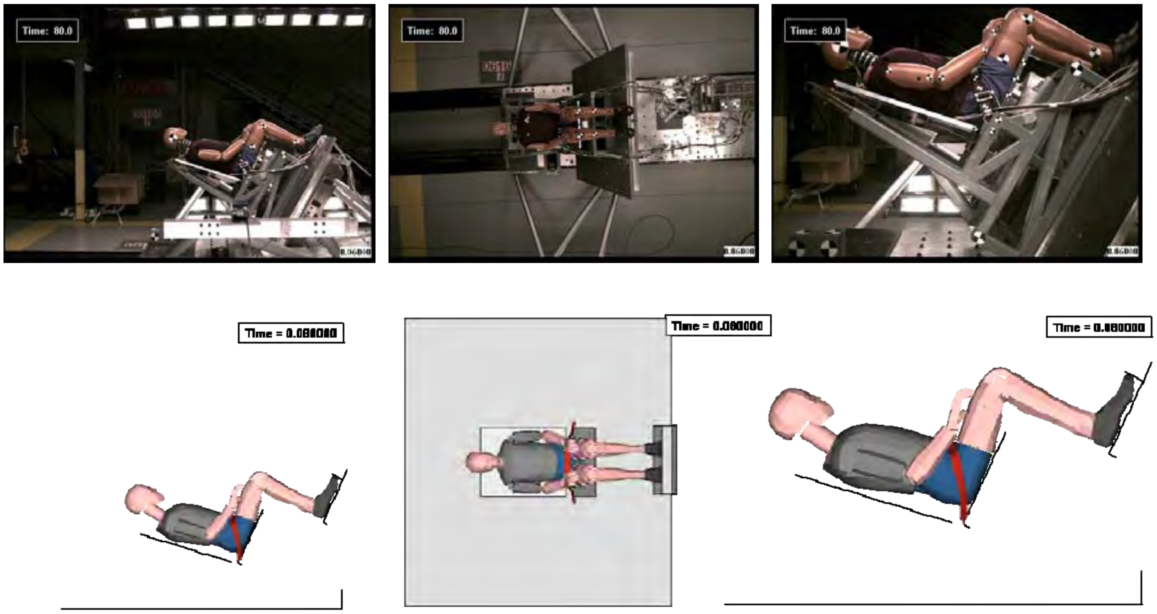

Figure 2. Example Occupant Kinematics Validation Hybrid III FAA.

Upon completion of the ATD databases validation process, the research will address the strain rate effects in the finite element modeling process of the seat structure. High strain levels are present in deformed crash components. Ignoring the strain hardening, and strain rate effects in numerical models may lead to an underestimation of the energy dissipated, and the structural performance of the aircraft seat. Currently there is no public domain strain rate dependant material database available for typical aluminums and steels grades used by seat manufacturers.

In order to define the appropriate strain rate domain for developing a material database, various seat components will be analyzed through analysis and sled tests of typical aircraft seats subjected to FAR 25.562 dynamic crash conditions. Using this strain rate domain, the materials will then be tested at the coupon level for various strain rates. These data will be available in the future for seat manufacturers to be used in the development of their numerical models.

Figure 3. Strain Rate Domain Study for a Typical Part 25 Aircraft Seat Structure.

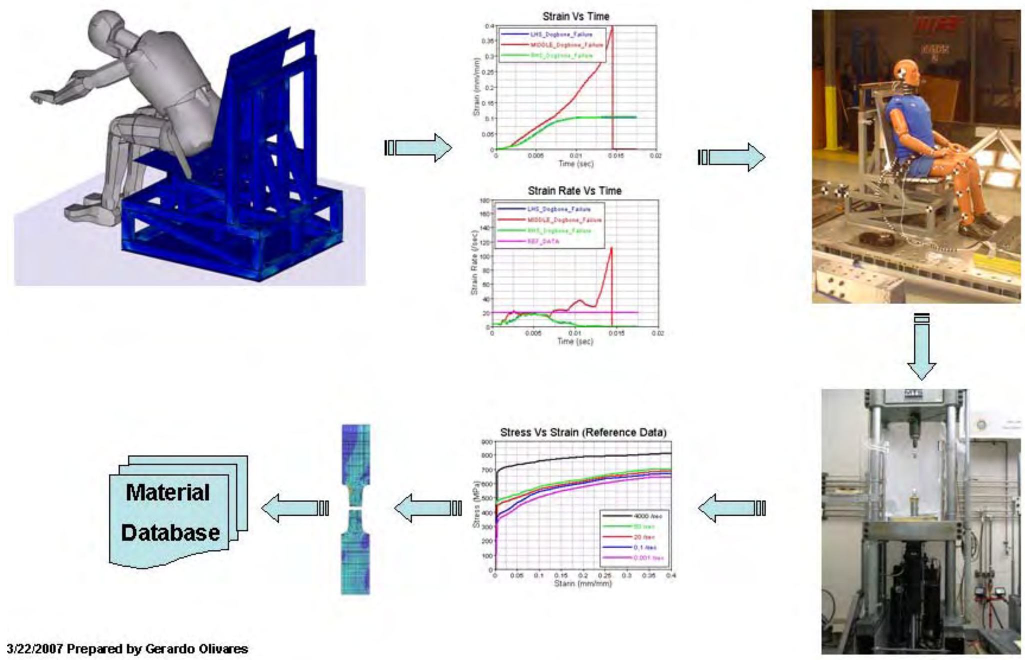

Figure 4. Material Database Development Process.

Expected Outcome:

- Validation criteria and procedures for numerical Hybrid II and Hybrid III FAA databases.
- Dynamic sled tests setup and data collection protocols.
- Strain rate dependant material database for typical aluminum and steel grades used by seat manufacturers.

Technology Assessment of the Airworthiness of Unmanned Aerial Systems

Allison Crockett, NIAR

Unmanned aerial system (UAS) vehicles have historically been flown only in restricted airspace or war zones and thus have largely avoided coming into conflict with manned civilian aircraft. This is changing. The traditional focus of the Federal Aviation Administration (FAA) has been to ensure that the multitude of aircraft flown in the National Airspace System (NAS) pose a minimum hazard to people or property on the ground and in the air. The operation of UAS vehicles in both military and paramilitary roles, such as the Department of Homeland Security and the Department of Defense (DoD), outside of restricted airspace has become increasingly likely. The need exists for the United States' NAS to be shared by both manned and unmanned users to support national defense and homeland security as well as other government and commercial applications. As a growing tool in these applications, UAS vehicles should be allowed to integrate into the current NAS infrastructure if they can be certified to conduct safe, efficient, and effective operations.

The potential of UAS vehicles cannot be fully realized until they can safely and routinely operate on an equal footing with manned aircraft in the NAS. Currently, the FAA allows temporary and restricted operations of UAS vehicles in civil airspace through the Certificates of Authorization (COA) process, through an experimental certificate (EC), or by temporary flight restricted (TFR) corridors. The COAs and ECs impose procedural constraints, such as chase aircraft or ground observers who must be within visual range of the UAS vehicles.

UAS vehicles come in a variety of shapes and sizes, as shown in figures 1. Figure 1 shows a UAS vehicle manufactured by Israel Aircraft Industries that is small enough to be held in the palm of a hand.

FIGURE 1. ISRAEL AIRCRAFT MOSQUITO UAS VEHICLE

The guidelines established thus far for the UAS design guidelines are built from working groups. These working groups have shown support in the UAS community primarily through working groups that have addressed issues relating to UAS operation within the National Airspace System. The focus of this chapter is to ultimately provide a framework of the guidelines available for the direction of future research so that general aircraft and UAS vehicles can safely occupy the same airspace.

All flights that travel within the United States National Airspace System are monitored and controlled by the Federal Aviation Administration. This regulation is done so under jurisdiction granted by law. The FAA regulates airspace within the Code of Federal Regulations Title 14—Aeronautics and Space. The bulleted titles below are the most popular working groups available to the UAS community, and each will be discussed in some detail.

- UAV Air Worthiness
- ASTM International F38 Committee Standards
- EuroCAE WG73
- EASA: European Aviation Safety Agency
- NATO STANAG 4671
- RTCA Sub Group SC-203

Safety is the key element that promotes future work and creates these working groups. Thus, safety is the reason for airworthiness certification and subsequent regulations, preventing airplanes that carry passengers from falling out of the sky during flight and thus the potential impact on people and objects at the site of the crash. UAS vehicles simplify this situation by not carrying passengers, but collateral damage at the site could be just as devastating.

Current FAA unmanned aerial vehicle (UAV) policy is specified in AFS-400 UAS Policy 05-01, published on September 16, 2005. This policy provides the framework for determining if UAS vehicles can fly within the National Airspace System. AFS-400 personnel will use this policy and the guidance material when evaluating each application for a Certificate of Waiver or Authorization (COA). Since UAS vehicles have had such a rapid growth in production and technology, this policy is subject to continuous review and will be updated when suitable. This policy is not meant to substitute any regulatory process. It was jointly developed by, and reflects the consensus opinion of, the following constituents:

- AFS-400, the Flight Technologies and Procedures Division, FAA Flight Standards Service (AFS).
- AIR 130, the Avionics Systems Branch, FAA Aircraft Certification Service (AIR).
- ATO-R, the Office of System Operations and Safety, FAA Air Traffic Organization (ATO).

The foremost path to a UAV operations ticket is via the standard airworthiness certification process. Airworthiness certification specific to UAS vehicles is found on the FAA's website. An Airworthiness Certification Overview and a complete Airworthiness Certification System for all airframes is maintained by the FAA. UAS vehicles are regulated under the Special Airworthiness Certificate, within the Experimental Category, and specifically within the Special Airworthiness Certification for Unmanned Aircraft Systems.

Future research is needed to further define each category of airworthiness assurance as related to UAS airframe technology so that general guideline material may be developed in light of these vast differences in UAS operational space. A large percentage of current UAS platforms are fixed-wing aircraft that are constructed using composite materials. Two particular areas of how further research investigations may proceed to further define these UAS airworthiness assurance guidelines that would be insensitive to UAS size, weight, or performance characteristics. The first area is material and process control, and the second is Load spectrums for UAS vehicles.

For material and process controls consider the following. A key component of the fabrication process for composite materials is to include the material properties of the composite into the structure. This is different from metallic materials. Hence, it is critical that the material and process specifications be very complete when producing composite structures. Acknowledging the key parameters of the fabrication process is fundamental to satisfying production control and adherence to engineered parts. Recently, the aviation industry has had an overwhelming variety of composite materials requiring certification for final use within a manned vehicle. This certification brings forth the need to control the materials to ensure overall safety.

Major components of the spectrum development include defining the operational usage model, developing maneuver loads, and integrating the two into a format suitable for analysis. UAS load spectrum may be substantially different from the methodology applied in fixed-wing manned aircraft. Typical usage describes aircraft configurations, types of maneuvers, and their expected frequency of occurrence in service. From a design perspective, usage is typically prescribed by the procuring agency via mission mixes and profiles, and is assumed to be known or given prior to load spectrum development. The focus of this proposed program will be to delineate a typical load spectrum for several UAS configurations and define what existing damage-tolerance methodology should be applied to guarantee satisfactory airworthiness assurance for a UAS vehicle operating within the NAS, regardless of size, weight, or performance characteristics.

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Course Development: Composites Awareness for the Aviation Safety Inspector

Principal Investigator: Charles Seaton, Edmonds Community College

Statement

The purpose of this project is to customize the previously developed course under Phases I through IV for FAA aviation safety inspectors. The previously developed course, titled Critical Composite Maintenance and Repair Issues, was an awareness course funded by the FAA in partnership with industry and academia, and provided by Edmonds Community College between August 2004 and December 2007.

This course will provide to students an awareness of safety issues regarding the maintenance and repair of composite materials utilized in aerospace with emphasis on regulatory issues. In addition, students will receive an inspector checklist which will be used as a job aid. As an awareness course, students are expected to develop an appreciation for the safety implications of the use of composite materials in commercial aerospace and in so doing develop a broad familiarity of composite materials technology.

Project background and motivation

The FAA currently employs 2072 ASIs in the Airworthiness Inspector specialties and 500 ASIs in the Manufacturing Inspection District Offices (MIDO). These inspectors perform inspection and surveillance of facilities involved in the manufacture and repair of critical aircraft structural composites, among other duties. Current regulations and policies require that the Flight Standards Service provide training of these ASIs to develop and maintain knowledge and skills necessary to perform their inspector job functions.

Based on recent developments in aircraft industry including advancements in manufacturing techniques and materials that use advanced structural composites, FAA has determined that training is required to provide ASIs with the knowledge and skills required for oversight and auditing of aircraft manufacturing and maintenance facilities for composite aircraft, to include both line maintenance and overhaul maintenance of aircraft such as Boeing 787, Cirrus, Diamond, Eclipse and Airbus models A-350 and A-380.

Approach

The course to be developed consists of two training outcomes.

In the first training outcome, students will be brought to a common level of understanding of composite materials technology during an initial online, self-study learning experience, culminating in an exam which students must pass in order to advance to the primary course material. Students will be able to take the exam as an open-book, and may re-take the exam as many times as necessary. The exam is the primary learning and retention tool to achieve this training outcome, referred to as the "Prerequisite: Basic Knowledge of Composite Materials Technology, Repair and Approved Data". Expected time commitment for the students is 20 hours over a two week period, and requires access to a personal computer.

After passing the Prerequisite Course exam, students will attend the second training outcome at a regional location in a traditional classroom format, combined with a laboratory, referred to as "Awareness of Safe Maintenance Practices with Composite Structures". This learning experience will require six days, in a classroom of not more than 15 students. Students will become aware of composite materials technology and how these principals are related to FAA regulations.

The following tasks are incorporated in this grant:

Customize the previously developed "Critical Composite Maintenance and Repair Issues – Prerequisite" web-based training course to the specific needs of aviation safety inspectors (ASI)
Set up and maintain a distance learning collaboration medium
Identify four (4) geographically dispersed training sites
Analyze and Report on student inputs from the Phase I – IV course
Evaluate and Report on prior Composites ASI training courses
Support SME Work Group Meeting on revisions to prototype training materials developed in Phase IV
Develop Training Development Plan
Develop Course Design Guide
Develop Testing Plan and Tests
Develop Training Materials (including revisions to prototype training materials developed in Phase IV)
Conduct Prototype Training Class and Incorporate Revisions as Required

A workshop was attended with representative ASIs and FAA curriculum development specialists in February 2008. A revised list of course objectives and a draft inspection checklist were completed, and form the basis for the revised course.

In parallel with the curriculum development, training sites have been identified as candidates for composite training, with a scorecard for performing evaluation. A target number of acceptable candidates for potential selection are four locations, and visitations are scheduled for those who are qualified.

The grant will be completed with a 6-day class in Reno, NV conducted at Abaris Training Resources facility.

Damage Tolerance and Durability of Adhesively Bonded Composite Structures

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[1] School of Mechanical Engineering, Purdue University
[2] School of Aeronautics and Astronautics, Purdue University

Predicting the Strength of Adhesively Bonded Structures—A Cohesive Zone Model Approach

Recent research effort has focused on the prediction of strength (failure load) of adhesively bonded composite structures. Past effort was related to the techniques of basic characterization of adhesive fracture and crack growth response, and these techniques were applied to characterize the adhesive Hysol EA9394). We have now employed that material characterization to predict the structural strength of several example composite structures including adhesive bonds. Three examples are described.

For double cantilever beam specimens a pronounced dependence of the crack growth resistance and the load carrying capability was observed experimentally. We have developed a computational fracture mechanics model that explains this finding. In particular, it is demonstrated how crack path selection can be influenced strongly by the variation in constraint arising from variation in the bondline thickness. Figure 1(a) depicts the experimental evidence of crack path deviation for low bondline thickness, while Figure 1(b) shows the related computational result.

(a)

(b)

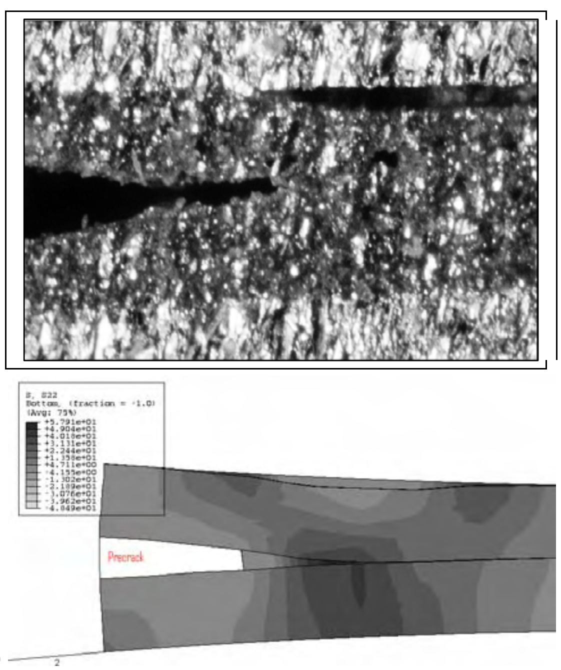

Figure 1: (a) Experimentally observed crack cohesive to adhesive transition; (b) computational model result.

For L-joints a pronounced dependence of the joint strength on filled radius was reported. We have developed computational fracture mechanics models that explain this finding. In particular it is demonstrated how crack initiation is influenced by the variation in fillet radius.

For lap joints an optimized joint strength can be obtained through geometrical variation in joint geometry. We have developed computational fracture mechanics models that can be used as design tools to optimize the joint strength. We demonstrate that a both a wavy joint as well as a reverse bent joint would possess high strength than a conventional lap joint. Implication of this finding to joining of sandwich panels is discussed.

Prediction of Lap Joint Strength – A CTOA Approach

In addition to a highly nonuniform interfacial stress distribution, the strength of adhesively bonded lap joints also depends on the thickness of the adhesive layer. For many years, researchers have been trying, but without much success, to develop a method that can accurately predict the strength of adhesively bonded lap joints. The objective of this research is to explore the use of crack tip opening angle (CTOA) as a crack extension criterion to predict the strength of lap joints. It is assumed that there exist small defects in the form of cracks in the adhesive layer. These cracks would trigger the failure process when the lap joint is under a sufficiently high load level so that the CTOA reaches the critical value. The critical CTOA for crack extension was first measured using double cantilever beam (DCB) specimens with various adhesive thicknesses. It was shown that the critical CTOA is not sensitive to the variation of adhesive thickness. To predict the strength of lap joints, a small crack (defect) was assumed to exist near the adhesive/adhered interface at the joint edge where the interfacial normal stress attains the maximum value. Thus, a single critical CTOA value can be utilized as the failure criterion and used to predict fracture for joints with different adhesive thicknesses. However, when the adhesive layer becomes very thin, the increasingly high interfacial stresses between the adherend and adhesive layer may become the dominant failure mechanism and cause a premature failure. As a result, the strength of the joint would be controlled by the interfacial failure and not by fracture. The present approach also been verified experimentally with Hysol EA9394 adhesive.

Education and Training

During the past year a one semester long advanced course with the title "Computational Fracture Mechanics" has been developed. The course was offered as an advanced graduate course in the School of Mechanical Engineering at Purdue University. The entire course material is available in electronic format. The course syllabus contains the following main chapters: (1) Review of classical fracture mechanics concepts for elastic materials; (2) Computational methods for classical fracture mechanics for elastic materials; (3) Computational methods of crack growth in elastic solids (including modeling with cohesive zone models, model generation, analysis, convergence criteria, fracture and fatigue); (4) Review of classical fracture mechanics concepts fro nonlinear material; (5) Computational methods for nonlinear fracture mechanic; (6) Continuum damage mechanics concepts and computational aspects. The course material includes a wide variety of example problems to be executed with the ABAQUS FE-code. The course can be offered for training to industry.

Development of Reliability-Based Damage Tolerant Structural Design Methodology

Principal Investigator: Kuen Y. Lin, University of Washington

Problem Statement
The overall objective of this project is to develop a probabilistic method to estimate structural component reliabilities suitable for design, inspection, and regulatory compliance. The research program spans a three year period, consisting of two phases of study. The first phase focused on the development and validation of the methodology. The second phase has concentrated on applications of the developed technology, such as inspection scheduling and maintenance service guidelines.

Project Background and Motivation
The development of this methodology is motivated by the increasing use of composite materials in aircraft structures. The current metal structural design and certification philosophy is based on a quasi-deterministic approach. The essence of this approach is that "if we can reliably observe the damage growth, this damage is tolerated until it becomes critical for strength". In this approach, the designer provides for the slow damage growth, and maintenance provides for its detection and repair. The fatigue damage accumulation, crack growth rate and inspection capability determine the maintenance schedule. Usually inspections become more frequent at the end of life, when crack initiation is more probable. As fatigue cracks are initiated in places with high tensile stress, inspectors will know when and where to inspect based on fatigue test results and service experiences.

Due to the large uncertainties related to composite material properties, structural behavior, and operational environments of composite structures, current design practices call for extra safety factors to cover such uncertainties, resulting in conservative designs and service guidelines. The proposed method will focus on a probabilistic model for (a) estimating the reliability of composite aircraft structures subject to accidental and fatigue damages as well as material degradation, and (b) for minimizing the life-cycle cost of inspection/repair based on reliability.

As the utilization of composite materials in primary aircraft structures is becoming more widespread, it is important to determine the risk associated with cumulative damage effects as measured by the probability of failure. The probabilistic method developed at the University of Washington by our team with support from the FAA and the Boeing Company encompasses a large number of variables to provide a realistic description of the problem, including material properties, environmental variables, damage statistics, maintenance practices, etc, and it will be considered here as the basis for the static failure risk analysis. The approach provides a risk assessment of the structure, and feedback information for design and maintenance improvements.

The method is based on probabilistic structural lifecycle simulation with the consideration of all related parameters, such as mechanical loads and structural temperature with appropriate correlations, material properties with their dispersions, material properties versus temperature and environmental variables, damage statistics, residual structural properties versus damage type and size, residual stiffness and mass properties versus damage type and size, possible damage growth at high stress levels, probability of damage detection (POD) depending on inspection type, quality of repair (and its effect on strength, stiffness, and mass), and more. This method has been implemented as the REliability-based Lifecycle Analysis of Composite Structures (RELACS) software.

The accuracy of the probabilistic method depends on the accuracy of the probabilistic inputs. While it is easy to collect data such as damage statistics, there is currently no data, nor method to convert these damage statistics into structural response. That is, given a probability that a number of damages may occur at random location on a structural component, we need to find the corresponding structural failure strength and its distribution. This structural failure strength distribution can then be used as input for probabilistic risk analysis. A specialized interface has been developed with commercial FEA software to perform the task of characterizing the distribution of structural strength due to damages and other factors. This has been implemented as the UW Virtual Testing (UWVTL) tool for composite structures.

Approach

The University of Washington's Virtual Testing Tools for Composite Aircraft

The Virtual Test Laboratory (VTL) tool has been developed, which is software for prototyping of large-scale structural systems. It allows proof-testing of designs prior to or in parallel with structure development. The applications driving development of the VTL are certification testing of advanced composite structures of civil aircraft. Traditionally, stress and failure analyses are performed with computational tools, while structural testing of various complexities would be performed to validate the computational tools and analysis model. However, the real structure would always be different from the deterministic model and analysis predictions. From the statistical point of view, the difference does not necessary mean analysis model is wrong or the structure is under-designed. If that difference falls within the possible scatter of test results, the analysis model and structural design could be demonstrated as sufficient, as long as the differences are explained. Instead of performing a large number of testing, which would be cost prohibitive, it would be must easier if the large number of testing could be performed analytically, which then predicts the range of possible and acceptable outcomes for the real structural test. This is known as virtual testing, and it is clear that a virtual test facility would be very useful especially for composites with wider scatter of material properties.

An important aspect of virtual testing is the ease in which the effects of changes in structural properties and material parameters can be examined. The VTL is designed to accelerate analysis and input data variation for FE structural models. The Virtual Strength Test Module (VSTM) has been developed for this purpose.

VSTM imports a deterministic FE model, with all the relevant information such as element section and material properties. With statistical scatter and correlation data obtained from coupon testing or manufacturing data (e.g. thickness, stiffness, strengths, etc), the material cards of the FE model is randomized realistically according to the scatter information collected. Damages can also be modeled by appropriate reduction of material properties. Given good scatter and correlation data, these FE models would adequately represent a random collection of real structures that would be manufactured, with the appropriate method and materials. A large collection of these random structures (the "population") would then be analyzed for strength margins, thus building a statistical distribution of structural responses that would correspond to the testing results of a population of real structures if they were to be testing to failure. The results can further be processed to determine the sensitivity of structural response to the various random parameters, which can greatly aid designers.

VSTM is a virtual strength test module designed primarily for characterization of the scatter of the strength margin obtained in static tests. Many reliability tools including RELACS require statistical input data, which are not routinely obtained in current design and substantiation processes. Some data may be obtained through extensive and expensive testing programs. The accuracy of probabilistic analysis relies heavily on the accuracy of input. When experimental data is not available from full-scale physical testing, results obtained by stochastic FEA and other deterministic methods with random inputs may be used instead as virtual experiments.

In order to provide the input data for RELACS, present work considers the influence of stiffness uncertainties on the behavior of two instances: (1) a simple trapezoidal wing of constant thickness made of quasi-isotropic material (Figure 1) and (2) an elastic vertical tail with a rudder made of CFRP (Figure 2). The FEA model for the trapezoidal wing has only 2055 degrees of freedom, it has been used for sensitivity and other parametric studies. As this realistic structure was assembled from several independently manufactured subcomponents, direct simulation of average panel properties and Markov field simulation has been used.

Figure 1 – Simple Model of Trapezoidal Wing

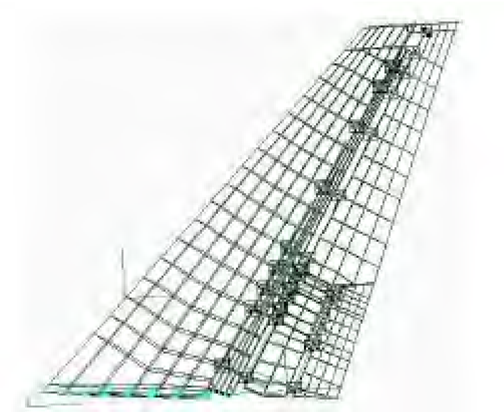

Figure 2 – Realistic Model of Vertical Tail with Control Surfaces

RELACS – Time-Dependent Virtual Testing

RELACS is intended for virtual life testing. A large number of structural damage size life histories (as well as material degradation, structural changes due to environmental effects and inspection/ repair program) are simulated using randomized input parameters. These are converted into the histories of residual structural property subject to environmental exposures, repairs, and other factors. The residual property may be strength, stiffness, flutter speed, vibration level, etc. depending on the considered failure mode. Random loading and flight conditions are also generated in the form of mechanical load/ temperature/ airspeed. The probability of generalized loading exceeding the generalized strengths is the probability of failure (POF).

The University of Washington's general and flexible method and corresponding software Reliability Lifecycle Analysis of Composite Structures (RELACS) can be used to determine any controllable parameters in the structural system to achieve a prescribed level of reliability. Design parameters that affect structural reliability can be actual sizes (thicknesses) of structural elements, topology and load paths in the structure, material type, joint type and strength, etc. Additionally, inspection/ repair planning is particularly important for meeting damage tolerance requirements and managing life-cycle risks.

To date, several practical and important real-case example problems have been demonstrated, in particular:

- Reliability-based damage-tolerance of composite aileron of commercial aircraft;
- Reliability-based damage-tolerance of bonded skin-stringer panel of a commercial aircraft (Figure 3) using Virtual Crack Closure Technique (VCCT) in ABAQUS;
- Optimum maintenance planning for vertical tail of commercial aircraft;
- Selection of Certification Tests for Composite Structures Using Optimal Statistical Decision (Table 1).

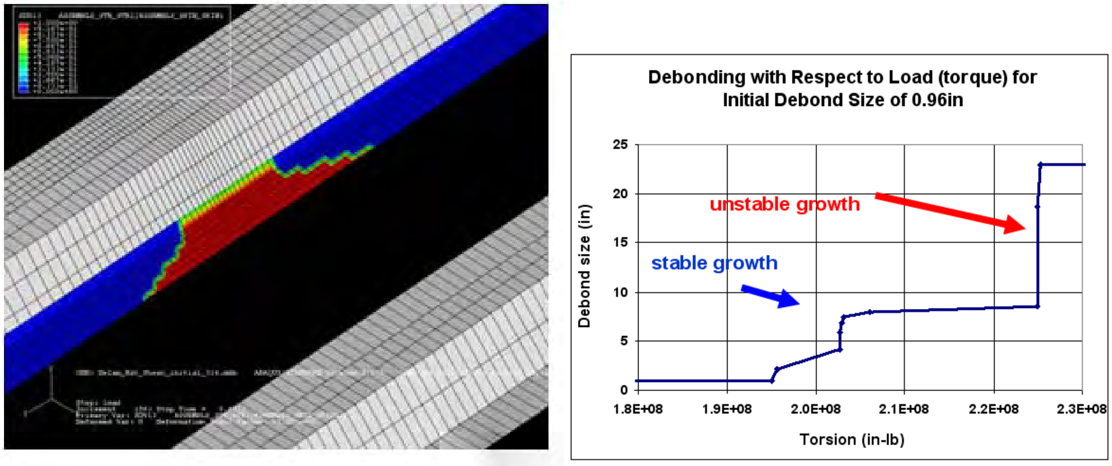

Figure 3 – Fuselage Skin-Stringer Model and Example of Disbond Damage Growth Characteristics

Table 1 – Comparison of Certification Tests and Their Corresponding Utility (Cost)

Metal structure: model mean COV = 5.2%, Fleet articles COV = 4%, Test Stress COV = 1%				
	Mean	C_V, %	POF	Utility
No certification tests = e_3	**0.9359**	6.7	2.2E-2	4.40E+00
Ultimate Load Tests = e_2	1.0275	6.7	4.59E-3	1.92E+00
Analysis Supported by Tests = e_1	1.1509	4.3	5.06E-4	1.01E-01
Composite structure: model mean COV=10.4%, Fleet articles C_V=8.1%, Test Stress C_V=2%, Overload =1.308				
No certification tests = e_3	0.9303	12.7	6.5E-2	1.30E+01
Ultimate Load Tests = e_2	1.0553	11	2.18E-4	1.04E+00
Analysis Supported by Tests = e_1	1.3239	8.5	3.79E-4	7.58E-02
Composite structure: model mean COV=10.4%, Fleet articles COV=8.1%, Test Stress COV=10%, Overload =1.308				
No certification tests = e_3	1.353	16.07	3.02E-3	6.04E-01
Analysis Supported by Tests = e_1	1.424	15.1	1.02E-3	2.02E-01

Future Research Direction

The disbond and delamination of composite laminates are of particular importance in terms of safety of large integrated composite structures (e.g. fuselage skin-stringer in Figure 3). However, to support the probabilistic analysis methodology introduced above, an efficient way of analyzing these disbond and delamination problems stochastically is needed. This is important because disbond and delamination may occur in any location and ply interface, yet the design must provide adequate safety level for all possible scenarios and load cases. This increases the size of the problem by one to two orders of magnitude (compared to the examples for VSTM above). A full order stochastic FEA becomes time-prohibitive. It is necessary to develop an analytical method to analyze these disbond and delamination problems in a stochastic fashion. The computation efficiency promised by analytical methods could realize the industrial use of probabilistic methods for safety, design and optimization analysis of composite structures.

Damage Tolerance and Durability of Fiber-Metal Laminates for Aircraft Structures

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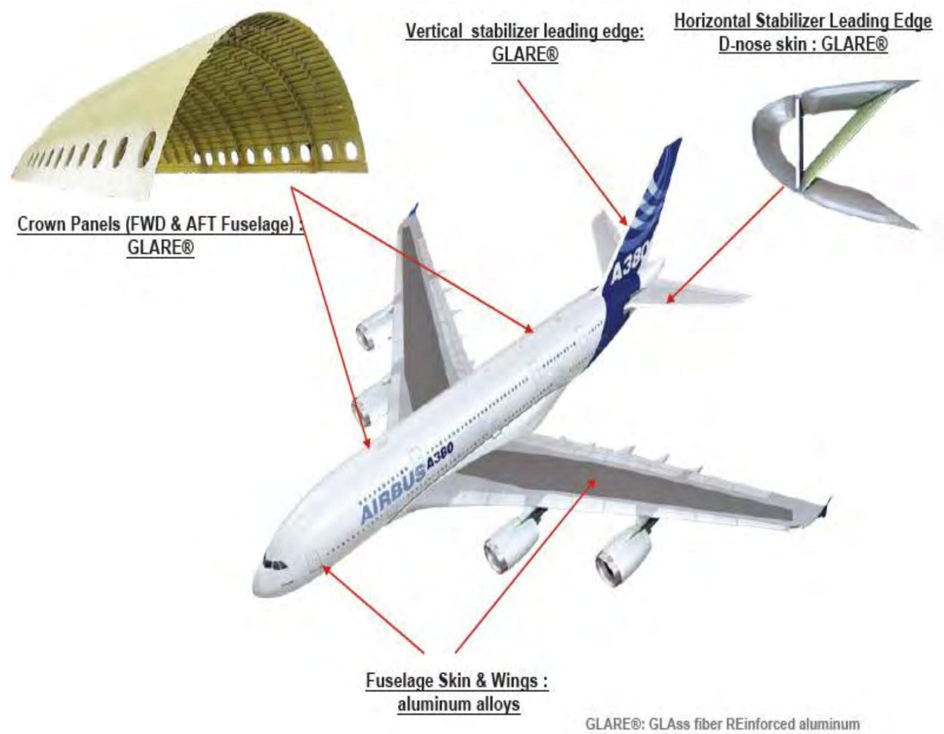
Project Background and Motivation

Fiber-reinforced metal laminates (FML) are hybrid composites consisting of alternating thin layers of metal sheets and fiber-reinforced resin prepreg. The most commonly used metal for FML is aluminum, and the fibers can be Kevar or glass. The FML with glass fiber (tradename GLARE) has been selected for applications in aircraft structures such as the crown panels and leading edges for A380 as shown in Figure 1. These laminates possess excellent properties of both metal and fibrous composite materials. This combination results in a new family of hybrid laminates with an ability to impede and arrest crack growth caused by cyclic loading, with excellent impact and damage tolerance characteristics and a low density. Also, the corrosion resistance is excellent because the prepreg layers are able to act as moisture barriers between the various inner aluminum layers, whereas the metal layers protect the fiber/epoxy layers from picking up moisture. As a result, GLARE laminates offer the aircraft structural designer a damage-tolerant, light-weight, cost-effective solution for many applications. GLARE laminates seem poised for a much larger future in the primary structure of pressurized transport fuselages.

The research and development activities to date have covered a variety of important aspects pertaining to mechanical properties of GLARE. However, there are still limited and insufficient information available about mechanical behavior of GLARE in published literature, especially for the cross-ply configuration of GLARE (GLARE 3, GLARE-4 and GLARE-5), and some areas still remains to be further verified by more detailed testing. Also, the damage tolerance and durability certification methodology of a GLARE laminate in comparison with a certification of aluminum structures needs to be established. The objectives of the proposed work are to investigate the damage tolerance and durability of bi-directionally reinforced GLARE laminates. Such information will be used to support the airworthiness certification of GLARE structures.

Approach to Solve the Problem

During 2004-08, we have conducted both experimental and analytical work to study the damage tolerance and durability of GLARE laminates with bi-directionally reinforcements. These include (1) open-hole notch strength, (2) impact behavior and damage development, (3) residual tensile strength after impact, (4) post-impact fatigue behavior, (5) off-axis fatigue behavior, (6) crack growth under constant-amplitude fatigue loading (7) crack growth under variable-amplitude fatigue loading and (8) multi-site fatigue damage. Specifically, during 2007-08, we have focused on modeling and validation of GLARE laminates



with multi-site damage (MSD) and numerical simulation of single and multiple impact behavior.
The results are briefly summarized below:

(a) Modeling and Validation of FMLs With Multi-Site Fatigue Damage

The multiple-site fatigue damage in metallic airframe has received considerable attention since Aloha airline accident in 1988. Many works had been conducted to study the multiple-site damage (MSD) problems in aircraft structures to avoid the catastrophic failure in flight. In monolithic aluminum alloy under fatigue loading, cracks emanated around the fastener holes and propagated. A number of neighboring cracks might coalesce to form a single dominant crack. When these fatigue cracks are present in the fuselage of an aging aircraft, significant load redistribution will occur especially ahead of the dominant cracks. The interaction of fatigue cracks will increase the stress intensity factor and enhance the crack growth rate in metallic alloys. Subsequently, the residual strength is reduced and the integrity of structural components is affected.

However, for fiber metal laminates, the crack propagation and fracture characteristics are essentially different from those of metallic alloys. When a fiber metal laminate was subjected to fatigue loading, cracks initiated and propagated in the metal layer of fiber metal laminates, and the crack opening in the metal layers was restrained by the intact fibers in the wake of the fatigue crack. The crack growth in the metal layer was impeded by the fiber bridging mechanism and resulted in far lower crack growth rate compared to monolithic aluminum alloy. As a result, it is anticipated that the influence of multiple-site damage on fiber metal laminates will be different from that of a metallic structure. A better understanding of the multiple-site fatigue damage behavior in fiber metal laminates, such as crack growth, cracks link-up, delamination link-up, etc, is necessary for ensuring the structural integrity.

The multiple-site fatigue damage behavior of fiber metal laminates was investigated experimentally and analytically under constant-amplitude fatigue loading. It was found that the presence of multiple-site fatigue cracks would accelerate the crack growth rates in the metal layers of fiber metal laminates as two propagating cracks approached each other as shown in Figure 2. An analytical methodology was proposed to calculate the fiber bridging stress based on the concept of virtual work. The predicted crack growth rate in the fiber metal laminate with the presence of multi-site damage was validated by the experiments as shown in Figure 3.

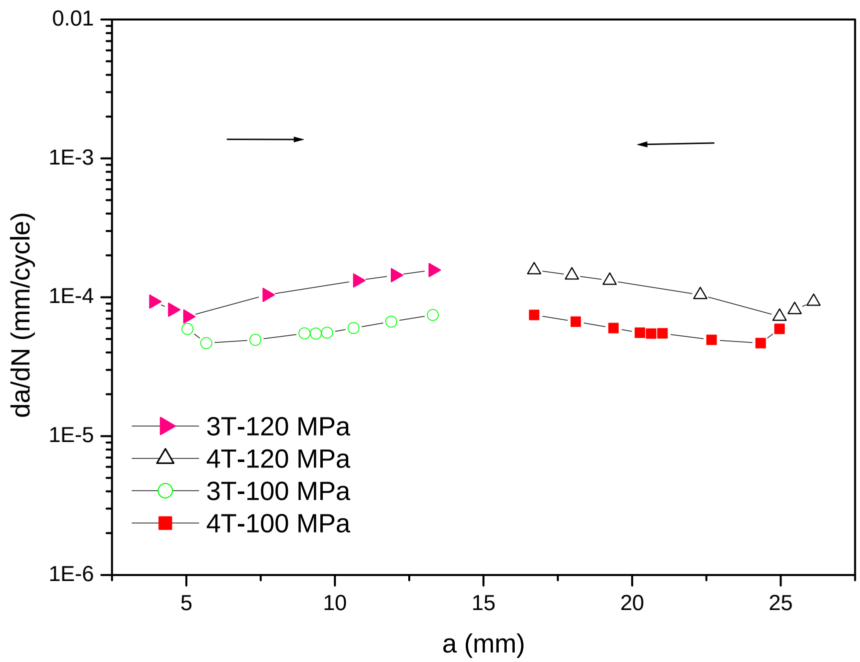


Figure 2. MSD crack growth rate in bottom row as a function of crack length at different applied stresses for Glare3-3/2.

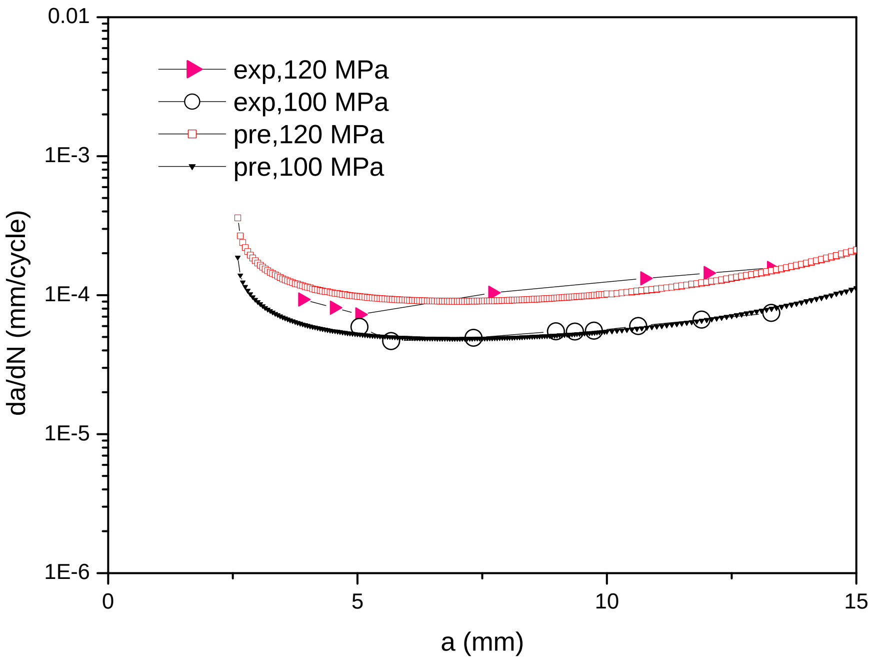
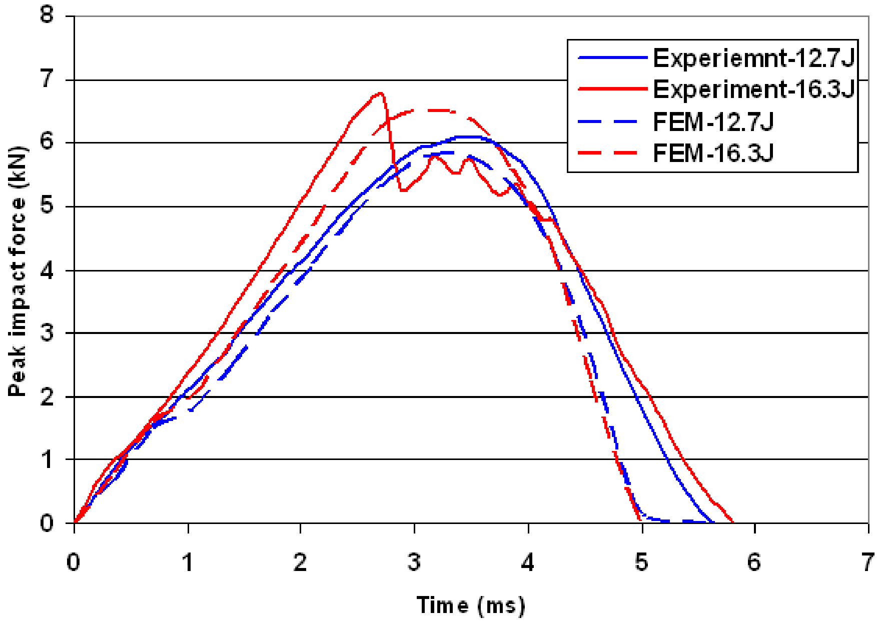


Figure 3. Comparison of MSD crack growth rates between predictions and experimental results as a function of crack length at different applied stresses for Glare3-3/2.

(b). Numerical Simulation of Single and Multiple Impact Behavior

The objectives of this task is to develop a numerical method to simulate impact damage under diverse impact energies and to predict peak impact force and stress at the outer aluminum layer and inner composite layer. For numerical analysis, the commercial software ABAQUS was used. To check the validity of the predicted results, numerical results were compared with experimental data. As shown in Figure 4, numerical and experimental results of peak impact force vs. time showed good agreement under different impact energies for both Glare-4 and Glare-5 laminates. The stress and strain incurred during impact at the outer aluminum and inner composite layer had also been analyzed. The effect of multiple impact on the damage development of the monolithic aluminum alloy and FMLs was also simulated.

(a)



(b)

Figure 4. Numerical simulation of impact behavior for (a) Glare-4 and (b) Glare-5 laminates

(c). Information System for Certification

Along with the analytical and experimental part of this work, an information system for damage tolerance design, and certification of GLARE laminate was developed. This system is based on the knowledge database that contains results of current experimental program as well as summary of the experimental data available in the literature. This information system will facilitate retrieval of critical data during design process and in making certification decisions regarding damage tolerance and durability of GLARE structures.

Wednesday, June 18, 2008

Structure Health Monitoring for Life Management of Aircraft

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1. Problem Statement: The objective of this work is to develop a structural health monitoring system for impact damage of a composite structural panel. The overall approach involves the following steps:
1. A laminated composite panel suffers an impact load which causes delaminations.
2. A structural health monitoring (SHM) system, consisting of ultrasonic sensors, detects the delamination and estimates the area(s) of delamination.
3. Under the influence of cyclic service loads, the delamination may grow.
4. The increasing size of delamination area(s), A, which is treated as the damage parameter, is monitored by the SHM system.
5. A probabilistic fatigue damage theory predicts the number of cycles at which the delaminations reaches a critical size.

2. Project Background and Motivation: Substantial efforts are being directed toward implementing efficient health monitoring procedures for aircraft structural components. SHM systems aim to minimize the economic loss generated by unnecessary and premature replacement of expensive parts, without compromising safety. Moreover, the risk of unexpected failure is dramatically reduced when the health monitoring techniques are implemented. The accelerated trend of replacing conventional materials with composites in aircraft construction requires the development of SHM systems that are appropriate to monitor the type of damage that the composite parts are likely to suffer. In this work, Lamb waves are used as an efficient ultrasonic method to investigate damage in composite components.

3. Work to date: Our effort during the first three years of this project involved:
- acquisition/fabrication of various composite specimens with seeded and real defects;
- development of modally-selective Lamb wave transducers;
- development of series/parallel configuration of array transducers for enhanced signal to noise performance;
- development of self-powered sensors using energy-harvested power;
- systematic study of detection and size estimation of mid-plane seeded delaminations in woven quasi-isotropic carbon-epoxy composites;
- Study of impact-induced delaminations.

In the past year, we were on a no-cost extension. During this period, additional work involved:
- further development of narrowband Lamb-wave sensors;
- investigation of the effect of transducer degradation vis-à-vis structural degradation; and
- investigation of a computer software program called GENOA for application to composite damage evolution prognosis

Selected results are discussed in greater detail next.

4. Details: In structural health monitoring, it is typically assumed that any change in the measured signal from a previously measured baseline signal is due to an anomaly in the structure. In reality this is not always the case since the changes could be due to transducer degradation. For piezoelectric transducers, the electrical admittance is a function of the mechanical impedance of the structure and transducer in addition to the electrical impedance of the transducer. This means that the stresses at the transducer interface changes the imaginary part (capacitance) of the admittance. It is important to ensure that the resulting changes are not wrongly interpreted as damage in the structure when in reality the damage exists in the transducer. It is therefore important to monitor the health of the transducer in order to perform accurate and efficient structural health monitoring (SHM). The admittance (capacitance) of the transducer changes due to two primary factors: (a) Damage to the transducer (b) Changes associated with the bond between the transducer and the host structure. These are reflected as changes in stresses that modify the capacitance of the transducer. Detailed theoretical analysis using the finite element method is performed to understand the variation of capacitance due to changes in the stress between the transducer and the structure. Changes due to damage to the transducer are not reported in this paper.

Three cases were investigated. These are: (a) Free unclamped transducer with zero stress in all directions (b) completely clamped transducer with zero strains in all directions and (c) transducer bonded on the surface of a host structure. For cases (a) and (b) the distribution of stresses and strains in the transducer is homogeneous. Linear piezoelectric equations are used to identify the changes in capacitance between cases (a) and (b). These are verified with the finite element model. An experimental prototype of a microcontroller monitoring the change in capacitance based on the theoretical studies was developed

The important conclusions reached regarding transducer health monitoring are: (a) Change in capacitance is associated with increase in the stresses associated with the transducer; (b) For a surface bonded transducer, the in-plane stresses contribute to the maximum change in capacitance; (c) The change in the thickness of the transducer does not significantly affect the capacitance.

THEORY – Transducer Health Monitoring

Figure 1 shows a piezoelectric transducer, perfectly bonded on the surface of a host structure. Electrodes are present only on the surfaces of the transducer and hence the electric field in the Y (3) direction is $E_3 = \frac{V}{t}$ where V is the voltage applied across the thickness (t) of the transducer. Shear stresses and shear strains are neglected and a simplified model of the linear piezoelectric constitutive equations relating stress, strains, electric field and electric displacement is [3]

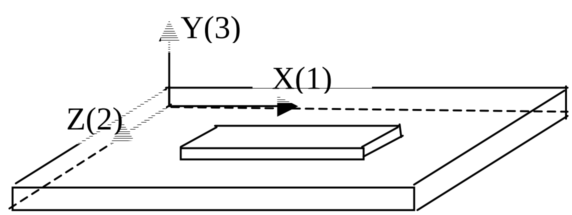

Figure 1. Geometric model of the piezoelectric transducer surface bonded on a host structure.

$$\begin{Bmatrix} D_3 \\ S_1 \\ S_2 \\ S_3 \end{Bmatrix} = \begin{bmatrix} \varepsilon_{33}^T & d_{31} & d_{32} & d_{33} \\ d_{31} & S_{11}^E & S_{12}^E & S_{13}^E \\ d_{32} & S_{21}^E & S_{22}^E & S_{23}^E \\ d_{33} & S_{31}^E & S_{32}^E & S_{33}^E \end{bmatrix} \begin{Bmatrix} E_3 \\ T_1 \\ T_2 \\ T_3 \end{Bmatrix} \quad (1)$$

where $(E)_{3x1}$ is the applied electric field (unit: V/cm), $(D)_{3x1}$ is the electric displacement (unit: coulombs/m^2), $(S)_{6x1}$ is the strain (unit: m/m), and $(T)_{6x1}$ is the stress (unit: Pa). Superscript T indicates that the material property is measured under a constant stress condition (zero stress) and superscript E indicates that the material property is measured under constant electric field (zero electric field is obtained by shorting the two electrodes). $(\varepsilon^T)_{3x3}$ is the dielectric permittivity measured under constant stress and relates dielectric displacement to the electric field (units: F/m). The parameter relating strain to the stress is the piezoelectric strain coefficient $(d)_{3x6}$ (units: m/V) and $(S^E)_{6x6}$ is the compliance matrix relating strains to the stress (unit: m^2 / N).

To analyze the change in the capacitance of the transducer three cases are studied. These are (a) Unclamped transducer with zero stress in all directions; (b) completely clamped transducer with zero strains in all directions; and (c) transducer bonded on the surface of a host structure.

Case (a): Unclamped transducer *(Stresses in all the directions are zero)*

The electrical displacement (D_3) for an unclamped transducer follows from equation (1)

$$D_3 = \varepsilon_{33}^T E_3 \quad (2)$$

$$C_{free} = \frac{Q}{V} = \frac{\varepsilon_{33}^T A}{t} \quad (3)$$

where A is the area of the electrodes, C_{free} is the capacitance of a free transducer which can be measured experimentally using a commercially available capacitance meter or using a simple RC (known resistor (R) capacitor (C)) circuit, and Q is the charge developed on the electrodes of the transducer. Equation (3) indicates that capacitance is a function of the dielectric property, the area of the transducer and the thickness of the transducer. Any change in these properties effectively changes the capacitance associated with the transducer. The area of the electrodes remains constant throughout the study and hence it is expected that a change in dielectric property and the variation of thickness contribute to the change in the capacitance of the transducer. In addition to the above, strains S_1, S_2 and S_3 get generated because of the dependence of strains over electric field (E_3) through the piezoelectric strain coefficients d_{31}, d_{32} and d_{33}, respectively. The strains obtained with zero stress is called free strain and corresponds to $S_1 = \frac{d_{31}V}{t}$, $S_2 = \frac{d_{31}V}{t}$ and $S_3 = \frac{d_{33}V}{t}$.

Case (b): Rigidly clamped transducer *(Strains in all the directions are zero)* The electrical displacement (D_3) for a transducer obtained from equation (1) with homogenous state of stress and strain yields:

$$D_3 = E_3\varepsilon_{33}^T[1 + \frac{d_{31}T_1}{E_3\varepsilon_{33}^T} + \frac{d_{32}T_2}{E_3\varepsilon_{33}^T} + \frac{d_{33}T_3}{E_3\varepsilon_{33}^T}] \quad (4)$$

where $T_3 = \dfrac{S_3 - d_{33}E_3 - S_{31}^E T_1 - S_{32}^E T_2}{S_{33}^E}$, $T_2 = \dfrac{\left[S_2 - \dfrac{S_{23}^E S_3}{S_{33}^E} + \dfrac{S_{23}^E d_{33}E_3}{S_{33}^E} - d_{32}E_3\right] - T_1\left[S_{21}^E - \dfrac{S_{23}^E S_{31}^E}{S_{33}^E}\right]}{\left[S_{22}^E - \dfrac{S_{23}^E S_{32}^E}{S_{33}^E}\right]}$,

and $T_1 = \dfrac{S_1 - d_{31}E_3 + \left[\dfrac{S_2 - AS_3 + BE_3 - CE_3}{F}\right][S_{13}I - S_{12}^E] - S_{13}^E\left[\dfrac{S_3}{S_{33}^E} - GE_3\right]}{\left[J + \dfrac{S_{13}^E ID}{F}\right]}$ and $A = \dfrac{S_{23}^E}{S_{33}^E}$, $B = \dfrac{S_{23}^E D_{33}}{S_{33}^E}$, $C = d_{32}$,

$D = S_{21}^E - \dfrac{S_{23}^E S_{31}^E}{S_{33}^E}$, $F = S_{22}^E - \dfrac{S_{23}^E S_{32}^E}{S_{33}^E}$, $I = \dfrac{S_{32}^E}{S_{33}^E}$, $G = \dfrac{d_{33}}{S_{33}^E}$, $J = \left[S_{11}^E - \dfrac{S_{12}^E D}{F} - S_{13}^E H\right]$, $H = \dfrac{S_{31}^E}{S_{33}^E}$.

Equation (4) can be rewritten as

$$\left(\frac{Q}{V_3}\right) = \left(\frac{A_e}{t}\right)\varepsilon_{33}^T[1 + \frac{d_{31}T_1}{E_3\varepsilon_{33}^T} + \frac{d_{32}T_2}{E_3\varepsilon_{33}^T} + \frac{d_{33}T_3}{E_3\varepsilon_{33}^T}] = C_{BONDED} \quad (5)$$

where C_{BONDED} is the capacitance of a bonded transducer which can be measured experimentally as before. Equation (5) indicates that the capacitance of a bonded transducer is directly proportional to the stresses T_1, T_2 and T_3 which are related to the strains $S_1, S_{2,}\, and\, S_3$ respectively. At an extreme condition of zero strain the term $\varepsilon_{33}^T[1 + \dfrac{d_{31}T_1}{E_3\varepsilon_{33}^T} + \dfrac{d_{32}T_2}{E_3\varepsilon_{33}^T} + \dfrac{d_{33}T_3}{E_3\varepsilon_{33}^T}]$ corresponds to ε_{33}^S which is titled as the dielectric permittivity measured under constant strain (zero strain).

Table 1. Comparison of the two models between two extreme cases (a) unclamped sensor (b) completely clamped sensor.

Model	Capacitance (nF)		% change
	Free	Clamped	
Finite element	1.875	0.793	-57.71
Mathematical model	1.874	0.790	-57.84

The capacitance changes between the conditions of free stress (unclamped free transducer) and free strain (rigidly clamped transducer with $S_1, S_{2,} S_3 = 0$) are compared in Table 1 between the above mathematical model and a finite element model (ANSYS). The material properties provided by the manufacturer are: (a) piezoelectric strain coefficient (meters/volt) $d_{33} = 390*10^{-12}$, $d_{31} = -190*10^{-12}$ (b) coupling coefficient $k_{33} = 0.72$, $k_{31} = 0.35$ (c) density (kg/m^3) $\rho = 7800$, (d) elastic modulus (N/m^2) $E_1 = 66*10^9$, $E_3 = 52*10^9$ and (e) Relative dielectric constant $K_T = 1800$. The dimensions of the transducer are (24 x 5 x 1.02) mm. An input of 100 volts is provided as input to the piezoelectric transducer. The comparison shows that there is a nearly 60%

change in capacitance of the transducer from its rigidly embedded (zero strain) to the fully unconstrained (zero stress) states. In reality, the sensor will be bonded on one side to a host structure, and an inhomogeneous state of strain and stress is expected. As the sensor debonds (or otherwise fails), it is expected that the stress state in the transducer will change which in turn will affect the capacitance of the transducer. Hence capacitance can be used to monitor the health of the transducer.

Case (c): Transducer bonded on the surface of a host structure

To study the variation of capacitance in this inhomogeneous case, the piezoelectric transducer is simulated in a finite element program. The transducer is perfectly bonded on the surface of an aluminum plate (Figure 1). The displacements are constrained to be zero at the bottom four corner nodes of the aluminum plate in the finite element model. The variation of capacitance as a function of change in length of the transducer is shown in Figure 2. The following parameters are maintained constant (a) width of the transducer: 5mm (b) thickness of the transducer: 1.02mm and (c) the host structure thickness: 5.1mm. If alpha is defined as the ratio between the transducers length to the thickness, large changes in capacitance are obtained at large alpha. As the length of the transducer increases, the normal stress Y(3) and shear stress on the bottom surface of the transducer decreases. The increase in in-plane stress contributes primarily to the change in the capacitance of the transducer. Also, at large alpha the in-plane stresses approach a constant value and hence the % change in capacitance reaches a constant value at large alpha. The other parameter that contributes to the change in the capacitance is the variation in the thickness of the transducer between the top and the bottom layers. The displacements in the Y(3) direction of the transducer for multiple lengths of the transducer is shown in Figure 3. For e.g, the difference in displacements between the top and the bottom layer of a 30mm long transducer long transducer is nearly 20nm. This small variation does not contribute to a significant change in the capacitance of the transducer. Hence the change in the capacitance is primarily due to the in-plane stresses within the transducer.

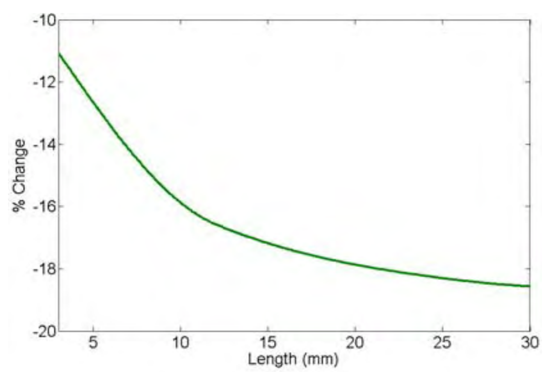

Figure 2. Variation of capacitance as a function of transducers length

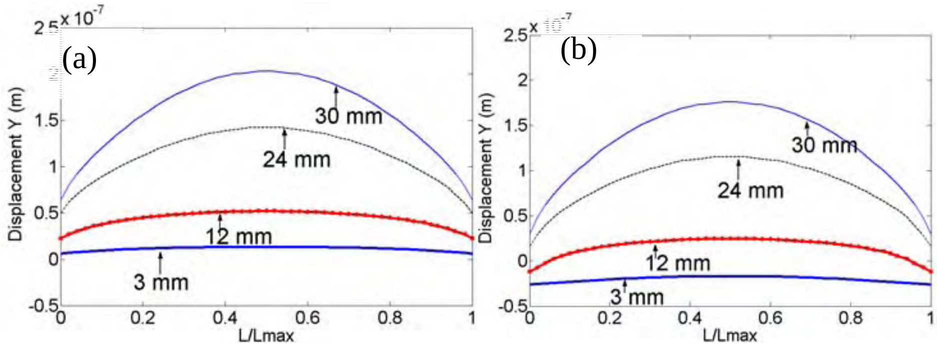

Figure 3. Variation of displacement in the Y direction (a) Bottom layer of the transducer (b) top layer of the transducer

The variation of capacitance as a function of change in the host structure thickness is shown in Figure 4a. The transducer dimensions are length: 24mm, width: 5mm and thickness: 1.02mm. Figure 4a indicates that the maximum change in capacitance possible for this transducer bonded on the surface of a structure is ~21%. It was shown previously that for a transducer embedded in a rigid structure the maximum change of capacitance due to large stresses is ~60%. Stresses in a surface bonded transducer are smaller compared to a completely rigid structure. Hence the change in capacitance is less than for a completely rigid case. Figure 4b shows the variation of capacitance as a function of the transducer thickness. The following parameters are maintained constant during this study: (a) Length of the transducer: 24mm (b) width of the transducer: 5mm and (c) thickness of the host structure: 5.1mm. As the thickness of the transducer increases the % change in capacitance decreases because of the decrease in the free strain of the transducer. Smaller free strain produces smaller stress and hence the change in capacitance is small.

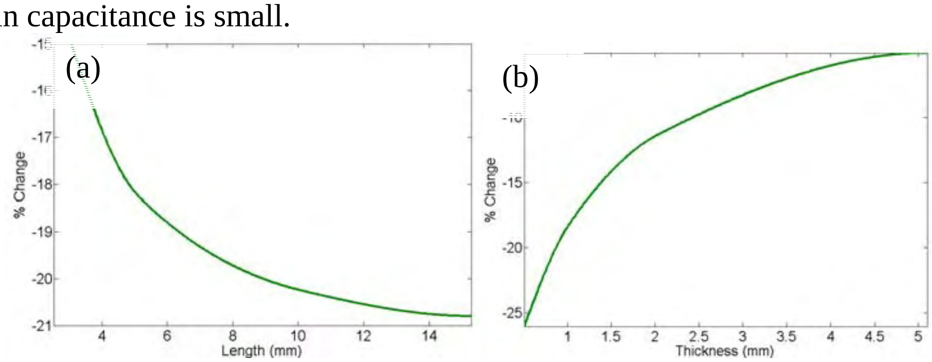

Figure 4. Variation of capacitance as a function of (a) host structure thickness (b) transducer thickness

EXPERIMENT – Transducer Health Monitoring

Piezoelectric transducers were bonded on a 25.9mm thick aluminum block. The approach of RC time constant is used to measure the capacitance instead of the admittance measurements reported in literature [2]. It is expected that the RC measurement has much lower power requirements on the associated electronics used to measure the change in capacitance. Arbitrary transducer dimensions were chosen to show the feasibility of the RC time constant measurement using a microcontroller to measure the capacitance of the transducer. The analog circuit shown in Figure 5a is used to excite the transducer with a 1Hz square input. The response is shown in Figure 5b. The time constant is measured at the 63% value of the full scale input (5V). The ratio of the time constant to the resistance is used to find the capacitance of the transducer. For example, the capacitance of unbonded transducer based on the measured time constant is 4.4nF. Once the transducer is bonded onto the structure the time constant decreases (0.035 m-sec) and the capacitance value is 3.54nF. The decrease in the capacitance is due to the large in plane stresses occurring in the transducer. The capacitance measured using RC circuit was verified using a commercially available capacitance meter. More experiments are currently in progress to compare the experimental capacitance variation to that shown by the finite element analysis as described in the previous section. Figure 6 shows a commercially available microcontroller performing a real time capacitance measurement based on a threshold value. In this example, any change of less than 2% in the capacitance measurement is considered as a good transducer and is indicated using "G" (Figure 6a). The other extreme is indicated using letter "B" (Figure 6b). More studies are currently being performed to determine realistic threshold voltages and incorporating the microcontroller with real time SHM devices.

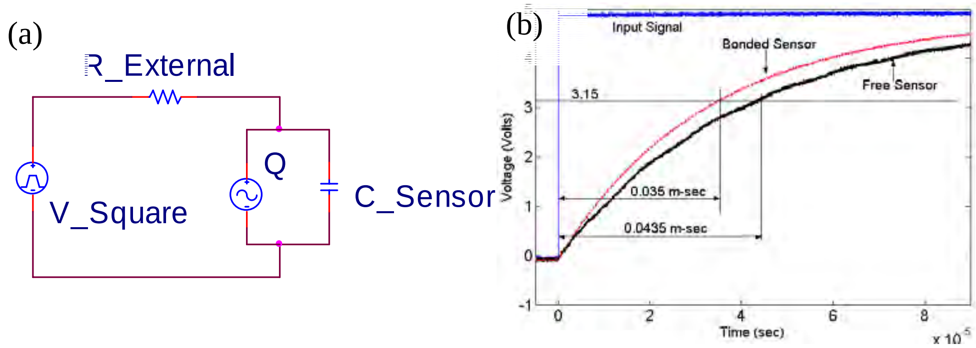

Figure 5. Capacitance measurement (a) Analog circuit used to measure the capacitance (b) RC time constant response of a piezoelectric transducer

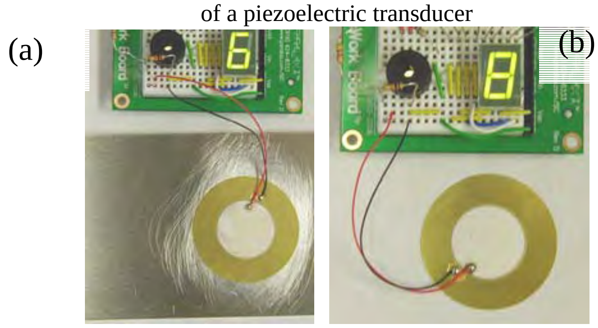

Figure 6. Transducer health monitoring based on a commercially available transducer (a) Microcontroller indicating "G" for a good transducer (b) Microcontroller indicating "B" for bonding issues with a transducer. The transducer shown here is unbonded.

Impact Damage Formation on Composite Aircraft Structures

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BACKGROUND. Impact damage resulting from collisions of ground vehicles/equipment with aircraft structural components, as well as from events such as hail and bird strikes, is a significant source of damage to commercial aircraft. Examples of ground vehicles and cargo equipment that pose as wide-area impact threats are shown in Figure 1. These include cargo and luggage containers, ground vehicles (fuel and service trucks), etc. With new all-composite fuselage transport aircraft coming into service, significantly more composite skin surface area is exposed to ground vehicles and equipment. To address the difficulties that exist in being able to predict and detect the damage resulting from blunt impact, and to aid in assessing its effect on structural performance, the development of basic tools to characterize blunt impacts is needed. Of particular interest is damage that can be difficult to visually detect from the exterior, but could be extensive below the skin's outer surface, as shown in the example of impact to a horizontal stabilizer by a lost access door (see Figure 2).

OBJECTIVES. The objectives of the proposed research will focus on impact damage formation by a range of sources, including: (i) low velocity wide-area impact – vehicle/ground maintenance collision, and (ii) high velocity hail, bird, and general impact:

Low-Velocity, High-Mass Wide-Area Impact:

1. Identify which blunt impact scenarios are commonly occurring and are of major concern to airline maintenance organizations and aircraft manufacturers.
2. Develop Methodology for Blunt Impact Threat Characterization.
3. Experimental identification of key phenomena and parameters governing blunt impact damage formation.

High Velocity Hail, Bird, and General Impact:

1. Investigate impact damage initiation and formation to composite panels, including those of skin-stiffened and sandwich construction.
2. Develop unified treatment methodology for predicting damage initiation by variety of impactor projectile types – e.g., bird, hail, tire fragment, runway debris, lost access panel, etc.

EXPECTED OUTCOME. Accomplishment of these objectives are intended to aid maintenance engineers in assessing whether an incident could have caused damage to a structure, and if so, what sort of inspection technique should be applied to resolve the extent of damage. Furthermore, it is expected that design engineers can make use of the research outcomes to: (i) improve the resistance of composite aircraft structures to damage from blunt impacts as well as a variety of other sources such as hail- and bird-strikes, runway debris, lost access panel, etc, and (ii) provide critical information on the mode and extent of seeded damage, particularly non visible impact damage, resulting from a wide gamut of impact threats – i.e., low to high velocity.

Figure 1. Maintenance/Service Threat Sources: Ground Vehicles, Cargo Containers, etc.

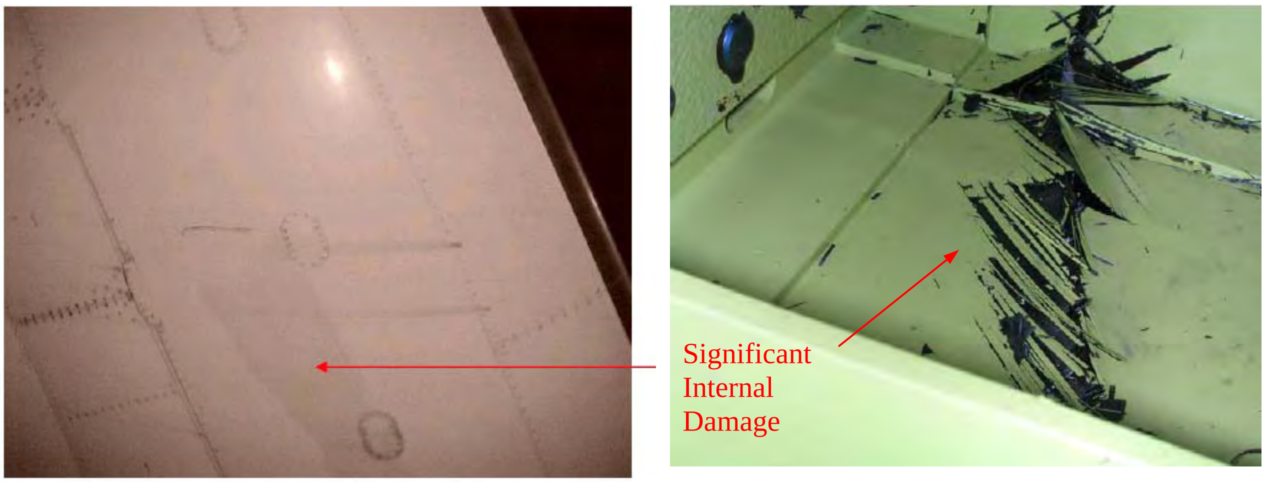


Figure 2. Significant Internal Damage to A330 Horizontal Stabilizer (Source: presentation by S. Waite at FAA Damage Tolerance and Maintenance Workshop, Chicago, July 19-21, 2006)

APPROACH. The stated objectives will be achieved via three major project tasks:

- Task 1. Identification of Common Impact Scenarios
- Task 2. Methodology for Impact Threat Characterization
- Task 3. Key Phenomena and Parameters Governing Impact Damage

Task 1 is the initial portion of this project. Surveys have been sent out requesting information regarding impact damage sources to aircraft, with particular emphasis on blunt impact damage sources and structures commonly damaged by blunt impacts.

Task 2 seeks to develop methodology for generally describing blunt, or wide-area, impact events. Simple models will be developed in support of this activity, with the objective of making predictions of impact contact force, energy absorption, and ultimately, help to estimate whether damage has occurred or not. Figure 3 shows a generic model of a wide-area impact event and the parameters to be considered in the model development. Additionally, curvature of the structure-target surface and curvature of the projectile affects the total contact area (and thus contact pressure), which could in turn control the visual detectability of any damage. Of critical interest is the condition in which a wide-area impact can produce a large amount of internal damage for least detectability on the impact side (i.e., outer surface).

Task 3 will experimentally identify key phenomena and parameters that govern blunt impact damage. The major difficulty of using the simple model described in Task 2 is knowledge of the stiffness and mass parameters and how these relate to damage initiation and development due to wide-area impact. While a series of blunt impact tests can be conducted in the laboratory for a range of contact force levels, such test results are of limited general use if the aforementioned models in Task 2 relating global parameters to local contact force information have not been established. Thus, the experimental activities will be closely coupled to the model development task, with the objective of producing a widely-applicable tool for wide-area impact damage assessment. Based on coordination with industrial partners, to include both OEMs and airlines, large scale tests will be conducted using representative stringer-stiffened panels and actual ground equipment, e.g., see Figure 4, as part of Task 3.

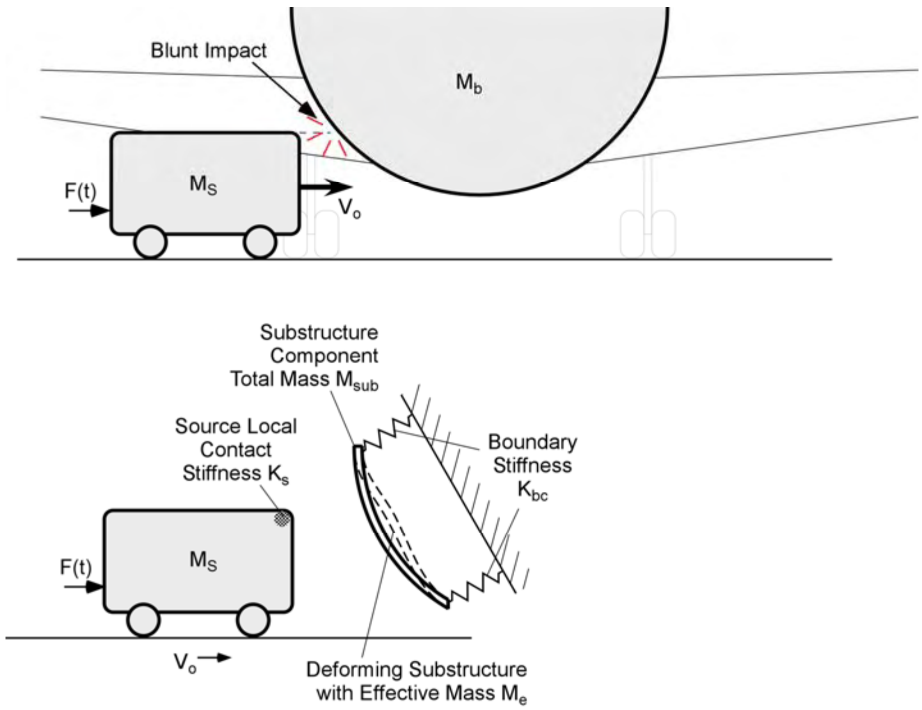


Figure 3. Wide-Area Impact Model Description

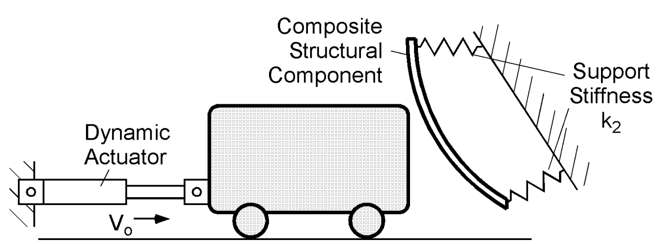


Figure 4. Large-Scale Test Configuration

Combined Global/Local Variability and Uncertainty in Integrated Aeroservoelasticity of Composite Aircraft

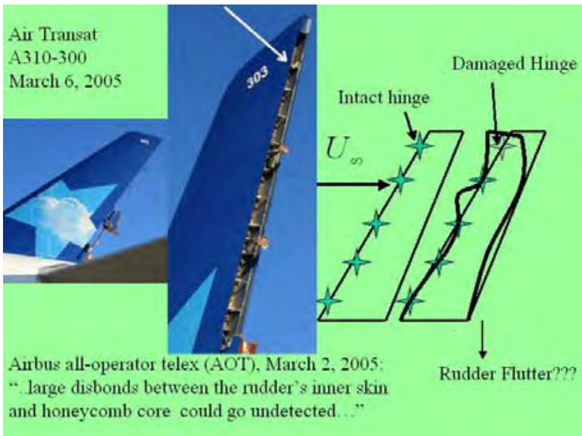
Principal Investigator: Eli Livne, University of Washington
Co-PIs: Mark Tuttle and Kuen Lin, University of Washington

Background:

With composite structures, potential sources of structural variation and deviation from original characteristics of an airframe over its lifetime in service are numerous: moisture absorption, crack and delamination progress, softening of bonded joints, damage due to impact, and material degradation resulting from radiation and other environmental effects. These variations and deviations from the nominal design may lead to stiffness and mass variation with time. They can start as localized effects, but develop to potentially affect the overall stiffness and mass distributions of major structural components. This may lead to increased loads caused by changes in aeroelastic deformation under load and to aeroelastic instabilities such as divergence and flutter.

The problem seems to be particularly severe for composite control surfaces. Over time, moisture absorption can lead to increased mass and inertia and lack of balance, and wear of hinges and linkages can lead to reduced stiffness or nonlinear stiffness of hinges. The combined effect might lead to flutter or limit cycle oscillations and fatigue.

Pictures of the damaged vertical tail / rudder assembly of an Air Transat Airbus A310-300 illustrate the problem. One possible scenario that can cause such failure is local damage that leads to a separation of a hinge, and as a result, to dynamic instability or destructive oscillations of the rudder on its other hinges.

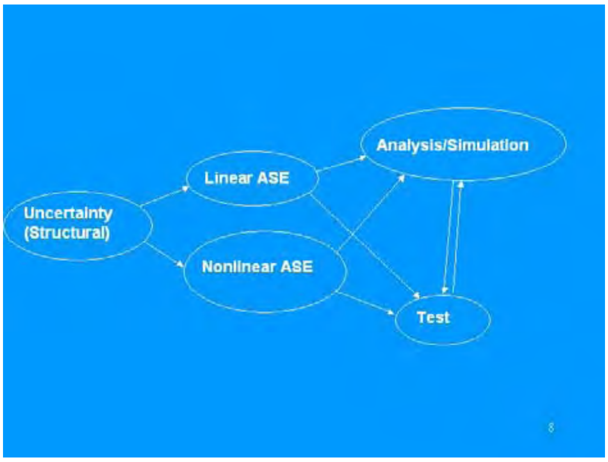


Problem:

- Develop better understanding of effects of local structural and material variations on overall aeroservoelastic integrity.
- Develop computational tools (validated by experiments) for linking variations in local structural properties to the resulting global linear/nonlinear behavior of the integrated aeroservoelastic system. Cover cases of multiple local variations/ damage.
- Develop statistical tools for the evaluation of structural variability in composite airframes on their reliability.
- Establish a collaborative expertise base at the University of Washington for future response to FAA and industry needs, R&D, training, and education.

Approach:

- Computational capability development will focus on:
  - Quantification of effects on stiffness of key local effects (including damage) in composite structures;
  - Global aeroelastic/aeroservoelastic (ASE) analysis capable of evaluating variations and uncertainty to such local effects;
  - Integrated local/global modeling capability for uncertain composite structures; and
  - Statistical methods with associated computational tools for the evaluation of the aeroelastic reliability of composite airframes as well as combined aeroelastic / damage-tolerance reliability.
- Capabilities for simulation of the effects of control surface and load-carrying structural nonlinearities on aeroelastic and aeroservoelastic behavior of full scale airplanes will be developed and used to study effects of nonlinearity and uncertainty mechanisms and guide maintenance practices.
- Simultaneously, an experimental structural dynamic/aeroelastic testing capability for composite airplane structure models will be developed at the UW, and tests will be planned and conducted to study effects of damage on stiffness of components and models.
- The analytical, numerical, and experimental technology developments will all be done in close collaboration with The Boeing Company.



Accomplishments & Results:

- Rapid, design-oriented simulation tools for linear aeroservoelastic problems of composite aircraft have been developed [1], both in-house as well as based on commercial modeling capabilities such as FEMAP and NASTRAN.

Figures: Passenger airplane fin / rudder NASTRAN model and simple prototype plate-like wing NASTRAN wing model for aeroelastic reliability studies

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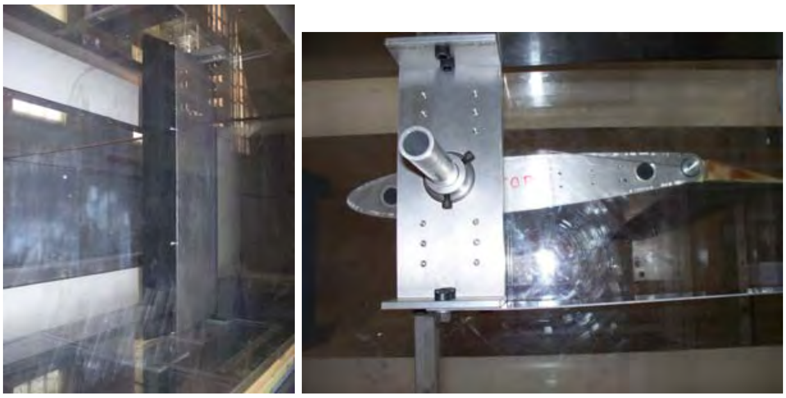
[1] Jackson, T., and Livne, E., "Integrated Aeroservoelastic Design Optimization of Actively-Controlled Strain-Actuated Flight Vehicles", Paper Number AIAA-2005-2170, 46th AIAA / ASME / ASCE / AHS / ASC Structures, Structural Dynamics, and Materials Conference, Austin, TX, April 2005.

- An aeroelastic reliability determination approach has been formulated, implemented in new University of Washington computer tools, and used to study prototype simple airfoil / control surface and fighter wing / aileron systems as well as a representative passenger airplane composite vertical tail / rudder system. [2] [3]
- A general method for coupling linear unsteady aerodynamics and nonlinear composite structural systems has been developed as part of the effort to create efficient methods for the analysis of nonlinear structural effects on the aeroelasticity of damaged airframes or airframes undergoing large deformation (local or global). [4] [5]
- Major progress has been achieved in the structural dynamic and aeroelastic wind tunnel model building / testing capability for composite airframes and airframe parts. First modal tests followed by flutter tests of a prototype fin / rudder system in the University of Washington's 3 x 3 wind tunnel have been completed, and preparation is currently underway to build an aeroelastic wind tunnel model of a damaged (hinge failure) fin / rudder system for flutter tests.



Figures: Fin / rudder aeroelastic wind tunnel model (composite rudder allowing for testing the effects of hinge failure)

[2] Styuart, A., Mor, M., Livne, E., and Lin, K., "Risk Assessment of Aeroelastic Failure Phenomena in Damage Tolerant Composite Structures", AIAA Paper 2007-1981, 48th AIAA/ASME/ASCE/AHS/ASC Structures, Structural Dynamics, and Materials Conference, Honolulu, Hawaii, Apr. 23–26, 2007

[3] Styuart, A., Demasi, L., Livne, E., and Lin, K., "Probabilistic Modeling of Aeroelastic Life Cycle for Risk Evaluation of Composite Structures", AIAA-2008-2300, 49th AIAA/ASME/ASCE/AHS/ASC Structures, Structural Dynamics, and Materials Conference, Schaumburg, IL, Apr. 7–10, 2008.

[4] Demasi, L., and Livne, E., "Dynamic Aeroelasticity of Structurally Nonlinear Configurations Using Linear Modally Reduced Aerodynamic Generalized Forces", accepted for publication, AIAA Journal.

[5] Demasi, L., and Livne, E., "Aeroelastic Coupling of Geometrically Nonlinear Structures and Linear Unsteady Aerodynamics: Two Formulations", AIAA-2008-1758, 49th AIAA/ASME/ASCE/AHS/ASC Structures, Structural Dynamics, and Materials Conference, Schaumburg, IL, Apr. 7–10, 2008.

Production Control Effect on Composite Material Quality and Stability

Principle Investigator(s): Dr. John Tomblin, Executive Director and Allison Crockett, Program Manager, NIAR; Yeow Ng, Associate Director and Beth Clarkson, Statistician, National Center for Advanced Materials Performance

Advanced composites have emerged as the structural materials of choice for many aerospace applications because of their superior specific strength and stiffness properties. First developed for military applications, composites now play a significant role in a wide range of current generation military aerospace systems. There has been a significant increase in the use of composite materials by the large commercial transport aviation industry during the past 25 years, and many advances have been made in general aviation and rotorcraft vehicles where composites are utilized for primary structural applications.

Unlike metallic materials used in structural part manufacturing processes, the material properties of composite structures are manufactured into the structure as part of the fabrication process. Therefore, it is essential that material and process specifications used to produce composite structures contain sufficient information to ensure that critical parameters in the fabrication process are identified to control production and adherence to the engineered part requirements. Due to the wide variety of composite structures now emerging for certification (particularly for general aviation aircraft), control of the materials is rapidly becoming a vital issue with respect to the overall assurance of safety.

This project will interrogate industry sources to determine which issues are understood and which may need further investigation. These issues will be explored by interfacing with appropriate members of aerospace supplier organizations and material user organizations to understand the successes and failures (lessons learned) under the current operating strategy of both aerospace and commercial products.

Objective

1. Identify what fiber, resin, and interface issues are possible which could lead to loss of material control in the product produced for aerospace applications.
2. Identify control strategies for process lines, levels control and their importance in the final product form reliability (correlation between constituent materials, fiber resin mixing and final material form [Prepreg, VARTM]).
3. Review acceptance tests, and specific selected acceptance values (minimum, average and maximum) affect on the detection of safety concerns (vs. economic).
4. Determine how associated safety risks could be mitigated.

Expected Outcomes

This program will develop essential information on the nature of the controls required at various producer levels to assure the continuation of stable and reliable composite raw material for aerospace usage. The intent of this investigation is to determine the level and types of material and process control that would confirm that the property values established during initial evaluation and characterization are not changed over time.

This program will review the delineation in control between aerospace and commercial products. The investigation will identify the differences and similarities in the control process and determine their effect on the reliability of the product produced. It is recognized that variation control related to high volume production is sometimes more restrictive than the controls provided on aerospace products due to economic factors. This may mean that the controls on “commercial” products meet the needs of the aerospace community but are not specifically aimed at end product assurance testing. The suitability of these alternate control strategies and their effect on the ultimate reliability for aerospace applications will be explored.

Wednesday, June 18, 2008

Evaluation of Friction Stir Weld Process and Properties for Aircraft Application

Principal Investigator: Dwight Burford,[1] Wichita State University
Co-Principal Investigator: Christian Widener,[2] Wichita State University

Motivation and Key Issues

Friction stir welding (FSW) and friction stir spot welding (FSSW) are emergent joining technologies that are being developed for a variety of applications throughout the aerospace industry. These applications are typically developed on a case-by-case basis to take advantage of manufacturing and implementation cost savings associated with these relatively new processes, such as part count reduction, improved material buy-to-fly ratios, cycle time reduction, reduced lead-time requirements, lowered environmental impacts, etc. However, because they are relatively new innovations, FSW and FSSW lack sufficient supporting (mature) industry standards and design (allowables) data. Therefore, development and implementation of applications incorporating these technologies require more effort in terms of testing and verification than do developing applications based upon conventional joining technologies (namely installed fasteners). Standards, specifications, and associated design data are needed to more efficiently and consistently design product with FSW and FSSW.

Objective

The primary objective of this work is the incorporation of FSW & FSSW design allowables data into the Metallic Materials Properties Development & Standardization (MMPDS) handbook – formerly the MIL-HDBK-5 handbook. Both butt joint and lap joint configurations are to be addressed. Tabulated data is to be based on performance and procedure specification methodologies that are supported by industry standards expected to be published in the near future (e.g. AWS, ISO, etc.).

Approach

Protocols for incorporating FSW and FSSW joint design data into the MMPDS handbook are being developed and demonstrated through a collaborative program involving two initiatives: 1) process path independence in FSW butt joints and 2) *in situ* fastener qualification of FSSW. A common aim of these two initiatives is to demonstrate that friction stir technologies can be controlled sufficiently through performance specifications to meet MMPDS standards for data incorporation.

Process Path Independence

The first program involves a process *path independence* approach for establishing minimum butt joint properties produced by FSW. The basis of this initiative is the observation that FSW has a sufficiently flexible process window that allows many aluminum alloys to be joined with a variety of weld tool designs. Therefore, in this program it was hypothesized that an unspecified number of tool designs could be used to make equally sound joints with independently developed process windows that may or may not be unique to the weld tool. Statistically based designs of experiments (DOE) were used to confirm the soundness of the joint property results produced by significantly different tools. Minimum joint efficiencies were produced and documented (e.g. static transverse tensile strength). The results of this investigation have demonstrated that a T99 and T90 allowable strength values can be met through a number of different tool designs with proper parameter selection and weld process controls at a single facility. It has also been demonstrated that the results of the testing can be pooled together and compare favorably with the existing data

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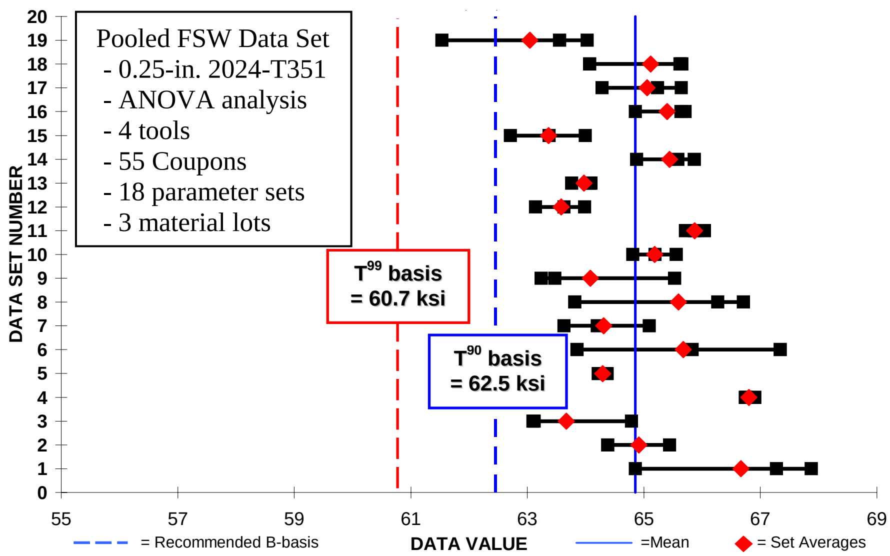
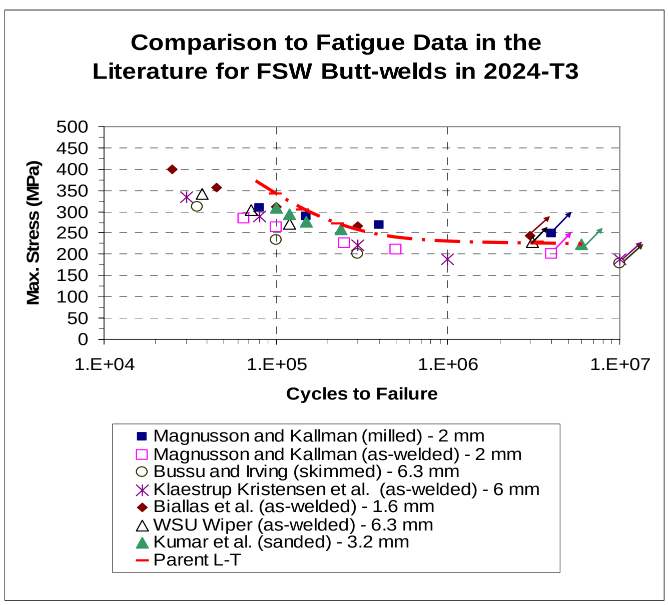
published in the literature. The results therefore suggest that if a sufficient amount of variation is represented by the material included in the allowables calculation, that a reasonable welding allowable will result that can be met by any number of welding facilities with the proper process controls. While fatigue allowables are not included in the handbook, reference curves are often included for designers with data representative of expected performance. Since an understanding of the fatigue performance of FSW butt joints is imperative for its safe implementation in any critical structure, an investigation of the fatigue performance of FSW butt joints compared to parent material was also conducted. In the study, it was found that fatigue results comparable to parent material can be achieved for friction stir welded butt joints. These findings support the continued development of general performance standards for friction stir welding.



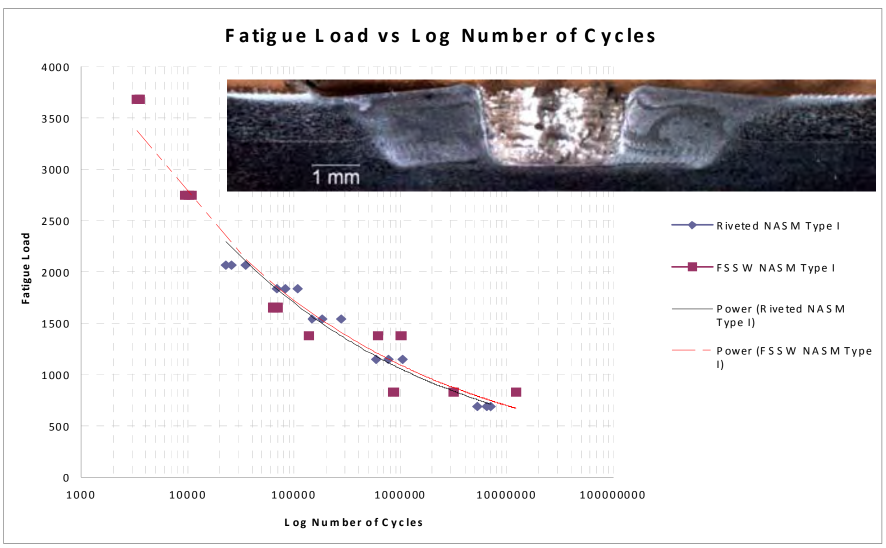


In Situ Fastener Qualification

The second program involves developing a methodology for qualifying different types of friction stir spot welding (FSSW) joints as *in situ* fastener systems. In this program, individual “spots” are treated like conventional installed fasteners. However, In FSSW parent material is used to form an integral mechanical fastener between two or more materials in a lap joint configuration. Advantages of spot welds treated as *in situ* fasteners include:

- Discrete fastener locations separated by parent material (similar to rivets)
- Part count reduction by elimination of filler material, i.e. fastener
- Tailorable spot size and shape provide more latitude than with rivets (diameter constraints, etc.)
- Randomize sequence of installation (to lower distortion)
- Potentially installed via robot vs. gantry to reduce costs and equipment constraints
- Simplified tooling (lower normal and lateral forces)

FSSW joints may be the most straightforward friction stir-related technology to qualify for allowables development because they are most like mechanical fasteners, e.g. discrete. In both static and dynamic tests, properly designed FSSW joints are proving to have similar or superior performance compared to rivets. A number of factors are being tested to confirm their potential benefit, including tensile, fatigue, and crack growth testing. In static tension testing, swept friction stir spot welds are being developed that possess twice the strength of countersunk rivets in 0.040-in. thick sheet material. A micrograph of the resulting microstructure and a sample swept path are shown below. In order to begin understanding the fatigue performance of these joints, an investigation which compared the fatigue life of riveted NASM 1312-21 four spot guided fatigue coupons with riveted coupons in 0.040-in. thick 2024-T3 material was initiated and the results are also shown below. While in some cases there was more scatter observed in the friction stir welded coupons, the fatigue results appear to be highly comparable. Damage tolerance and welding through faying surface sealants for corrosion protection are other areas of study that are being investigated by leveraging funds with an NSF industry and university cooperative research program.



Benefit to Aviation

This program is benefiting the aviation industry in a number of ways, including providing a verified qualification methodology and procedure for incorporating FSW and FSSW data in the MMPDS handbook. The methodology will be based on standard testing and certification procedures related to appropriate process controls and acceptance criteria. Specifically, the expected outcome of this program will be the establishment of organized and certified design data, including S, A, and B basis data sets which could be published in the MMPDS (MIL-HDBK-5) or similar design data handbook. These data sets will be established through process procedure and performance specifications that support design parameters and process development guidelines. This work lays the foundation for a safe and responsible introduction of new technology into commercial aerospace structures. Based on this work, design trade studies can be initiated from the comparative data generated by competing FSW and FSSW with resistance spot weld and rivet data. Ultimately, this program facilitates the delivery of a safe, cost-effective, lean, and green technology to the aerospace industry, a technology that involves low energy use, reduced cycle (manufacturing) time, part count reduction, weight savings, low emissions (no sparks, fumes, noise, or harmful rays) and low ergonomic impact.

Failure of Notched Laminates Under Out-of-Plane Bending

T. C. Kennedy, S. Gonzalez, and B. K. Bay
Oregon State University

Introduction

The design of aircraft structures made of composite materials is heavily influenced by damage tolerance requirements. The problem of predicting failure in notched laminates has been the subject of numerous studies. In general, these investigations have focused on the response of laminates to in-plane tension, compression or shear. In spite of the fact that out-of-plane bending, twisting, or shear can be an important load situation, very little research has been devoted to this topic. The overall goal of this research is to develop analysis techniques that are useful for the design of composite aircraft structure subjected to general out-of-plane loading. The various analytical models that are developed must have the appropriate level of sophistication to meet the designer's needs from simple hand calculations to computer simulation of complex, ply-level response exhibiting multiple failure modes. For this project we limit ourselves to the out-of-plane bending case and focus on some very basic experiments and modeling efforts involving simple structures (center-notched, unstiffened laminates) under pure bending. In partnership with the Boeing Commercial Airplane Company, we are determining the modes of failure of the laminates and evaluating the capability of some currently existing analysis techniques for predicting these failures. Accomplishing our objective requires both experimental and computational efforts. The project can be divided into three main tasks. The first task involves performing 4-point bending tests on laminates with center notches in the form of elongated ellipses. The second and third tasks focus on modeling. For task 2 we are using the general purpose finite element analysis program ABAQUS to construct models of each of the laminates tested under 4-point bending. Bending moment concentration factors are calculated, and their relationship to notch strength will be determined. For task 3 we focus on progressive damage modeling using the progressive damage model for composites contained in the ABAQUS program. Progress on each of these tasks is described below.

Testing of Notched laminates under 4-Point Bending

Very few experimental results for notched laminates subjected to out-of-plane bending are available in the literature. Some experimental results for laminates with ¼-in holes under 4-point bending are available in the Boeing test data base. However, prediction of failure of fuselage panels containing notches two orders of magnitude larger than this are often necessary, and the behavior of laminates with small notches is generally not a good predictor of failure of laminates with large notches. Therefore, we are conducting tests on laminates with larger notches. We are performing 4-point bending tests on laminates with notches in the form of ovaloids, i.e., straight slits with a notch radius of 0.125in. This provides a 0.25-in. gap between the faces of the notch to prevent contact on the compression side. Two notch lengths are being considered: 1in. and 4in. Two laminate thicknesses are being studied: 20 plies and 40 plies. For each thickness, three laminate types are being studied: one with 10% 0-degree plies, one with 30% 0-degree plies, and one with 50% 0-degree plies. In most cases the notch width to specimen width ratio is 5. However, some tests are being conducted with this ratio equal to 10 to evaluate the effect of plate width. A total of 48 tests are planned. At the current time, 4 tests have been conducted. Early observations indicate that failure typically initiates on the compression side of the specimen with delamination of the outermost 0-degree ply and fracture of the plies between this ply and the surface.

Modeling Stress Concentrations in Notched Laminates Under Bending

For design under ultimate load, stress concentration factors are needed for notches around configured structure. Finite element models of plates with a 0.25-in circular hole, a 1-in long ovaloid hole, and a 4-in wide ovaloid hole were constructed using two types of shell elements – one with transverse shear effects and one without. The moment concentration factor at the edge of the notch was calculated for each case. For the 20-ply thick laminates, the moment concentration factors were on average 27 percent higher in the models with transverse shear effects than the models without transverse shear effects. For the 40-ply thick laminates, the moment concentration factors were on average 41 percent higher in the models with transverse shear effects than the models without transverse shear effects. These calculations were repeated using 3-D solid elements rather than shell elements in the finite element model. The moment concentration factors calculated with these elements were generally different from those in the shell element models. The reason for this difference appears to be a free-edge effect. Singularities in the interlaminar normal and shear stresses are known to develop at the ply interface at a free edge. The development of these stresses makes the usual approximations inherent in shell elements invalid at the notch edge. For points a short distance away from the edge, the 3-D element models and shell element models produced very similar results. In determining the far field strain allowables for composite aircraft structures, it is probably more useful to deal with strain rather than internal bending moment. Therefore, the previous calculations were repeated for each laminate, but this time a strain concentration factor was calculated based on the maximum strain in the outermost 0-degree ply. An examination of the strain distribution through the thickness at the edge of the notch in the 3-D solid model indicated a linear distribution (similar to the shell element model) except for pronounced bulges in strain in the 0-degree plies. These bulges in strain led to significantly higher strain concentration factors than was predicted by the shell element models. On average the strain concentration factor predicted by the 3-D solid model was 36% higher than that predicted by the shell model with transverse shear effects. Again, the reason for this difference appears to be a free-edge effect. For points a short distance away from the edge, the 3-D element models and shell element models produced very similar results.

Modeling Progressive Damage in Notched Laminates Under Bending

We would like to be able to simulate the propagation of a notch in a composite laminate under out-of-plane bending. In a composite material, a zone of damage of considerable influence is known to develop in advance of the notch. This is the result of a combination of failure modes including fiber breaking, matrix cracking, etc. Consequently, the usual fracture mechanics procedures that have worked successfully in metal structures do not work well for composites. The simulation of damage progression in a composite is best done with theories that incorporate principles from the field of damage mechanics. In the case of bending, there is a non-uniform strain through the thickness of the laminate. A theory that treats damage progression at the ply level is needed for this case. The ABAQUS finite element analysis program has a damage model specifically for this situation. For each ply a damage initiation criterion based on Hashin's theory is used which considers four different damage initiation mechanisms: fiber tension, fiber compression, matrix tension, and matrix compression. The progression of damage manifests itself in the form of progressive degradation of material stiffness according to an evolution law. In addition to the degradation of the ply properties, delamination is modeled by placing cohesive elements between each ply (which is typically modeled with continuum shell elements) and allowing the cohesive elements to fail when a delamination criterion is met. Material properties for the plies and cohesive elements were provided by Boeing based on coupon tests. Models have been constructed for laminates with 1-in long notches. Preliminary calculations using simplified models indicate that these models typically over-estimate the failure loads by approximately 20 percent.

VARTM Variability and Substantiation

Principal Investigator(s): Dirk Heider, and John W. Gillespie, Jr.

Research that will support the implementation of Vacuum-Assisted Resin Transfer Molding (VARTM) for aerospace applications is being performed at the University of Delaware Center for Composite Materials (UD-CCM), a member of the Center for Composites and Advanced Materials (CECAM), and part of the FAA's Joint Advanced Materials and Structures (JAMS) Center of Excellence. The long-term objectives of the research are to improve VARTM repeatability equivalent to autoclave processing with specific properties (property/weight) that are close to autoclave-processed part levels at a lower cost.

The current study has developed the fundamental understanding to predict dimensional tolerances, created unique characterization equipment to better understand the sources of variability and has investigated various VARTM variations used in aerospace. In addition, a repeatable and automated VARTM workcell has been implemented allowing elevated temperature infusion of typical toughened epoxies currently used in aerospace. The cell has been transitioned to an industrial partner for production of aerospace components and the cell is used to initiate a mechanical property database for VARTM processed components.

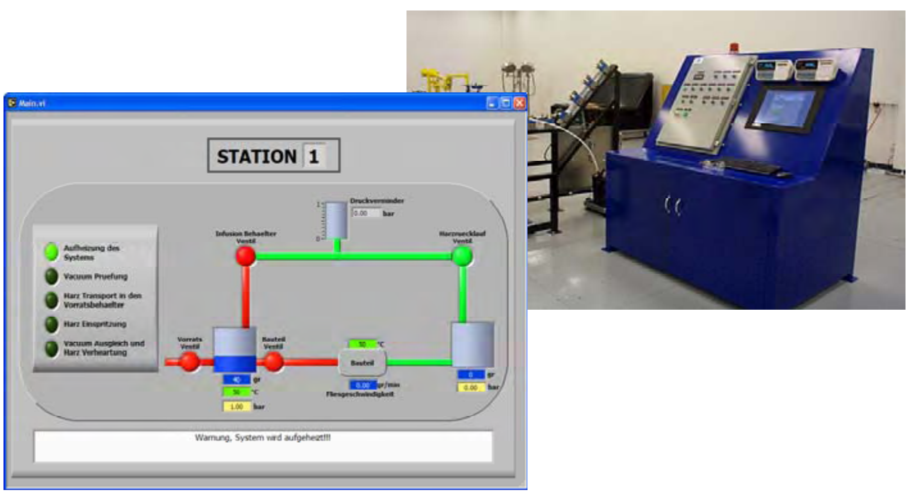


Figure 1: An elevated VARTM station has been developed automating the infusion of aerospace toughened resin systems

The research has developed process models to allow prediction of flow, compaction and pressure behavior during typical VARTM processing. The process models have been validated for the Controlled Atmospheric Pressure Resin Infusion (CAPRI) process, patented by Boeing Co. and the Vacuum-Assisted Process (VAP) patented by EADS. The simulation can be used to predict the important process parameters and allows optimization of the process setup. Here, a new apparatus has been developed capable of accurately measuring the transverse permeability as a function of compaction and debulking cycles using both gaseous and liquid flow. The experimental cell provides insight into the variability of the incoming material and provides the needed understanding of the material changes during debulking to fully understand the CAPRI process.

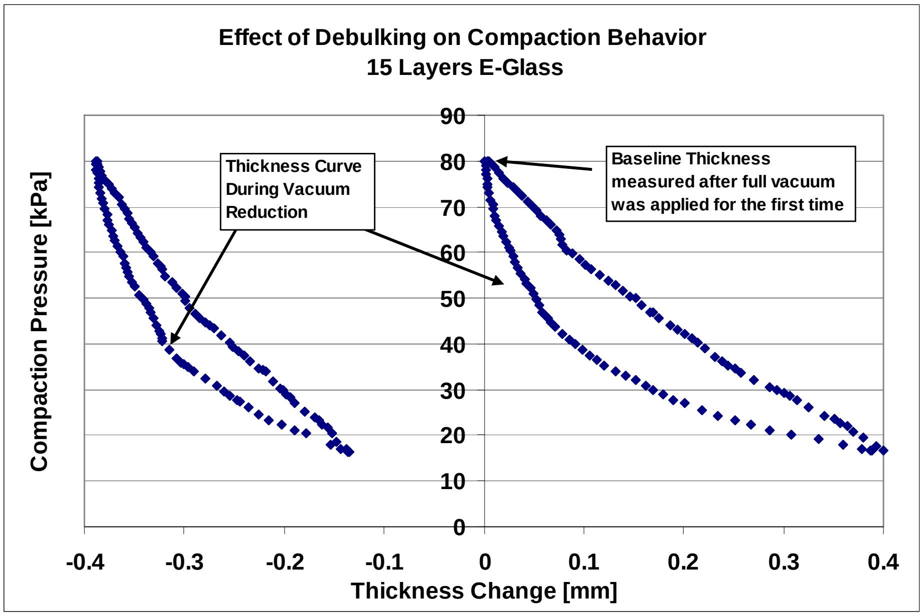


Figure 2: Debulking of dry reinforcement can increase fiber volume fraction but also influences the infusion behavior

The VAP process provides an alternative approach to reduce variability. The air-permeable, resin-proof membrane allows application of continuous vacuum compaction on the complete surface even during infusion reducing the thickness gradient. The membrane also enables a more robust VARTM process that minimizes/eliminates the potential for dry spot formation and lowers void content due to continuous degassing of the resin during impregnation. Here, volatiles generated during processing can escape through the membrane layer and reduce the void content well below 1% for typical epoxy resin systems. To fully control the membrane-based process and extend its use to a wider range of resins, a fundamental understanding of compatibility issues is currently being developed. Industry is experiencing membrane failure for parts with complex draping requirements. To address this issue, basic characterization and a study of stretching of the membrane is under current study. Ultimately, the goal is to optimize both the membrane material as well as the support layer for both optimum permeability behavior and mechanical robustness.

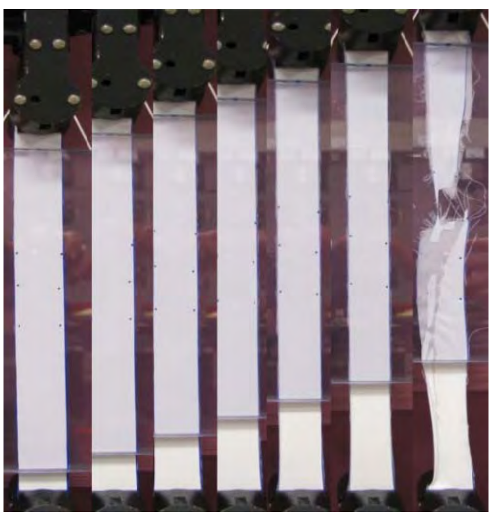


Figure 3: Stretching Behavior of Membrane and Ultimate Failure

Progressive stretching does not show visible damage to the membrane until the support tears. The maximum pore size is responsible for the pressure driven flow through the membrane. Characterization of the membrane with a capillary flow porometer has shown that the maximum pore size does not change dramatically until approximately 20% strain where the support begins to fail.

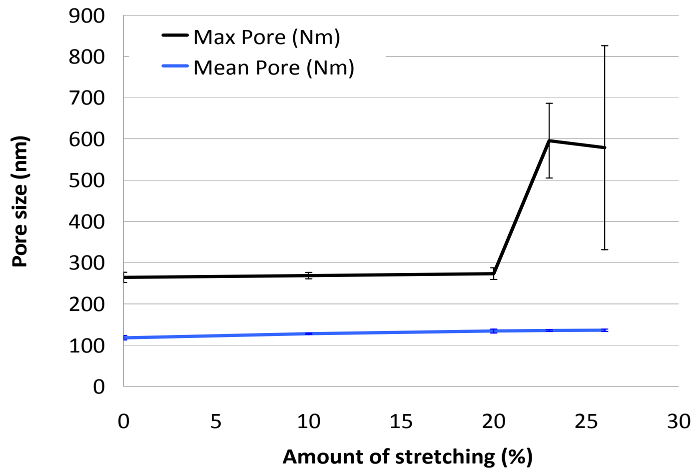


Figure 4: Pore Size v. % Stretch

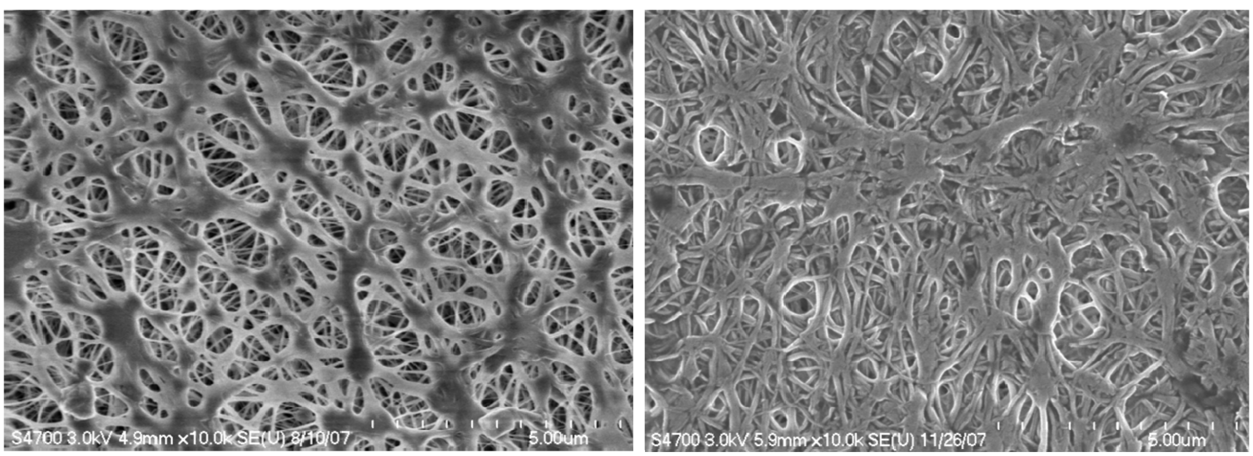


Figure 5: SEM Images of unstretched (left) and stretched (23% Strain - right) membranes

The microstructure responsible for the sudden pore size change around 20% strain is clearly visible in the SEM images. Fibril tears are evident and are believed to be responsible for the drastic increase in the max pore. The membrane draping issues are continuing to be studied in order to isolate the permeability issues caused by stretch.

In summary, the VARTM process is poised to penetrate the aerospace market. New process developments and a better fundamental understanding of the process allow part fabrication with improved dimensional tolerances and good mechanical properties at reduced total fabrication cost. Typical fiber volume fraction above 55% with 1% void content or lower can be repeatable, achieving near autoclave properties. In addition, the material suppliers have commercialized new toughened resin systems and non-crimp fabric materials. Automation is also available to reduce the expert's input currently required to fabricate components. Still, continued research is on-going to allow for a better understanding of the infusion process in particular when the system is scaled up to large and complex geometry components.

Development and Evaluation of Fracture Mechanics Test Methods for Sandwich Composites

Principal Investigator: Daniel O. Adams, University of Utah

Although the development of test methods for fracture mechanics of monolithic composite laminates has reached a high level of maturity in recent years, relatively little attention has been given to the development of fracture mechanics test methods for sandwich composites. Further, a majority of the efforts to date has focused on determining the fracture toughness of a particular sandwich material or the effects of specific environmental conditions. As a result, there is no consensus on a suitable test configuration or specimen geometry for either Mode I or Mode II fracture toughness testing. Thus, the objective of this research investigation is to develop test methods for the determination of both the Mode I and Mode II fracture toughness of sandwich composite materials that are suitable for ASTM standardization.

Through a series of coordinated tasks, both Mode I and Mode II fracture toughness test methods are to be developed in parallel. The initial tasks have focused on identifying and assessing candidate test methodologies. Additionally, the various materials and geometries currently in use for structural sandwich composites have been identified. Based on this initial review, several candidate Mode I and Mode II test configurations were investigated using finite element analysis. These analyses were used to evaluate whether a suitable fracture mode mixity (i.e. Mode I vs. Mode II) is present for a particular test configuration. Additionally finite element simulations were used to assess whether undesirable failures may occur within the sandwich specimen during testing. The objective of these finite element simulations was to provide an in-depth assessment of the candidate test configurations such that the potential Mode I and Mode II test configurations may be selected for initial mechanical testing. Additionally, these simulations were used to determine optimal geometries for prototype test fixturing and test specimens.

Once the initial test configurations were identified from finite element analyses, two sandwich composite configurations were selected for initial testing; woven carbon-epoxy facesheets/polyurethane foam core and carbon-epoxy facesheets/Nomex honeycomb core. All composite sandwich panels were fabricated at the University of Utah. Initial testing was performed using the selected Mode I and Mode II test configurations. Emphasis was placed on producing stable, self-similar delamination growth at the facesheet/core interface without producing secondary failures at other locations in the specimen. Details of the selected test configurations and initial test results are presented separately for the Mode I and Mode II tests in the following sections.

Mode I Test Methods

A total of five Mode I test configurations were selected for investigation. These configurations were based on three basic loading configurations: the double cantilever beam (DCB), the single cantilever beam (SCB), and the three point bend (TPB) configurations. Both experimental and finite element analysis was performed for each test configuration and sandwich composite combination. Results from finite element analysis showed that all five test configurations were Mode I dominant. The SCB beam displayed the lowest percentage of Mode II energy release rate, whereas the modified DCB displayed the greatest percentage. Results from mechanical testing showed crack growth away from the interface and into the core using both DCB configurations and one SCB configuration, an undesirable result. Based on the results of analysis and testing, the SCB test configuration shown in Figure 1 was selected as the best suited Mode I test configuration for sandwich composites.

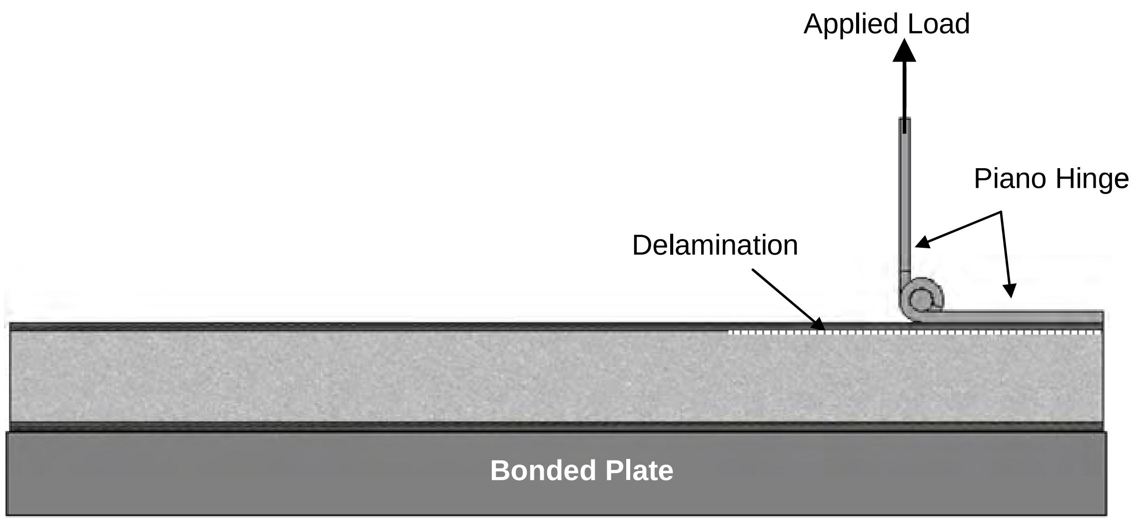


Figure 1. Single cantilever beam (SCB) test configuration selected as best suited Mode I test configuration.

Mode II Test Methods

A literature review of previously performed work along with modified and newly created tests led to the identification of ten candidate Mode II test configurations methods for evaluation. Initial evaluation was performed using finite element analysis to evaluate mode mixity and to ensure that Mode II dominated crack growth occurred. The degree of crack opening produced during loading was also examined to ensure that frictional forces are not created at the interface which could produce artificially high energy release rate values. Based on results obtained from finite element analysis, seven test configurations were identified for follow-on mechanical testing. Mechanical testing of the seven configurations led to only two which produced the stable crack growth at the facesheet/core interface: the Mixed-Mode Bending (MMB) test and a hinged Cracked Sandwich Beam (CSB) test. In follow-on testing with aluminum honeycomb cores, however, the MMB test produced core crushing whereas the hinged CSB test produced stable interface crack growth. Thus, the hinged CSB test, shown in Figure 2, has been selected as the best suited Mode II test configuration for sandwich composites.

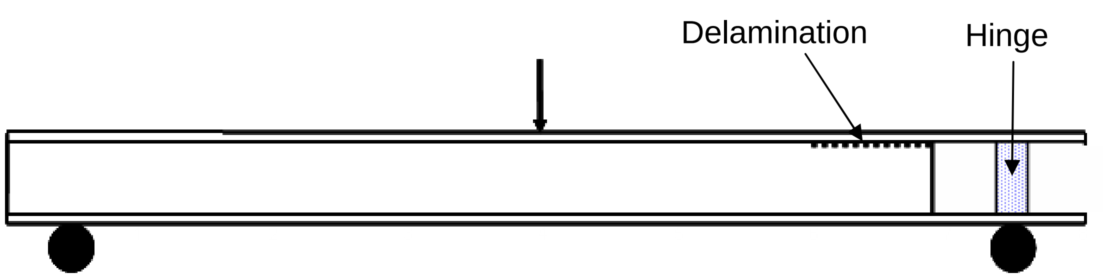


Figure 2. Hinged Cracked Sandwich Beam (CSB) test configuration selected as best suited Mode II test configuration.

Planned Research

Following the preliminary selection of both Mode I and Mode II test configurations, further finite element analysis and mechanical testing will be performed to determine the range of materials and sandwich geometries from which stable, self-similar delamination propagation is obtainable. Within this second phase, two distinct rounds of testing are anticipated such that results obtained in the first round may be used to select additional sandwich configurations or specimen geometries for follow-on testing. Additionally, it is anticipated that relatively minor changes to either the Mode I or Mode II prototype test fixtures may be made following the first round of testing. The goal of this second phase of testing and analysis is to arrive at a final test fixture design, specimen configuration, data analysis methodology, and a preliminary range of acceptable sandwich materials and geometries for both Mode I and Mode II fracture toughness testing. Based on these results, a final test configuration and analysis methodology will be selected for both Mode I and Mode II testing. Additionally, acceptable ranges of sandwich materials (facesheet and core), material parameters (facesheet fiber orientations, core densities), and sandwich geometries (facesheet and core thicknesses) will be proposed for inclusion in both the Mode I and Mode II test methods. These efforts will lead to the third phase of the research program, which will focus on the development of draft ASTM standards for determining both the Mode I and Mode II fracture toughness of sandwich composites.

Effects of Repair Procedures Applied to Composite Airframe Structures

Principal Investigators: Dr. John Tomblin, Executive Director, NIAR and
Sam Bloomfield Distinguished Professor of Aerospace Engineering
Lamia Salah, NIAR, Senior Research Engineer

Problem Statement

With the increasing use of composite materials in aircraft structural components, it has become essential to answer not only the fundamental questions related to the proper repair methods/systems to restore the aircraft part structural integrity but also the question of how long the repair will last under the specified design conditions and what are the most critical factors affecting the static performance and the long-term durability of the repair.

In other terms, it is necessary to assess how long the repair will last under the specified design conditions and what are the most critical factors affecting the static and fatigue performance of the repair. The lack of fatigue data to assess the durability of repairs, added to the lack of confidence in bonded repairs especially when dealing with large damaged areas, has led to the use of fasteners to reinforce the adhesively bonded areas in some cases. The ultimate goal of a bonded repair is to achieve a good level of confidence in bond strength as well as the ability to avoid long-time service failures of these joints.

Furthermore, the current NDI methods fail to assess bond quality (i.e., fail to detect a weak bond due to poor surface preparation, pre-bond moisture, under or over-cure, surface contamination, etc.). As a consequence, a bad repair is not detected until it actually disbonds leading to a possible catastrophic failure of the repaired part. It is therefore essential to quantify the fatigue life “knockdown” of these weak joints.

Objectives:

The purpose of this research program is to assess the effects of different variables on the static and fatigue performance of scarf repairs applied to composite laminate and sandwich structures and the long time durability of these repairs especially when a faulty process has been implemented and was not detected by NDI. The main research program has been divided into three tasks.

The first task consists of investigating the effects of different variables on the strength performance of repairs applied to moderately thick solid laminates and sandwich structures. Variables considered include different substrate stiffnesses, lap lengths, laminate thicknesses and temperatures. Loading modes considered are tension and bending. Coupons are being tested for static and fatigue properties to establish baseline performance. Constant amplitude fatigue testing is being conducted on the coupons and residual strength is measured after 165,000 fatigue cycles.

The second task consists of evaluating the durability of “poor” bonded repairs that passed NDI (undetected weak repairs). Factors such as poor surface preparation, pre-bond moisture, contamination, impact and improper cure (over-cure/under-cure) are being investigated. The goal is to evaluate the static and/or fatigue life knock-down when one of the repair steps was not properly implemented or when the structure has been damaged after repair. This task will assess the weak repair fatigue capability on both sandwich and laminate structures.

The third task consists of a validation of safety standards required for composite repair and inspection technicians as related to composite repairs. The commercial aircraft composite repair committee

(CACRC) developed industry standards to be followed by composite repair technicians. A previous study has indicated the quality of training and experience of repair technicians may have a much larger role in the technician's successful development of a repair as shown in figure 1. This study has indicated the quality and reliability of a composite repair is much more directly linked to the skills/knowledge of the repair technician than was previously believed and specified in the CACRC standard.

The main goal of this task will be used to generate a report evaluating the technical value of the CACRC standards for technicians performing aircraft composite material repair and inspection and to provide recommendations pertaining to the importance of the various steps in a repair process.

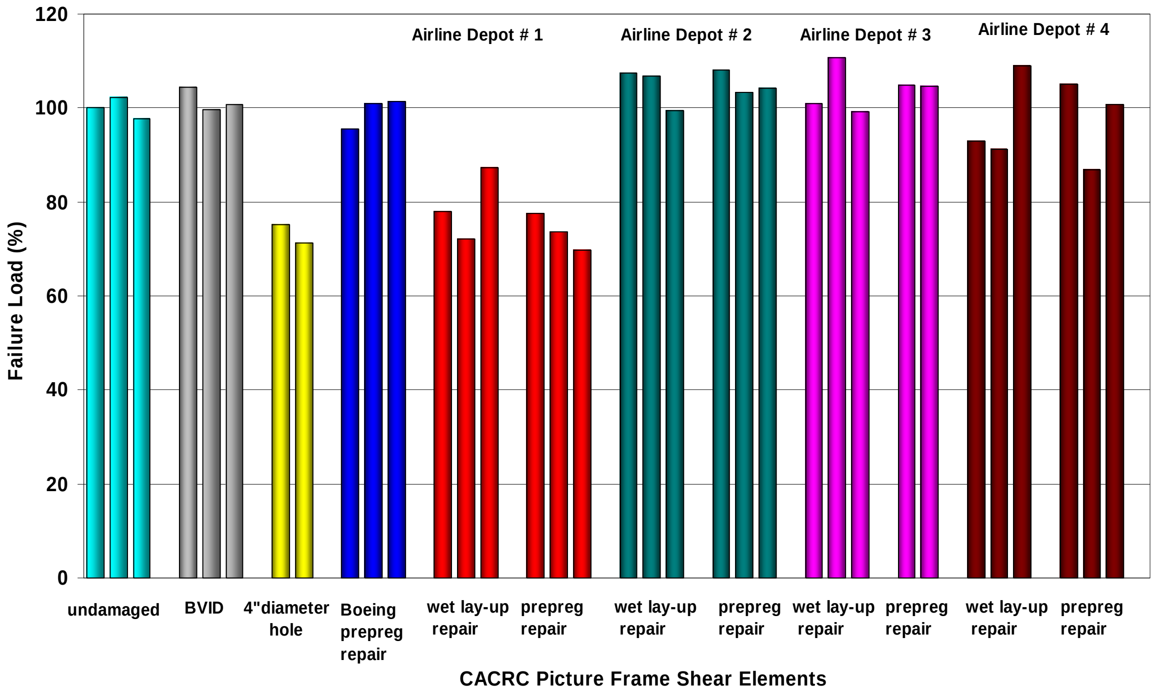


Figure 1. Mechanical Test Results of Picture Frame Shear Elements Repaired by Different Operators

Methodology

The proposed methodology consists of generating baseline repair data (static and fatigue) for both laminate/ sandwich configurations using OEM/ Factory but also field repairs. The strength and durability of poorly bonded and/or contaminated repairs that passed NDI (Laminate/Sandwich) is also evaluated. The damage tolerance of repairs subjected to BVID inflicted at three different locations on the repair is assessed through static and fatigue testing. Finally, based on the process parameter and damage tolerance investigations, recommendations are provided pertaining to process improvement to ensure repair bond repeatability and structural integrity.

The data generated will be used to understand the effects of different variables on the performance of bonded repairs and the basic degradation mechanisms that bonded repairs undergo under sustained long-term mechanical loads especially when the repair is damaged or contaminated. The ultimate goal will be to assess the level of criticality of each step in the bonded repair process and to provide recommendations pertaining to bonded repair process repeatability to ensure the structural integrity of the repaired joint.

Expected Outcomes

The repair data for OEM/ factory materials, generated as part of this program, can be used to demonstrate acceptability of alternate materials to use for repair when the parent material is not available or cannot be used for repair.

Furthermore, data generated will help correlate contamination and process parameter deviation to the performance of bonded repairs. In addition, information on repair damage tolerance is also provided, which is a function of damage area and location.

Data generated by this program will also help identify the most critical steps in repair implementation and the effects of process deviations on the static and fatigue performance of a bonded repair. This information can be used to re-evaluate the existing CACRC technician standards leading to a new FAA composite repairman qualification. Finally, by implementing more rigorous processes in bonded repairs and minimizing repair failures attributed to process deviations will help establish confidence in bonded repairs.

Improving Adhesive Bonding of Composites through Surface Characterization

Principal Investigator: Brian Flinn, University of Washington

Problem Statement

It is becoming common knowledge that proper surface preparation is the key to successful adhesive bonding of composites. However there is no robust method available to test if a composite surface has been properly prepared for adhesive bonding. This is due in part to a lack of fundamental understanding of the surface characteristics that produce a strong, durable bond. This research aims to contribute to the fundamental understanding of the surface characteristics that are required for safe and reliable adhesive bonding of composites.

Background

Adequate surface preparation of composite parts for adhesive bonding by use of peel plies is highly desirable, yet it remains necessary to test each combination of composite resin system, peel ply, and adhesive for successful adhesion. In practice, peel ply treatment has been found to be an effective and efficient for the manufacturing of some primary bonded composite structures. Peel ply is a woven synthetic fabric added as the last layer in the lay-up and cured to the composite part to be bonded. After cure, the peel ply is then removed from the surface immediately before bonding. The characteristics of a surface created by peel ply removal are directly influenced by how the peel ply separates from the laminate and any mixing or interactions that take place between peel ply and substrate resin systems. Additionally, peel plies that may work for one resin or adhesive system may be ineffective with others.

The possible modes of peel ply removal can be seen in Figure 1, and are either 1) the fracture of the epoxy resin between the peel ply and the underlying carbon fibers, due to strong bonding of the peel ply to the composite matrix during cure (dark blue) or 2) interfacial fracture between the peel ply fabric fibers and the epoxy matrix, (pink). 3) the peel ply fibers may fracture and leave material on the composite surface (green) or 4) there may be interlaminar failure in the composite (turquoise). In the first mode, a fresh epoxy surface is created that should be chemically active and easily bonded. Though ideal, this is rarely the only mode of fracture present. In the second mode, the chemistry of the surface created may be affected by the nature of the peel ply material surface. Peel ply coatings or fiber surface treatments may be transferred to the surface to be bonded and affect the future bond. [3] The third mode may occur if the bond between the peel ply and epoxy is stronger than the peel ply fibers, and the fourth if the interlaminar strength of the laminate is low or the peel ply is removed incorrectly. Typically, a combination of modes 1 and 2 is observed.

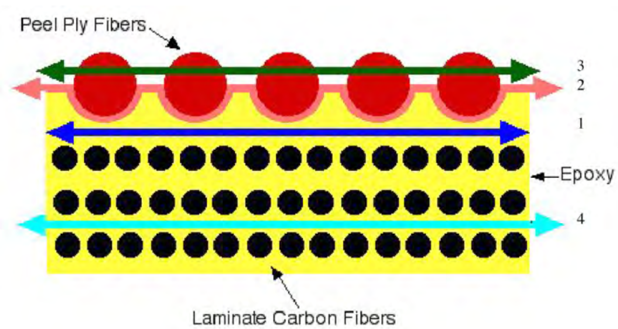


Figure 1: Possible fracture paths during removal of peel ply

Contamination of surfaces to be bonded is also of concern. The use of gloves is a standard practice and serves to protect the human body from chemicals AND to keep contamination (body oils, etc) away from bonding surfaces. However the proper type of gloves must be worn, or the gloves themselves may contaminate the composite surface.

Approach to solve the problem

Our method to solve this problem is to relate basic surface characteristics such as surface energy, surface chemistry and surface topography to the surface preparation technique, material system (adhesive and substrate) and bond quality.

One way of simultaneously measuring the effects of several of these variables on practical adhesion is by measuring the surface energies of the adhesive and adherends. A simplified view of the surface energy of a solid is that it is, essentially, a measure of its attractive force on other materials, due to unsatisfied chemical bonds, hydrogen bonds, and Van der Waals forces at its surface. In the case of liquids, this is the surface tension and is the reason that liquids tend to "bead up" into spherical shapes; they are minimizing their energy level by minimizing the amount of exposed surface area. Alternatively, the surface energy of a solid can be defined as being equal to the surface tension of a liquid that will just wet the solid, with a contact angle $\theta = 0$ [2]. The surface energy has units of force per distance, and the underlying relationship $\gamma_{sv} = \gamma_{sl} + \gamma_{lv}\cos\theta$, known as Young's equation.[3] In theory, a high-energy liquid should bead up on a low-energy solid surface, while a low-energy liquid should "wet out," or form a thin layer, on a high-energy surface. Wetting out of the adhesive on adherends is generally recognized as a necessary, although not sufficient, requirement for creating a strong adhesive bond [4]. The Owens and Wendt model [4] has two parameters for surface energy, giving it a "dispersive" and "polar" component, allowing a better prediction of wetting relationships. This relationship allows a two-dimensional "wetting envelope" to be constructed for any solid. The wetting envelope is the line on a plot of dispersive energy vs. polar energy that represents the breakeven energetic point for a liquid to wet out on that surface. Such a diagram should be useful, because in theory, adhesives that fall outside of the envelope for a given adherend should not function well due to lack of wetting [5]. A more recent model of surface energy, known as the Lewis acid- base model may also hold promise for understanding the role of surface energy in adhesive bonding.

Given the wide range of variables that affect bond quality, successful approaches to the use of adhesives need to be based on both theoretical considerations and empirical results; that is, it is possible to predict how some adhesives and adherends will interact, but success cannot be guaranteed until it has been demonstrated in practice, with attention paid to all of the procedural details. One test for this purpose is the double cantilever beam ("DCB") test, commonly used to measure the fracture energy (strain energy release rate, G_{IC}) of adhesive bond described in ASTM D5528-01 [6]. The DCB test measures both mode I fracture energy (G_{IC}) and gives a mode of failure, such as interlaminar or adhesion. Unfortunately, the DCB test is expensive in terms of time and required machining accuracy, giving a motivation to find a quicker and less expensive method to evaluate a large number of material combinations. The Rapid Adhesion Test ("RAT") was developed as a simple, fast Mode I screening process for rejecting component combinations that fail in adhesion, taking advantage of the qualitative nature of this determination [7]. The RAT takes advantage of the fact that there is a strong correlation between mode of failure and G_{IC} , so merely determining the mode of failure gives very important information regarding bond quality. An undesirable failure mode in a specimen – namely, a failure in adhesion, with its much lower G_{IC} – does not require a quantitative measure of performance, because that combination would be rejected. It has been shown that RAT results correlate to DCB results regarding the mode of failure [7]. Furthermore, failure in adhesion, as opposed to interlaminar or cohesive failure, is usually unacceptable for any application. The recently developed Instrument Rapid Adhesion Test or *i*-RAT is a peel test patterned off the climbing drum peel test; and is intended to give a fast, easy way to quantitatively screen new adhered – surface preparation – adhesive combinations by utilize a peel torque value in combination with failure mode.

Discussion

The important question to be answered is: Can the surface characteristics presented above predict bond quality? Theoretically wetting is a necessary but perhaps not sufficient condition for the formation of strong bonds. Wetting is controlled by the surface energies of the surface, and the adhesive in a given environment. If the surface energies of the adhesive fall within the wettability envelope of the surface the adhesive should wet the surface. However, this does not necessarily mean that strong bonds have been formed. This point has been illustrated in this work by the fact that poor bonds were formed even though the adhesives fell with in the wettability envelopes of the prepared surfaces. DMSO has been suggested as a suitable fluid to replace water in a “water break” test to check for contamination, since its polar and dispersive surface energies are close to most epoxies. [5]. DMSO wet out on all of the substrate surfaces in this study, even when poor bonds were formed. Caution is advised in using this approach to determine the suitability of surfaces for bonding. Poor wetting of DMSO would certainly be a red flag and likely indicated a surface is not suitable for bonding, however the converse is not necessarily true. More detailed surface characterization using SEM and XPS provided more information regarding the condition of the substrate surface to be bonded. When remnants of polyester and nylon peel ply fibers on the surface were revealed by high magnification inspection and XPS spectra with high levels of oxygen or nitrogen respectively, adhesion or mixed adhesion/cohesive failure were seen in all but one system in these cases.

Conclusions

1) Peel ply surface preparation that works with one peel ply-prepreg-adhesive system will not necessarily work if any one of the three is changed.
2) Peel ply fiber and resin interactions during cure, and the resulting matrix properties near the peel ply fibers play an important role in the surface created when the peel ply is removed.
3) The fracture path during removal of peel ply has a strong effect on the quality of the bond. Peel ply remnants on the substrate surface were shown to be detrimental to bond quality.
4) Contact angle measurements, surface energy and wettabilty envelopes are useful in understanding the nature of the surface, but have not be shown to predict the quality of bonds
5) Detailed surface analysis using SEM and XPS provide information need to determine the nature of surfaces created by peel ply removal.

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Quantifying Methods for the Evaluation of Carbon Based Composite Surfaces for Subsequent Adhesive Bonding

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"Combining the results of DSC and DMA Experiments to determine OEM Parameters Governing the Cure of T700-2510PW Prepreg"

Introduction

T700-2510PW prepreg is a fast curing candidate resin for repair applications. The techniques of Differential Scanning Calorimetry (DSC) and Dynamic Mechanical Analysis (DMA) have been combined to shed some light on the cure profile of this material. We begin by examining the cure kinetics of the material under ramped conditions by DSC. We continue by examining the cure kinetics of the material under isothermal conditions by DSC. We next combine the two to determine cure (and rate of cure) under ramp and soak conditions. We end by examining the storage and lass shear modulus by DMA as a function of position in a ramp and soak cure cycle.

Due to space limitations in this summary, only the programmed DSC results will be discussed here.

Cure of T700-2510PW Prepreg under Ramped conditions by DSC

Extent of cure

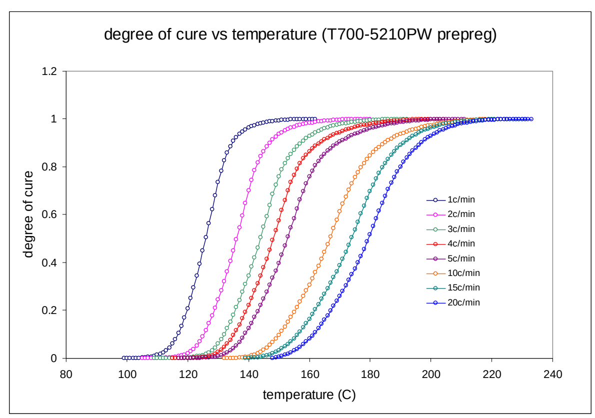
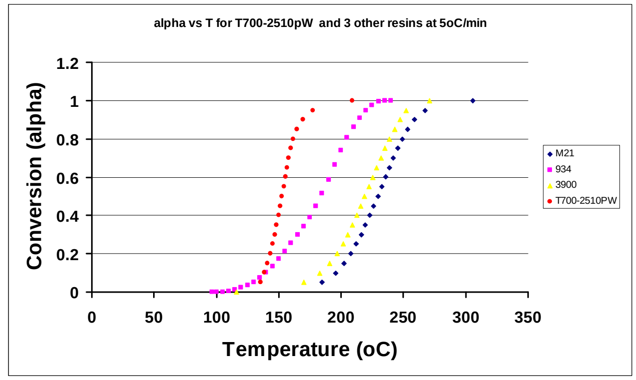


Figure 1. Degree of cure (α) versus temperature (^{o}C) for T700-2510PW prepreg cured under programmed conditions

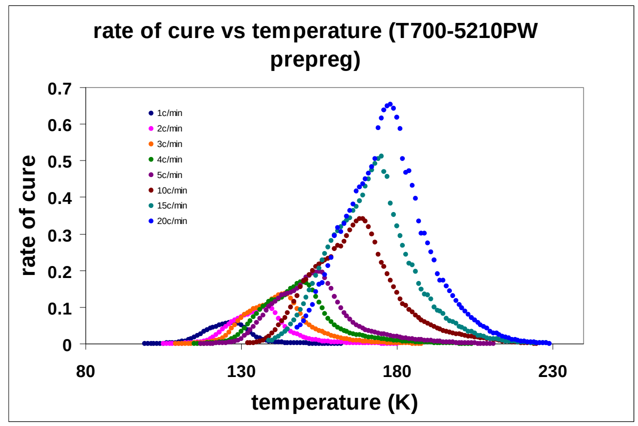
T700-2510PW prepreg was cured in the DSC apparatus under programmed conditions. Data analysis has been detailed in earlier reports. Conversion versus temperature plots are reproduced in **Figure 1**. The conversion versus temperature curve, at a heating rate of 5^{o}C/min, for 2510PW resin, is compared with 934 resin (in T300/934 prepreg), 3900 resin (in TORAYCA(IM350F) prepreg), and the resin in M21 Unitape in **Figure 2**. It can be seen that 2510PW resin is even more reactive than the BF_3 catalysed resin in T300/934 prepreg, a very reactive resin.

Figure 2. Degree of cure (α) versus temperature (^{o}C) for resin in T700-2510PW prepreg and resin in three other common high performance aviation prepregs, all cured under programmed conditions at 5^{o}C/min.



Cure rate

Figure 3. Rate of cure (dα/dt) versus temperature (T) for T700-2510PW prepreg

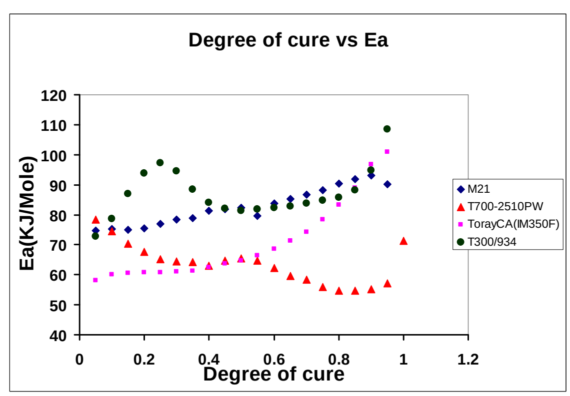


The determination of cure rate as a function of temperature (^{o}C) over a range of heating rates has been discussed in earlier communications. Plots or rate of cure (dα/dt) versus temperature (T) for T700-2510PW prepreg are reproduced in **Figure 3**. As expected, the rate maxima for the cure reaction shifts to higher temperatures at higher heating rates. Values of (dα/dt) as a function of α are determined from this graph in order to construct plots of ln(dα/dt) vs 1/T for the activation energy plots for programmed cure (below)

Activation Energies (Ea) under Ramped Conditions

Activation energies for the cure of resin in T700-2510PW prepreg were determined under programmed heating conditions as a function of conversion using a modified Flynn Wall approach. A plot of activation energy for the cure of T700-2510PW prepreg is plotted as a function of conversion in **Figure 4**. It can be seen that the activation energy of cure of resin in T700-2510PW prepreg is lower than that measured for cure of 3900 resin in TORAYCA(IM350F) prepreg and resin in M21 unitape throughout most of the cure reaction. (Individual plots for activation energy for T700-2510PW prepreg are reproduced in **Appendix I**). The resin in T700-2510PW prepreg does not express a low degree of cure maximum in the Ea plot as does the BF_3 catalysed 934 resin in T300/934 prepreg; although the activation energy for cure of T700-2510PW is high for low conversions and lower for higher conversions, a result in conflict with the progression of Ea versus cure for uncatalysed resin.

Figure 4. Ea versus conversion (α) for resin in T700-2510PW prepreg, M21 pultrusion unitape, for resin in TORAYCA(IM350F) prepreg, and for resin in T300/934 prepreg



Cure of T700-2510PW prepreg under isothermal conditions by DSC

T700-2510PW prepreg was cured under isothermal conditions in the DSC apparatus. A plot of conversion versus time for cure, over a range of isothermal temperature, is reproduced in **Figure 5**. A plot of rate (dα/dt) versus time, for cure over a range of isothermal temperatures, is reproduced in **Figure 6**. From **Figure 5**, it can be seen that conversion for cure over isothermal temperatures from 155-180^{o}C proceeds to completion. Cure is rapid so that complete conversion, even at the lowest temperature is achieved in under 15 min. Examination of the rate curves yields more insight into the cure reaction. Most high performance epoxy resin prepreg mixtures are catalysed by products of reaction – ie, polar hydroxyl groups, etc. When cured under isothermal conditions, these resin systems begin cure at some fixed rate that rapidly rises to a maximum then drops off at higher conversions. This can be seen in the rate curve for reaction at 180^{o}C. (**Figure 6**). However, at lower temperatures, a pair of rate maxima is revealed. This is strong circumstantial evidence for the presence of an external catalyst in the prepreg that catalyses early cure of the resin.

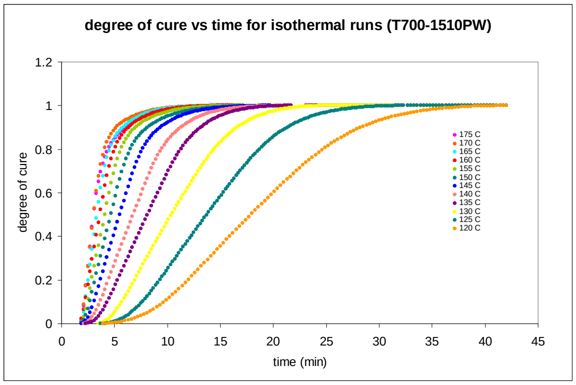


Figure 5. Plot of conversion (α) versus time for cure, over a range of isothermal temperature of T700-2510PW prepreg

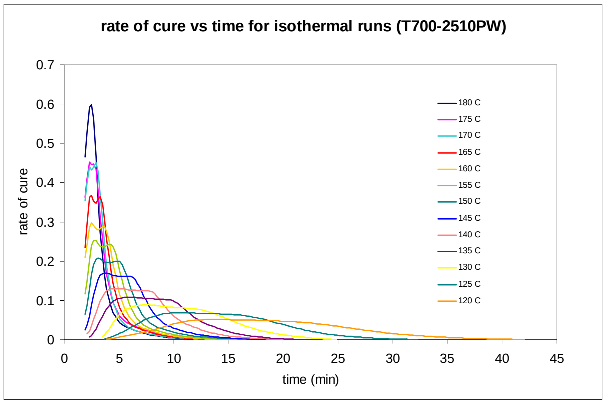


Figure 6. Plot of rate (dα/dt) versus time for cure, over a range of isothermal temperature of T700-2510PW prepreg

Identification and Validation of Analytical Chemistry Methods for Detecting Composite Surface Contamination and Moisture

Principal Investigators: Dwayne McDaniel and Xiangyang Zhou, Florida International University/University of Miami

Problem Statement

The objectives of the proposed research are: 1) identify surface quality assurance methods that are currently being used by aircraft manufacturers and repair service providers and determine whether the current quality assurance tests including the wedge test are sufficient to ensure the contaminated peel plies are detected and not used, and 2) to identify and validate definitive analytical chemistry methods to provide sufficient in-field quality assurance. The benefits of this research to the aviation industry are as follows: 1) Better understanding of the pre-bond surface preparation methods; 2) Better understanding of bond strength and durability versus surface preparation; 3) Novel infield, online certification and assurance technology for surface preparation and adhesive bonding processes; 4) Reduced costs for surface preparation and adhesive bonding processes.

Project Background and Motivation

Adhesive bonding has been used in the manufacture and repair of primary aircraft structures for over 50 years and is still in use on current aircraft projects as a direct competitor to riveting. Adherend surface preparation is a critical issue to structural integrity of bonded structures. Inadequate surface roughening, possible chemical contamination on peel ply, release fabric and release film, and surface moisture may result in poor adhesion, i.e. a weak bond between the adhesive and adherend, and reduced long-term durability. The problems with chemical contaminations from peel ply, release fabric and release film that prevent adhesion of the adhesive to the substrate are now fairly well known. What is far less understood is the adverse influence of pre-bond water moisture that is unavoidable during manufacture, repair, and service. Water inclusion in pre-bond adherends could affect short-term or long-term strengths of adhesive bonding depending on the rate of diffusion and accumulation processes. As presented in the recent FAA meeting on bonding structures, water moisture is claimed as one of the most adverse factors in adhesive bonding processes. Current adhesive bonding quality assurance practice relies on tightened surface preparation process control, mechanical testing on bonded specimens and non-destructive inspection (NDI) after bonding. Thus, in the absence of a definitive surface quality control method, laborious and sometimes inadequate measures are used to ensure the quality of adhesive bonding, thereby creating an undue expense on an otherwise economic manufacturing process.

Approach to Solve the Problems

FIU will benchmark or improve the understanding of surface preparation processes, surface assurance and certification procedures using mainly information collection and analysis approach. FIU will also use atomic force microscopy (AFM), scanning electron microscopy (SEM), energy dispersive X-ray spectroscopy (EDS), and electrochemical measurements to study the surface morphology, surface chemistry, and activity of surface for peel ply samples. These analytical studies in coupling with the information analysis should allow establishment of criteria for in-field, online analytical chemistry methods for the surface preparation assurance. In addition, FIU will identify and validate technologies that are promising for the in-field surface preparation assurance. The technologies on which FIU and UM will focus include various kinds of solid-state electrochemical sensors.

Results to Date

Information collection and data analyses
FIU has reviewed a large number of articles and data with emphasis on quality control procedures for fabricating environmentally durable adhesive bonds to benchmark the understanding of current adhesive bonding technology. Analyses on conflicting results of surface pretreatments lead to the following conclusions:

1. Post-bond mechanical strength tests including the Boeing wedge test are not sufficient for certifying environmentally durable adhesive bonds.
2. Variations in bond strength and durability with the same pretreatment bonding method implied that an effective quality control procedure is needed to control and reduce the variations. Adhesive bonded joints fabricated under quality control procedures will have predictable in-service performance.
3. It was found that bond quality was affected by the nature and timing of surface hydrocarbon contamination during pretreatment peel ply or tear ply procedures, while pre-bond moisture on the adherends was the most detrimental to bond integrity.
4. The analysis indicates that a contaminate-free adherend surface is a prerequisition but not a sufficient condition for forming a strong and durable adhesive bond. A chemically activated adherend surface can enable covalent bonds between the adherend and adhesive. The covalent bonds can effectively inhibit the bond displacement due to contaminants and ingress water during service. The surface preparation certification criteria should evaluate both cleanliness and activity of the surface.
5. Certification of pre-bond surface preparation quality requires implementation of effective surface chemistry inspection technologies for each and every step of the surface preparation procedure to ensure the strength and durability of the bonded aviation structures.

Identification and validation of solid-state electrochemical sensors
FIU and UM have identified two potential sensor technologies: 1) carbon nanotube based humidity sensors and 2) mediator-assisted solid-state electrochemical sensors.

A prototype carbon nanotube (CNT) based humidity sensor has been developed (Figure 1). Experiment studies indicate that this sensor is sensitive to moisture.

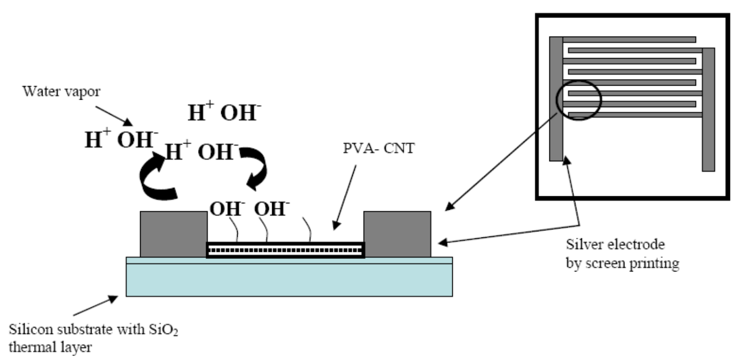


Figure 1. The schematic depicts the possible humidity sensing mechanism by PVA functionalized Y-junction single wall carbon nanotubes.

A mediator-assisted solid-state electrochemical sensor (Figure 2) has been devised, fabricated, and tested. The maximum current for fresh peel ply prepared surface is one order of magnitude lower than that of sulfuric acid treated surface. See Figure 3 for typical results. The sensor is sufficiently sensitive to surface variations. In the period of the current project, we have used Ce(I)/Ce(II), Mn(II)/Mn(III), I/I-, and Cu(II)/Cu(III) redox pairs or mediators to fabricate solid-state electrochemical sensors. Among these mediators, Mn(II)/Mn(III) showed the highest sensitivity. We have also designed and fabricated miniaturized electrochemical sensors with the sizes from several centimeters to millimeters. Reduction of the size will result in less contamination during inspection and higher sensitivity.

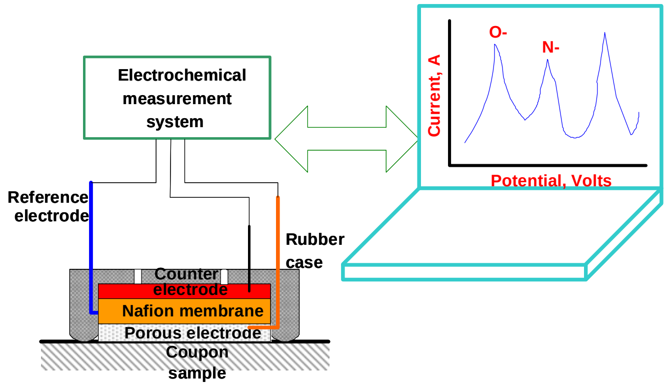


Figure 2. The schematic depicts the all solid-state electrochemical sensor for surface chemistry analysis.

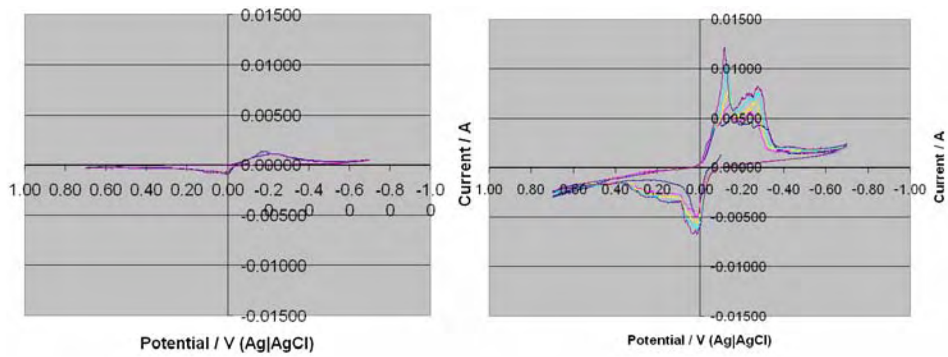


Figure 3. Left: Response (cyclic voltametry) of the Ag(I)/Ag(II) assisted solid-state electrochemical sensor on the surface of a fresh peel ply prepared laminate surface. Right: Response (cyclic voltametry) of the Ag(I)/Ag(II) assisted solid-state electrochemical sensor on the surface of a fresh peel ply prepared laminate surface treated with 50% sulfuric acid.

Atomic force microscopy (AFM) and chemical force microscopy (CFM)

AFM studies were conducted on laminate surfaces prepared with polyesther (PE), super red blue (SRB), and nylon peel plies. A new atomic force microscope with a higher resolution (5 nm) from FIU's Advanced Materials Research Institute is available. The new capabilities include an optical microscope and a tapping mode. We have been focused on force spectroscopy studies on laminate surfaces prepared with polyester, nylon, and SRB peel plies. Hundreds of readings of the maximum attractive force were obtained. The statistical analysis of the data indicate that the attractive force on the SRB prepared laminate surfaces are 4-5 times lower than those for the surfaces prepared with polyester and nylon peel plies. Functional groups have been attached to the AFM tips to conduct chemical force microscopy (CFM) studies of the laminate surfaces. These studies will help to detect the chemical composition of the laminate surface. In addition, first principle quantum mechanics modeling methods were established for evaluating the interaction between an AFM tip and laminate surface. The model will help to interpret AFM and CFM force spectroscopy.