

Full-Scale Test and Analysis of a PRSEUS Fuselage Panel to Assess Damage-Containment Features

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I. ABSTRACT

Stitched composite technology has the potential to substantially decrease structural weight through enhanced damage containment capabilities. The most recent generation of stitched composite technology, the Pultruded Rod Stitched Efficient Unitized Structure (PRSEUS) concept, has been shown to successfully arrest damage at the sub-component level through tension testing of a three stringer panel with damage in the form of a two-bay notch. In a joint effort undertaken by the National Aeronautics and Space Administration (NASA), the Federal Aviation Administration (FAA), and the Boeing Company, further studies are being conducted to characterize the damage containment features of the PRSEUS concept. A full-scale residual strength test will be performed on a fuselage panel to determine if the load capacity will meet strength, deformation, and damage tolerance requirements. A curved panel was designed, fabricated, and prepared for residual strength testing. A pre-test Finite Element Model (FEM) was developed using design allowables from previous test programs to predict test panel deformation characteristics and margins of safety. Three phases of testing with increasing damage severity include: (1) as manufactured; (2) barely visible impact damage (BVID) and visible impact damage (VID); and (3) discrete source damage (DSD) where the panel will be loaded to catastrophic failure. This paper presents the background information, test plan, and experimental procedure. This paper is the first of several future articles reporting the test preparations, results, and analysis conducted in the test program.

II. INTRODUCTION

The integrally stitched composite technology is an area that shows promise in enhancing structural integrity of aircraft and aerospace composite structures [1-5]. The National Aeronautics and Space Administration (NASA), the U.S. Air Force (USAF) and the Boeing Company have a history of developing stitched composite structures and have successfully

demonstrated this advanced structural concept in several applications, such as the C-17 main landing gear doors. Compared to conventional stiffened composite panels with co-cured or bonded interfaces, stitched composite technology offers superior out-of-plane load and damage-containment capabilities [2]. This paper introduces and provides a general overview of this joint research program and describes the development of the Pultruded Rod Stitched Efficient Unitized Structure (PRSEUS).

Stitched composite technology has been studied extensively over the past two decades [6-8]. Many coupon and element level experimental studies sponsored by NASA's Advanced Composites Technology (ACT) program investigated the effects of stitching on fracture toughness, in-plane mechanical properties, damage tolerance, fatigue response, and impact response [9]. A large amount of this mechanical property data generated by previous testing was critically reviewed to summarize the effects of stitching on stiffness, strength, and failure mechanisms [10]. Building on the ACT program, the Advanced Subsonic Technology (AST) program culminated in a full-scale test of a stitched composite wing [11]. Stitched composite technology developed during the ACT and AST programs was successfully applied to C-17 production parts in the lightly loaded Large Aircraft Infrared Countermeasure (LAIRCM) fairing, which first flew in 2003, and the multi-rib stiffened, moderately loaded, main landing gear doors, which first flew in 2007. These applications demonstrated the manufacturing and structural advantages of stitched composites [2].

After the successful production of secondary structures, Boeing's composite structure development focus shifted to large primary structures. The most recent generation of primary structure, stitched composite technology, is the PRSEUS concept, developed to avoid the weight and cost penalty of mechanical fasteners required to sustain out-of-plane loads in primary fuselage structure. Of the several goals of the PRSEUS concept, the goal most relevant to this test program is to provide damage arrestment capability for composite structures while reducing overall structural weight [2]. A key feature of the PRSEUS concept is through-the-thickness stitching, which suppresses out-of-plane damage and creates a damage arresting behavior. This advantage over conventional composite structures allows it to operate at higher strain levels, directly translating to weight savings.

The development of the PRSEUS concept to date includes testing and analysis by NASA, USAF, and the Boeing Company. The damage arrestment capability of the PRSEUS concept was demonstrated using a flat panel containing three stringers and two frames with a two-bay saw cut severing the central stringer and the adjacent skin [5]. The results from this tension test revealed that the PRSEUS concept was successful in containing the damage within a two-bay damage zone and was able to sustain 130% of the design limit load. Additional, subcomponent-level tests of damage arrestment in a minimum gauge panel, out-of-plane loading of a minimum gauge panel, and buckling of a large span were conducted to further validate the PRSEUS concept [12]. However, further research is still needed to validate the damage arrestment capabilities of the PRSEUS concept. Of particular interest is the application of the PRSEUS concept to a fuselage structure under combined internal pressure and axial tension loading.

In the current program, NASA, the Federal Aviation Administration (FAA), and Boeing have partnered in an effort to assess the damage-containment features of a full-scale curved PRSEUS

panel using the FAA Full-Scale Aircraft Structural Test Evaluation and Research (FASTER) facility. The objectives of this test are to characterize the damage arrestment features and demonstrate compliance with the strength, deformation, and damage tolerance requirements of Title 14 Code of Federal Regulations (CFR) part 25 using guidance provided by the FAA Advisory Circular (AC) 20-107B. For this purpose, the FASTER fixture has been modified to accommodate a PRSEUS panel's geometry. Enhanced axial load capacity required to meet anticipated catastrophic failure loads resulted in further fixture modifications. The test program will include loading and inspections of the panel (1) as-built, (2) with barely visible impact damage (BVID) and visible impact damage (VID), and (3) with discrete source damage (DSD) in the form of a two-bay saw cut. Various mature and developmental nondestructive inspection (NDI) methods will be used to monitor and record the extent of damage, including high-resolution visual, digital image correlation, acoustic emission, thermography, and ultrasound.

In this paper, the background information on composite structure certification is highlighted first. Then, the test panel design and fabrication are described including the geometry, load introduction reinforcements, and materials. Next, the FASTER fixture modifications and new capabilities are reviewed. Then the experimental procedure including test phases, applied loads, and inspection and monitoring methods are described. Finally, pre-test finite element modeling (FEM) is discussed. Future papers will provide results and analysis as the information becomes available.

III. BACKGROUND ON COMPOSITE STRUCTURE CERTIFICATION

Composite structure certification is based on the strength, deformation, and damage tolerance requirements of CFR 25. Guidance for an acceptable means of showing compliance with airworthiness certification requirements for composite aircraft structures is available in the FAA AC 20-107B.

Damage tolerance evaluation must show that catastrophic failure due to fatigue, environmental effects, manufacturing defects, or accidental damage will be avoided throughout the operational life of the aircraft. The damage tolerance design requirements are addressed with design requirements for load levels, where increasing damage severity requires a lower sustained load, as depicted in Figure 1. The damage tolerance and design load requirements lead to the definitions of allowable damage limit (ADL) and the critical damage threshold (CDT). The structure may have allowable (undetected) damage provided the design ultimate loads are sustained. The critical damage threshold defines damage that is easily detected and must be repaired immediately after flight.

Due to the large number of possible damage scenarios, AC 20-107B recommends conducting a damage threat assessment and suggests classifying damage into five categories summarized in the following:

- Category 1 damage: allowable damage that may go undetected. The structure must sustain ultimate loads with Category 1 damage. Some examples of Category 1 damage include barely visible impact damage (BVID) and allowable manufacturing defects.

Substantiation data must show ultimate load capability is retained for the life of the aircraft structure.

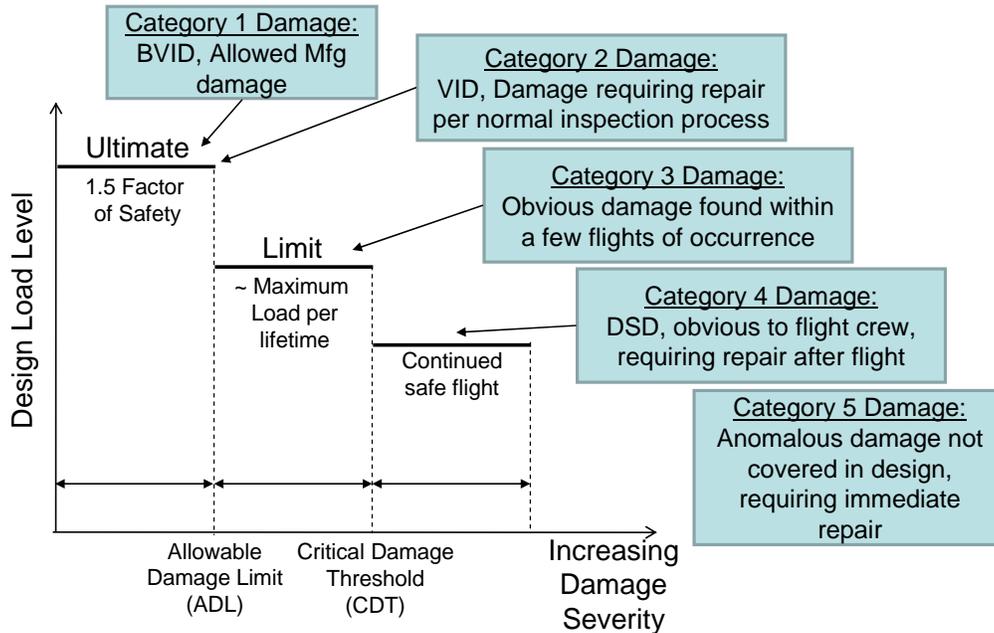


Figure 1: Increasing damage severity requires lower sustained loads.

- Category 2 damage: reliably detectable damage. The structure must sustain loads above limit load, with exact residual strength requirement dependent upon inspection method and interval. Some examples of Category 2 damage include visible impact damage (VID), detectable delamination or debonding, and visibly significant local heat or environmental degradation.
- Category 3 damage: reliably detectable by untrained personnel. The damage must be in a location such that it is obvious by clearly visible evidence. Structural substantiation requires reliable and quick damage detection while retaining limit or near limit load capability. Some examples of Category 3 damage include large VID or other obvious damage that will be caught during walk-around inspection.
- Category 4 damage: discrete source damage (DSD) from a known incident. Structural substantiation requires the structure retain damage tolerant load capability. Some examples of Category 4 damage include rotor burst, bird strikes, tire bursts, and severe in-flight hail.
- Category 5 damage: severe damage created by anomalous ground or flight events, which is not covered by design criteria or structural substantiation procedures.

In this study, the load requirements to sustain categories 1, 2, and 4, damage severity, shown in Figure 1, will be demonstrated.

IV. TEST PANEL DESIGN AND PREPARATION

The test panel geometry, materials and layout, load introduction reinforcements, and the fabrication process are highlighted in the subsections below; details are available in reference [13].

4.1 Geometry

The fuselage test panel has a 90-inch radius, 127-inch length and 75-inch width with seven full-length rod-stiffened stringers and five frames as shown in Figure 2. The nominal skin thickness in the test section, the 2-bay region identified in Figure 2, is 0.052 inches, which is a minimum gauge thickness for a 55 ksi operating stress level. The interior and exterior surface of the panel are termed the inner mold line (IML) and the outer mold line (OML), respectively.

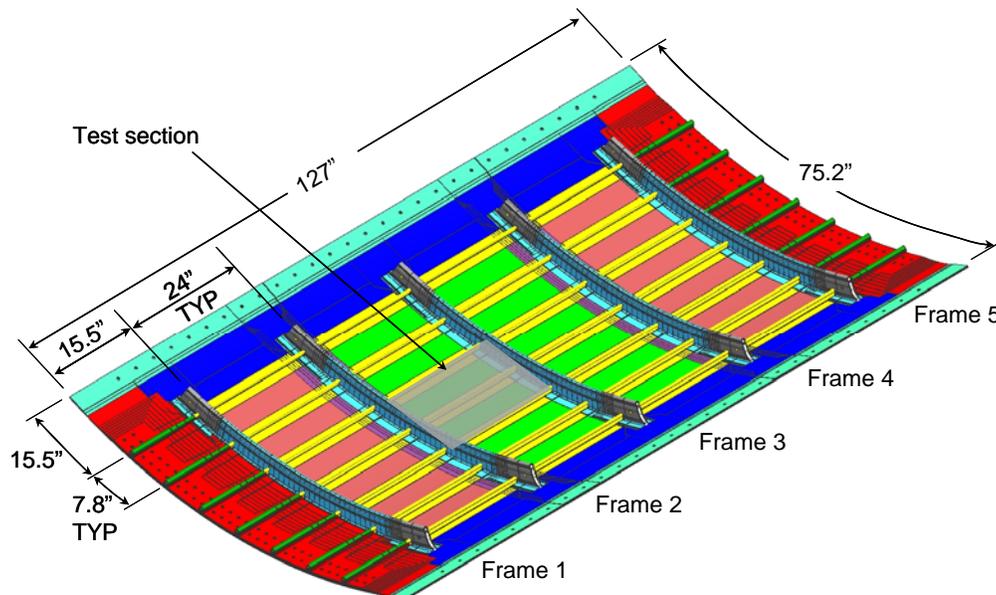


Figure 2: Overall panel geometry showing the inner mold line (IML) surface.

The skin was assembled in three pieces with splices under the second and fourth frames. Tapered splice straps were added to the OML and IML of the skin to facilitate a double-shear splice joint with uniform thickness and were stitched using a two-sided stitching process with Vectran thread.

The panel substructure comprises integrally stitched frames and stringers. The frames were formed around a half-inch thick foam core support with a 3.8-inch wide tear strap at the base. The top of the frame was reinforced with a frame cap. The detailed frame cross-section geometry, including the single-sided frame stitching denoted by dashed red arrows labeled 1 and 2, is shown in Figure 3a. The stringers are formed with a pultruded rod for high structural efficiency [1]. The stringer web is stitched using a two-sided stitching machine before pre-assembly to the panel. The detailed stringer cross section geometry, including the single-sided stringer stitching denoted by dashed red arrows labeled 1 and 2, is shown in Figure 3b.

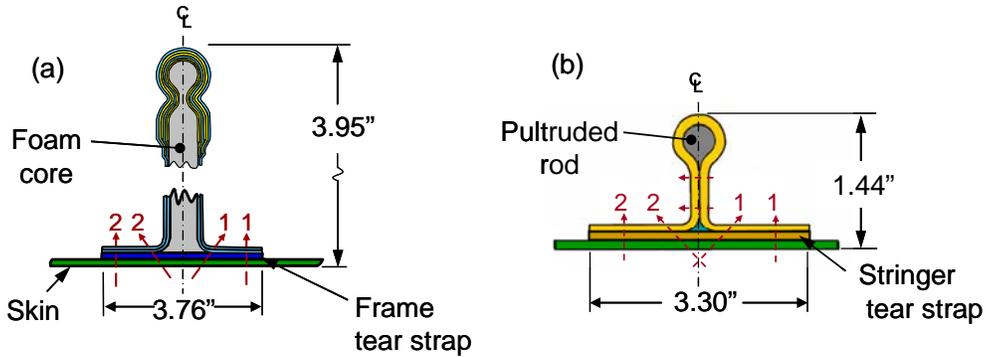


Figure 3: Frame and stringer cross section.

4.2 Test Load Introduction Reinforcements

Doublers were co-cured with the panel to reinforce the load introduction areas at the axial and hoop edges as well as frame and stringer terminations as shown in Figure 4. Transitions between the doubler regions and the test section were tapered to maintain a constant neutral axis from the load introduction region to the test section. The axial edge skin doublers were interleaved with four stringer doublers to aid in transferring load into the pultruded rod. Doublers were added to the frame ends. Bolted radius blocks were also added locally at each frame end to tie the frame flange to the skin because high peel stresses were anticipated in this region due to the test setup.

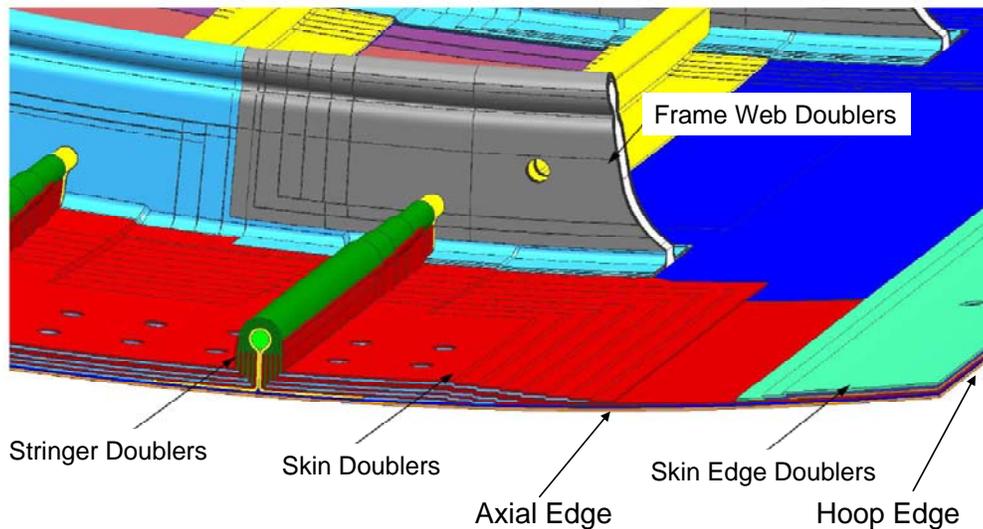


Figure 4: Load introduction doublers.

4.3 Materials & Layup

The test panel dry pre-form was fabricated using multi-axial warp-knit dry carbon fiber fabric. The fibers used in forming the warp-knit fabric were standard modulus AS4 carbon fiber. The laminates used in the panel are summarized in Table 1 for each region, with the layups given in terms of the global coordinate system where the 0°-direction is aligned with the stringers.

The frame flanges, frame tear straps, stringer flanges, and stringer tear straps were stitched to the skin using a one-sided stitching technique with 1200 denier Vectran thread at 5 penetrations per inch.

Table 1: Laminate Definitions.

Laminate	Layup % (0/45/90)	Thickness (in)
Skin	12/43/45	0.052
Splice Straps	0/100/0	0.063
Stringer Wrap	45/43/12	0.052
Stringer Tear Strap	45/43/12	0.052
Frame Web	0/100/0	0.031
Frame Tear Strap	12/43/45	0.052
Frame Cap	33/33/33	0.094
Axial Load Doublers	20/43/27	0.572
Hoop Load Doublers	23/43/34	0.312
Frame Load Doublers	33/33/33	0.141

4.4 Fabrication and Inspection

Fabrication of the panel was performed in the Marvin B. Dow Stitched Composites Development Center at the Boeing facility in Huntington Beach, California using existing tooling from a prior PRSEUS project [13]. The panel fabrication began with pre-assembly of the frames and stringers, and was followed by pre-form assembly. With final assembly completed, the pre-form was then placed in the mold tool. The pre-form was infused with VRM-34 epoxy resin and cured, using a Boeing proprietary process called Controlled Atmospheric Pressure Resin Infusion (CAPRI), an out-of-autoclave low-pressure process that results in higher fiber volume fractions than the conventional Vacuum Assisted Resin Transfer Molding (VARTM) process [2].

The skin, frame and stringer flanges, and edge doubler regions of the panel were then inspected using through-transmission ultrasonic C-scan at 5.0 MHz. No defects were identified. The mean average dB levels were consistent over the entire panel, with actual deviation at any one location within the acceptable ± 6 dB allowed variation.

V. FASTER FIXTURE MODIFICATIONS

The Full-Scale Aircraft Structural Test Evaluation and Research (FASTER) fixture was originally designed for accelerated fatigue studies of metallic fuselage structures [14]. The fixture can apply combined internal pressure and axial load with appropriate hoop reactions to narrow and wide body fuselage panels. For the purpose of conducting the PRSEUS panel test, the FASTER fixture was modified to accommodate a PRSEUS panel including the enhanced axial load capacity required to apply catastrophic failure loads. The new longitudinal loading mechanism now operates using hydraulic jacks with a stand-alone hydraulic system, controlled by the existing computer control system.

The FASTER fixture modifications include accommodations for a larger panel radius, a modular seal for full-length stringers, a frame loader slide system for large axial displacements, and an increased axial load capacity. A comparison of the existing fixture and modified fixture is shown in Figure 5, highlighting the new axial load mechanism. The axial loading features of the modification are discussed in the following subsections.

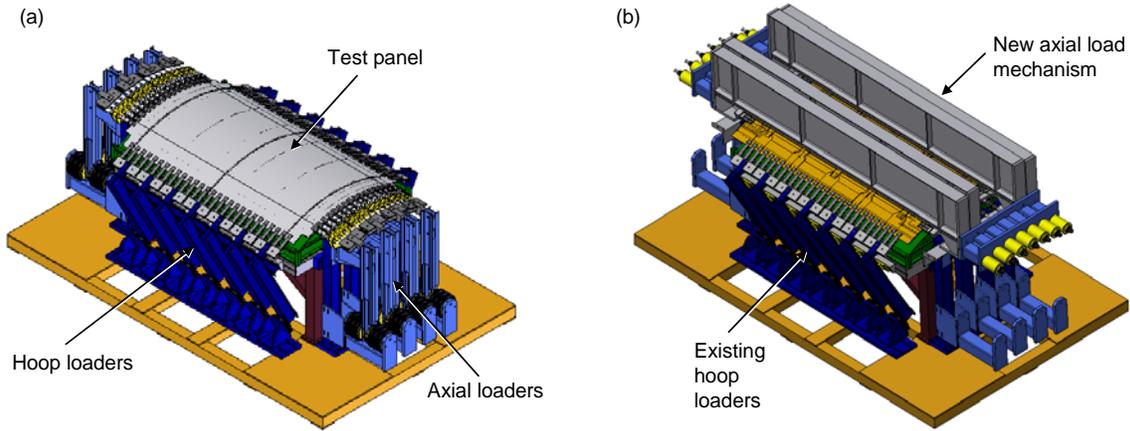


Figure 5: Comparison of (a) existing FASTER fixture and (b) modified fixture.

5.1 Frame Loader Slide-System

Initial test predictions show an overall axial displacement of up to 0.25 inches between the outer frames. This displacement would have induced large reaction forces into the frames with the existing rigid frame loader design that are not characteristic of flight loads. A frame loader slide-system was designed that allows the frame loaders to move freely in the axial direction avoiding non-flight loads, as shown in Figure 6.

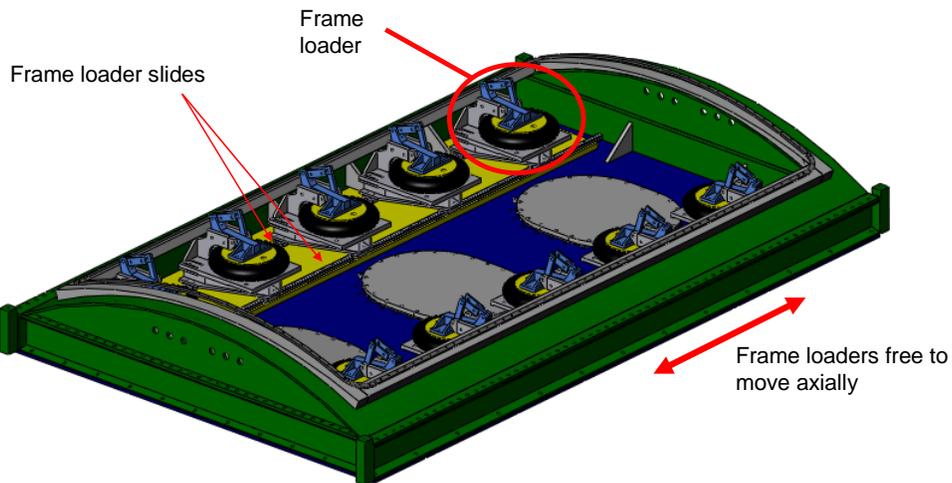


Figure 6: Frame loader slide-system.

5.2 Axial Load Mechanism

Failure loads for the PRSEUS panel were predicted to exceed the existing FASTER fixture axial load capacity of 100 kips. Based on previous PRSEUS testing, a new axial load mechanism was designed with a capacity of 840 kips to catastrophically fail the PRSEUS test panel. The significant increase in axial load capacity was accomplished using fourteen 60-ton hydraulic jacks connected to each end of the panel with a whiffletree assembly as shown in Figures 7a and 7b. The axial loading is reacted by four built-up beam columns, which measure 178 inches long and 24 inches tall, each weighing approximately one ton. A detail view of a single axial load whiffle tree assembly is shown in Figure 7c.

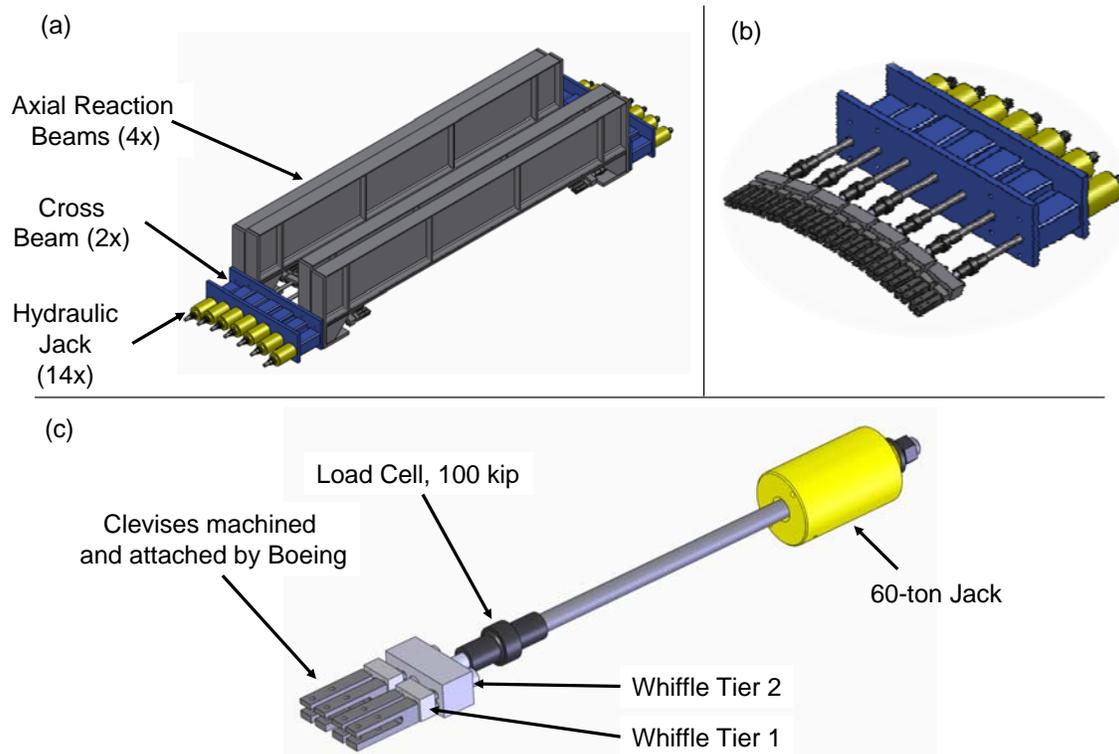


Figure 7: Axial load mechanism assembly (a). A cross beam with seven whiffletrees is shown in (b). A detail view of an axial whiffle tree is shown in (c).

VI. EXPERIMENTAL PROCEDURE

6.1 Test Phases

Testing will be performed in three phases with increasing damage severity in order to show compliance with CFR 25 requirements for structure. The three phases of testing follow the flow chart illustration in Figure 8. In the following subsections, each phase is described in detail including the objective, damage case, applied loads, and inspection and monitoring methods.

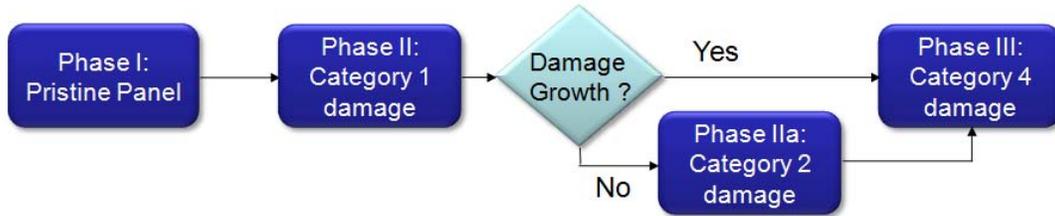


Figure 8: Test phase flow chart.

6.1.1 Phase I: As Manufactured Pristine Panel

The objective of phase I is to demonstrate that the as-manufactured pristine panel meets strength and deformation requirement of CFR 25.305. During phase I, strains and displacements will be monitored to validate future analytical models of the PRSEUS panel.

6.1.2 Phase II: Category 1 Damage: barely visible impact damage (BVID)

The objective of phase II is to demonstrate that the panel with Category 1 damage in the form of BVID, meets the strength and deformation requirement of CFR 25.305. Category 1 damage is defined as damage that may go undetected by scheduled or direct field inspection or allowable manufacturing defects by AC 20-107B. The Category 1 damage will be created using a portable impactor with a 1" diameter hemispherical tup on a cylindrical weight and impact energy of 40 ft-lbs. The damage location is shown in Figure 9a in reference to the panel OML and in Figure 9b in reference to the central stringer. During phase II, testing damage growth will be monitored and quantified using NDI techniques.

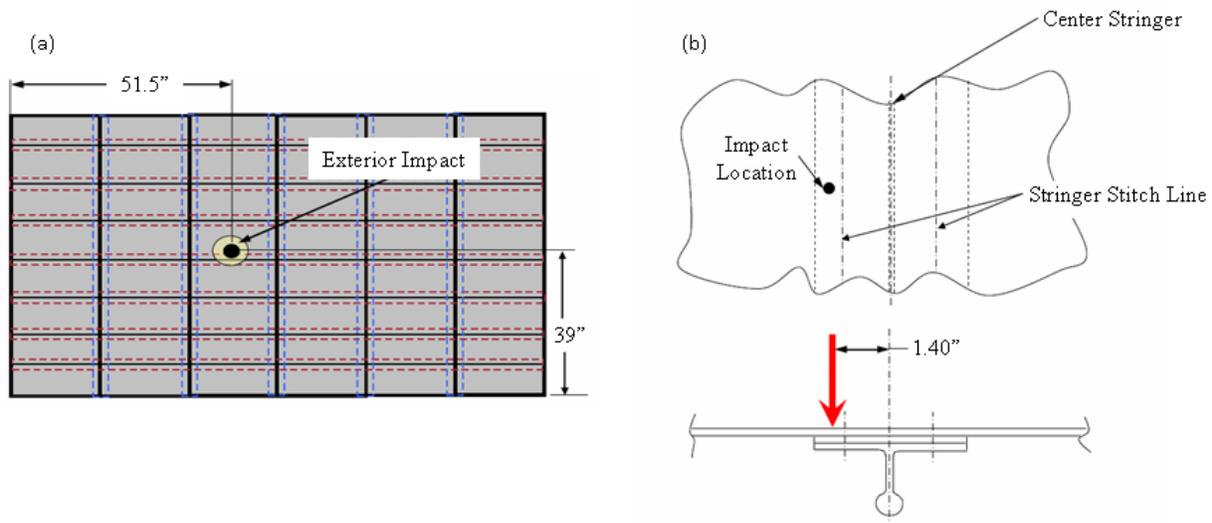


Figure 9: Phase II damage location with respect to (a) the panel OML and (b) the central stringer.

6.1.3 Phase IIa: Category 2 Damage: visible impact damage (VID)

In the event that no damage growth is observed after loading the panel with BVID to ultimate load, Category 2 damage will be created at the same location using the portable impactor. Category 2 damage is defined by AC 20-107B as damage that can be reliably detected by scheduled or directed field inspections performed at specific intervals. The objective is to show the panel can withstand Category 2 damage subjected to the corresponding load levels, as required by CFR 25. Loading stages will mirror phase II. During phase IIa testing, damage growth will be monitored and quantified.

6.1.4 Phase III: Category 4 Damage: discrete source damage (DSD)

The objective of phase III is to demonstrate damage tolerance for DSD, in the form of a 2-bay saw-cut notch severing the central stringer, as shown in Figure 10. Compliance with CFR 25.571 is demonstrated by withstanding limit loads with discrete source damage. Then loading will be increased to catastrophic failure. During phase III testing damage growth and ultimate failure load will be monitored and quantified.

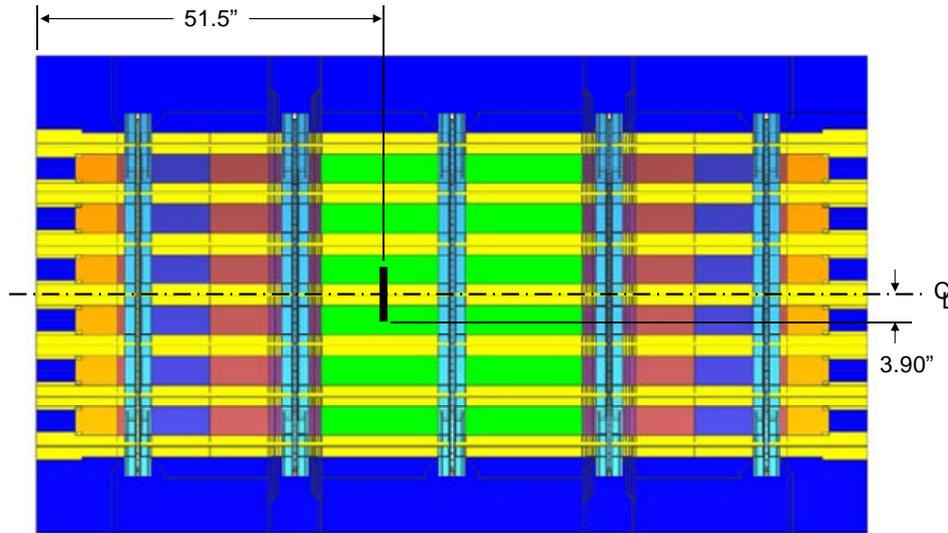


Figure 10: Discrete source damage saw cut position and dimensions.

6.2 Applied Loads

Axial tension load and internal pressure with corresponding reaction loads will be applied. Loads are specified relative to the operating loads. For this panel, the axial Design Limit Load (DLL) is 340,000 lbs and the design internal operating cabin pressure referred to as P is 9.2 psi. The axial Design Ultimate Load (DUL) is defined as 1.5 times the DLL. The load definitions are listed in Table 2. The test matrix shown in Table 3 indicates which loads are applied during each phase.

Table 2: Applied Load Definitions & Corresponding CFRs.

Loading Mode	Load Description	CFR	Relative Pressure	Numeric Load
Internal Pressure	Operational cabin pressure	-	P	9.2 psi
	Limit load pressure	25.571	1.15P	10.6 psi
	Pressure cabin loads	25.365	1.33P	12.2 psi
	Ultimate pressure cabin loads	25.305	2P	18.4 psi
Axial	Design limit load (DLL)	25.571	DLL	340,000 lbs
	Design ultimate load (DUL)	25.305	1.5DLL	510,000 lbs

Table 3: Test Matrix.

Phases		Load Levels			
		30% Limit Loads	CFR 25.571 Damage Tolerance	CFR 25.365 Pressurized Compartment	CFR 25.305 Strength and Deformation
I	Pressure	✓	✓	✓	✓
	Axial	✓	✓	N/A	
	Combined	✓	✓	✓	
II	Pressure		✓	✓	✓
	Axial		✓	N/A	✓
	Combined		✓	✓	✓
III	Pressure		✓		
	Axial		✓	N/A	
	Combined		✓		

6.3 Inspection and Monitoring Methods

Test results will be acquired using visual, strain and displacement, and damage detection methods. The inspection and monitoring methods are summarized in the following subsections.

6.3.1 Visual Inspection

Two real-time methods of high-magnification visual inspection will be used to monitor this test: (a) a remote-controlled crack monitoring (RCCM) system for exterior and interior imaging and damage measurement and (b) a high-speed camera system for recording video of exterior damage events.

6.3.2 Strain and Displacement

Strain gages, linear variable displacement transducers (LVDTs), and an ARAMIS Photogrammetry system will be used to record strain, displacements, and deformation. The test panel will be instrumented with strain gages to monitor real-time strain distribution to ensure proper load introduction from the load application points. Strain gages will be installed in

proximity to the skin, frames, and stringers, on the OML and IML as shown in Figures 11 and 12.

Several strain gages will be installed in close proximity to stitched features to monitor the behavior of the panel near the stitches during the test. Select gages will function as back-to-back gages to monitor and record bending during the test. LVDTs will be installed to measure total axial displacement and out-of-plane deflection of the panel as shown in Figure 11. Strain gage and LVDT readings will be recorded continuously to a buffer file at a rate of 150 Hz during all testing phases.

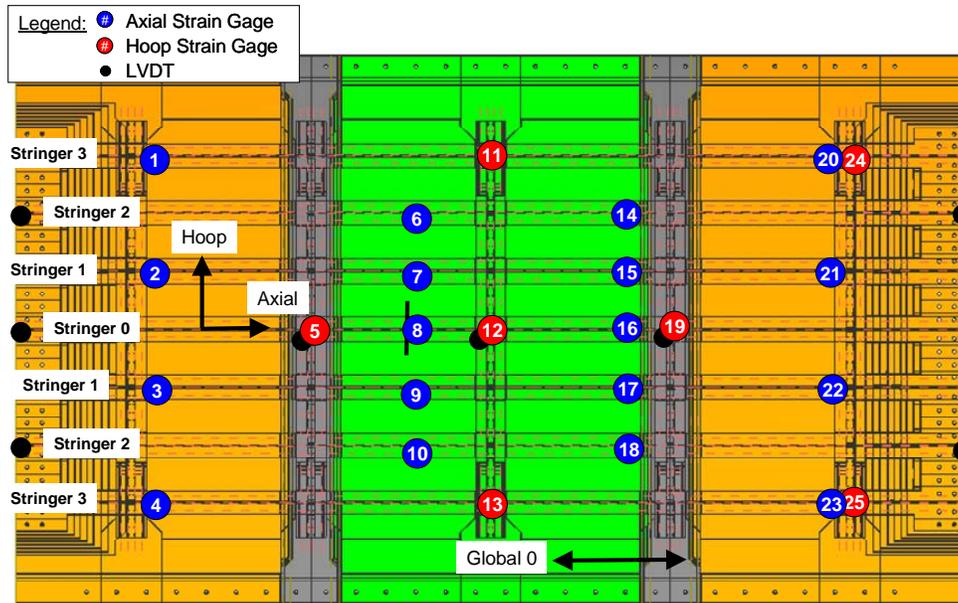


Figure 11: OML strain gage and LVDT locations.

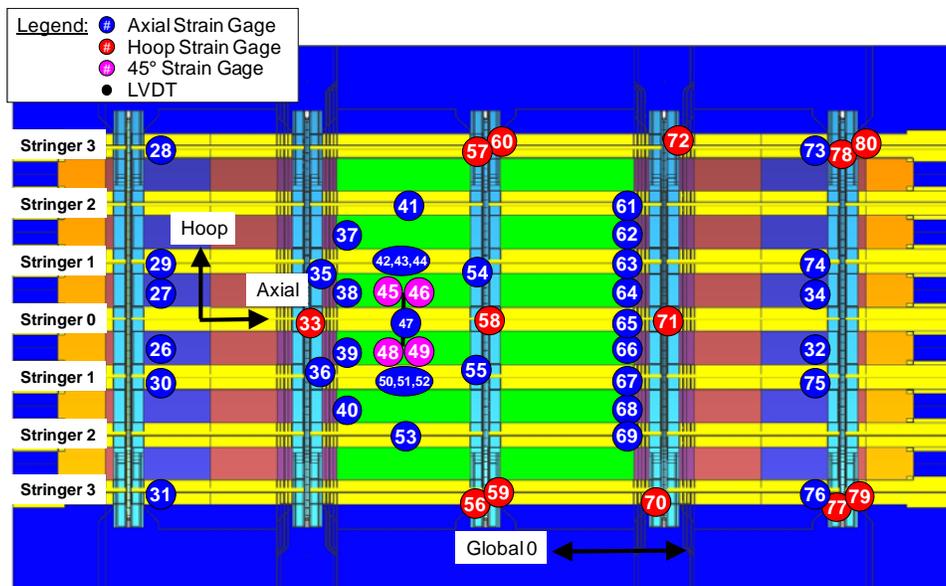


Figure 12: IML strain gage locations.

Full-field deformation and strain data will be recorded during the three phases of testing using ARAMIS™ three-dimensional strain photogrammetry systems. One system will use standard 4-megapixel cameras capable of accurately measuring full-field strain within 50 $\mu\epsilon$ and a second system may use high-speed video cameras for high-speed ARAMIS results. Prior to testing, the areas to be monitored by the photogrammetry system will be coated with a high-contrast stochastic speckle pattern. Baseline images will be taken using both systems at zero load to establish a noise floor. Deformed images will be recorded using both systems at several applied load levels. During phase I, load introduction will be monitored using a real-time measurement feature of the ARAMIS software whereby full-field strains are computed immediately after images are captured. During phase II and III, images of damage areas will be captured for post processing.

6.3.3 Damage Inspection and Detection

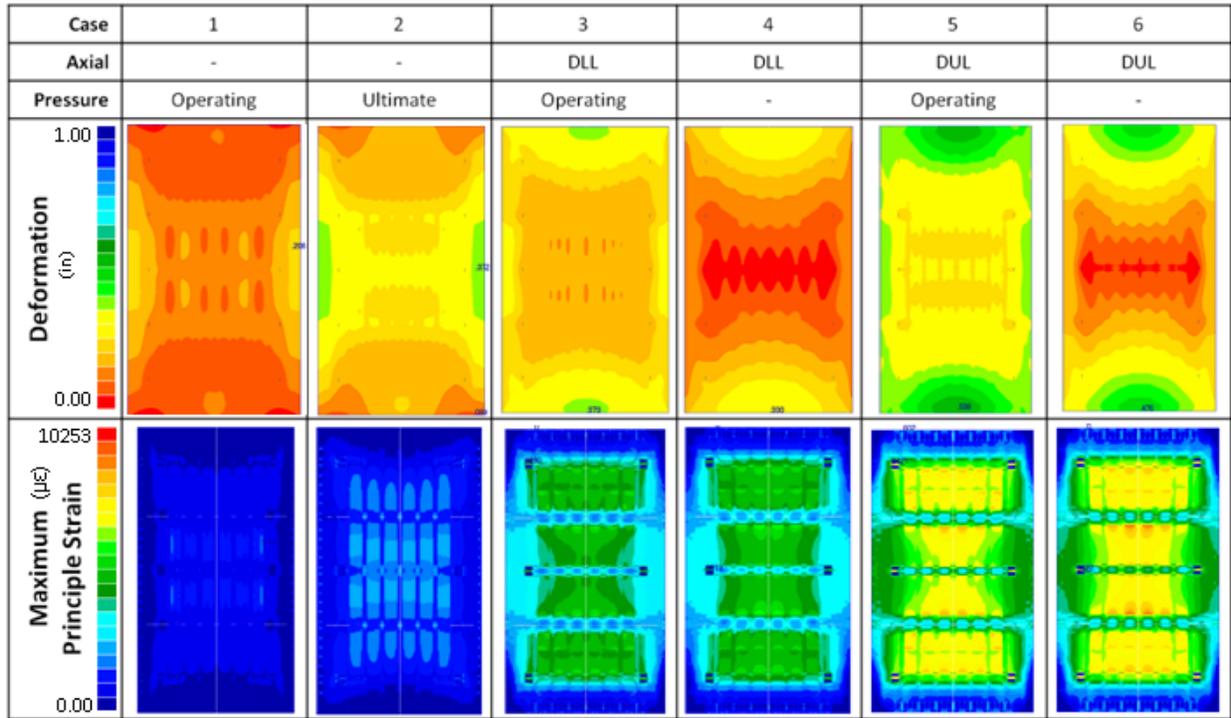
Damage will be quantified at several intervals during the testing using a portable c-san. Additional mature and emerging inspection methods such as, flash thermography, computer-aided tap test, ultrasonic resonance, sonic infrared, and acoustic emission, may also be used to identify and quantify damage.

VII. PRE-TEST ANALYTICAL PROCEDURE

A linear elastic finite element model (FEM) was developed for use in determining the test loads for each actuator, corresponding ply-level strains, and placement of data acquisition sensors. The analysis model, which used shell and bar elements to represent the panel and the FASTER fixture, was also used to ensure that the final failure would occur in the test section [13]. Details including frame foam cores and radius blocks, sub-structure interface, and load introduction joints were simplified to minimize modeling effort. Material properties from ACT program test results were used [4].

The FEM was used to predict displacements and strains, and to calculate margins of safety. The test sequence was written with loads applied in increasing severity based on FEM results. As shown in Table 4, fringe plots were generated to illustrate the deformed shapes and the maximum principle strain distribution for six loading cases. All deformation fringe plots are consistent with the applied symmetric loading. The plots predict that the load introduction edge-effects are minimized in the test section where damage will be introduced.

Table 4: Deformation and maximum principle strain fringe plots for six loading cases.



Predicted maximum principle strains, $\epsilon_{p,max}$, in the skin, stringer wrap, stringer tear strap, and frame tear strap were compared to allowable strain, $\epsilon_{allowable}$, from the ACT program tests and previous PRSEUS tests [4]. The stringer rod was also included in the comparison; however, maximum principle stresses, $\sigma_{p,max}$, were compared to allowable stress, $\sigma_{allowable}$, because allowable strain was not available. Table 5 shows a summary of the margin of safety comparison, where margin of safety is defined as $\epsilon_{allowable}/\epsilon_{p,max}-1$ and, in the case of the stringer rod, as $\sigma_{allowable}/\sigma_{p,max}-1$. As expected, the margin of safety for the skin approaches zero under ultimate axial load.

Table 5: Margin of safety summary.

Load Case		Skin 9172 μ in/in*	Stringer Wrap 10254 μ in/in*	Stringer Tear Strap 10254 μ in/in*	Stringer Rod 390 ksi**	Frame Tear Strap 9172 μ in/in*
Axial	Internal Pressure					
-	Operating	2.82	12.78	5.63	25.66	7.23
-	Ultimate	1.08	5.89	1.50	12.57	2.99
DLL	Operating	0.57	0.73	1.01	2.74	0.91
DLL	-	0.50	0.66	0.94	2.64	0.88
DUL	Operating	0.06	0.17	0.36	1.54	0.26
DUL	-	0.01	0.13	0.31	1.45	0.266

*Allowable principle strain, $\epsilon_{allowable}$, for each structural component.

**Allowable principle stress, $\sigma_{allowable}$.

VIII. SUMMARY

A new test program to characterize the damage containment features in a full-scale curved fuselage panel subject to combined loading has been undertaken by NASA, the FAA, and the Boeing Company. Composite strength, deformation, and damage tolerance certification requirements were highlighted. The test panel was designed, fabricated, and prepared for the residual strength testing. A pre-test FEM was developed using design allowables from the ACT program testing to predict test panel deformation characteristics and margins of safety. Using the FEM results, a test plan including three phases of testing, applied loads, and inspection and monitoring requirements was completed. The test will comprise three phases of testing with increasing damage severity including (1) as manufactured, (2) barely visible impact damage, and (3) discrete source damage where the panel will be loaded to catastrophic failure. In addition, modifications to the FASTER facility were designed to accommodate the PRSEUS test panel. Results from this on-going test program will be published as they become available.

IX. REFERENCES

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