TEST AND ANALYSIS OF FUSELAGE STRUCTURE TO ASSESS EMERGING METALLIC STRUCTURES TECHNOLOGIES

Yongzhe Tian[1,2], David Stanley[2], John G. Bakuckas, Jr.[2], Kevin Stonaker[2], Mike Kulak[3], Walt Sippel[4], Kimberly Maciejewski[5], Marcelo R. B. Rodrigues[6], Fabricio Fanton[6], and Carlos E. Chaves[6]

[1] Corresponding Author: yongzhe.tian@faa.gov
[2] FAA William J. Hughes Technical Center, Atlantic City, NJ 08405, USA
[3] Diakon Solutions LLC, Cape May Court House, NJ, 08210, USA
[4] FAA Policy and Innovation Division, Des Moines, Washington 98198, USA
[5] Arconic Technology Center, New Kensington, PA, 15068, USA
[6] Embraer S.A., São José dos Campos, Brazil

ABSTRACT: In partnership with Arconic and Embraer, the Federal Aviation Administration (FAA) is assessing emerging metallic structures technologies (EMST) using the FAA's Full-Scale Aircraft Structural Test Evaluation and Research (FASTER) facility. In this collaborative effort, full-scale fuselage panel test data are being generated to assess the effect of EMST fuselage concepts on damage tolerance performance as compared to the baseline aluminum fuselage structures located on the crown of a typical single-aisle aircraft forward of the wing. Several technologies will be considered, including advanced aluminum-lithium (Al-Li) alloys and selective reinforcement using fiber metal laminates (FML). Data from this study will be used to verify improved weight and structural safety performance of the EMST and will also be used to assess the relevance of existing regulations and to inform whether additional safety standards and regulatory guidance should be developed to provide improved safety beyond that afforded by the existing airworthiness standards. The focus of this paper was on assessing two fuselage panels with advanced aluminum alloys, which demonstrated improved damage tolerance performance compared to conventional baseline aluminum panel.

Keywords: Advanced Technologies, Damage Tolerance, Crack Growth, Residual Strength, Aircraft Fuselage Structure

INTRODUCTION

The aircraft industry is striving to improve aircraft performance and reduce costs in fabrication, operations, and maintenance by introducing advanced materials in conjunction with innovative manufacturing and production technologies. Significant advancements have been made over the past decade by the aerospace industry in developing new lightweight alloys and product forms, improved structural concepts, and manufacturing processes aimed at being competitive with composite materials in terms of manufacturing cost and performance. Collectively, these advances fall under the umbrella classification of emerging metallic structures technologies (EMST). Substantial investments have been made to demonstrate the potential to design and build durable and damage-tolerant fuselage and wing

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structure using EMST technologies including advanced alloys [1,2], bonding and joining methods [3-6], and metallic-composite hybrids [3, 4, 6-9].

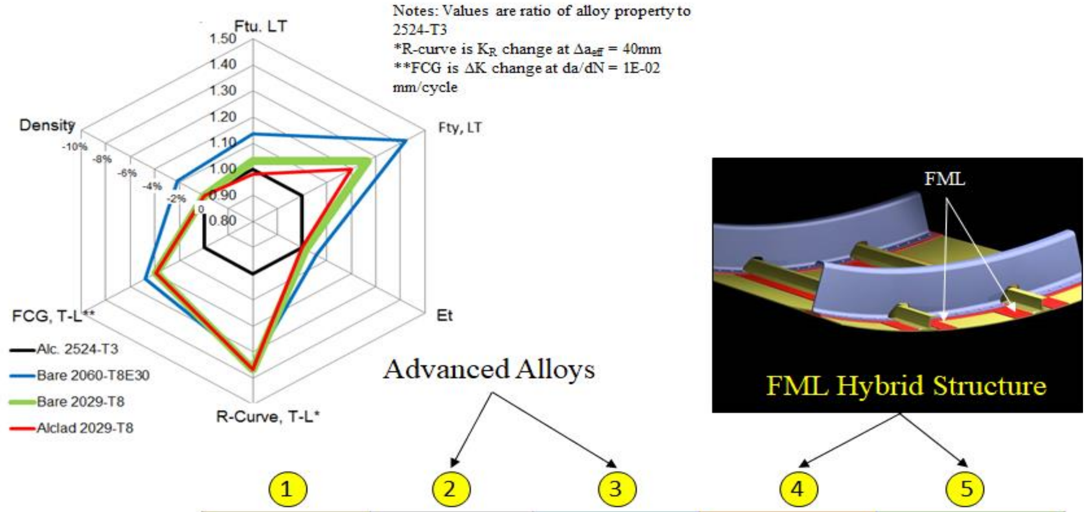
However, the introduction of a new material or concept in the aerospace industry can be quite challenging. A significant amount of test data at the coupon, substructure, and full-scale level are needed to fully vet and properly assess a new technology and understand potential certification and continued airworthiness issues. In addition, the assessment should also include an evaluation of the existing regulations and guidance materials to determine whether they are still relevant for the new technology or need changing. For these reasons, regulators and industry ideally should work together in preparation for the application and certification of EMST. In recognizing these challenges, the Federal Aviation Administration (FAA), Arconic, and Embraer are collaborating in a research effort to evaluate EMST for fuselage applications through full-scale testing and analysis. The goal is to assess and verify the use of EMST to improve durability and damage tolerance performance compared with the baseline aluminum fuselage located on the crown of a typical single-aisle aircraft forward of the wing spar.

For this program, five fuselage panels are planned to be tested, as shown in Figure 1. The first fuselage panel is baseline Panel 1, and is constructed by using conventional materials and processes. The other four fuselage panels incorporate varying EMST. Panels 2 and 3 are constructed from advanced aluminum alloys, such as aluminum-lithium (Al-Li). Panels 4 and 5 will include fiber-metal laminates (FML) reinforcing straps. Both fastened and bonded straps underneath stringers and bonded straps under frames will be evaluated. All panels will be tested using the FAA's Full-Scale Aircraft Structural Test Evaluation and Research (FASTER) facility. This equipment is specifically designed to test fuselage panels, and is capable of simulating aircraft service load conditions through the synchronous application of mechanical and environmental load conditions [10], as shown in Figure 2.

A phased approach is being used to study two damage scenarios in all panels: 1) a two-bay skin crack, along the circumferential direction, with central stringer severed; 2) a two-bay skin crack, along the longitudinal direction, with the central frame severed. For each damage scenario phase, strain surveys will first be conducted and compared to finite element predictions to verify proper load and panel alignment. The panels will then be subjected to fatigue crack growth (FCG) testing using an equivalent constant amplitude load sequence determined through coupon-level tests that represent the complex loading of a fuselage panel, which is assumed to be located on the crown of the aircraft, and forward of the wing [11,12]. An elevated fuselage pressure differential was used in the load sequence to demonstrate potential improvements in the durability and damage-tolerance characteristics of aircraft equipped with EMST. The pressure differential used for the testing was approximately 15% higher than what is typical for a single-aisle transport category aircraft, such as the Boeing 737 and Airbus A320 model airplanes. The final stage of testing will be a residual strength test to certain limit load conditions.

Initial efforts focused on the baseline Panel 1, which was the first panel to be tested. Baseline Panel 1 was constructed by riveting Al-clad 2524-T3 skin to conventional 7000-series aluminum substructure. The results from the baseline Panel 1 test including both circumferential and longitudinal damage scenarios were presented at ICAF 2019 and published [13] and are also included in this paper for the purpose of comparing those results with Panels 2 and 3 results. The latest tests of fuselage Panel 2 and Panel 3 will investigate the durability and damage-tolerance characteristics of certain advanced alloys and will evaluate both damage scenarios. Fuselage Panel 2 and Panel 3 are built with 2060-T8 Al-Li and 2029-T3 clad aluminum skins, respectively.

During all phases of testing, crack growth was monitored and recorded using high-magnification cameras, several nondestructive inspection (NDI) methods, strain gages, and a digital image correlation (DIC) system. For each phase, prior damage was repaired. Results from first three panels are presented in this paper, summarizing experimental and analytical procedures, and will be compared with future panels using FML reinforcement to assess the damage tolerance performance.



		Baseline	Advanced Density Reduction	Advanced Materials	FML Reinforced	FML Reinforced (Optimized for Weight)
Component	Skin	2524-T3 sheet	2060-T8E30 Al-Li sheet	2029-T3 sheet	2524-T3 sheet	2524-T3 sheet
Component	Stringer	7150-T77511 extrusions, riveted	2055-T84 Al-Li extrusions, riveted	2055-T84 Al-Li extrusions, riveted	7150-T77511 extrusions, with FML straps	7150-T77511 extrusions, with FML straps
Component	Frame	7075-T62 - shear tied, extruded, riveted	2099-T83 Al-Li integral extrusions, riveted	2099-T83 Al-Li integral extrusions, riveted	7075-T62 - shear tied, extruded with FML straps	7075-T62 - shear tied, extruded with FML straps
Schedule	Start	Oct-17	Jan - 19	June - 21	Sep - 23	Jan - 25
Schedule	Finish	Dec-18	May -21	Aug - 23	Dec – 24	Dec - 25

Figure 1: EMST project test matrix.

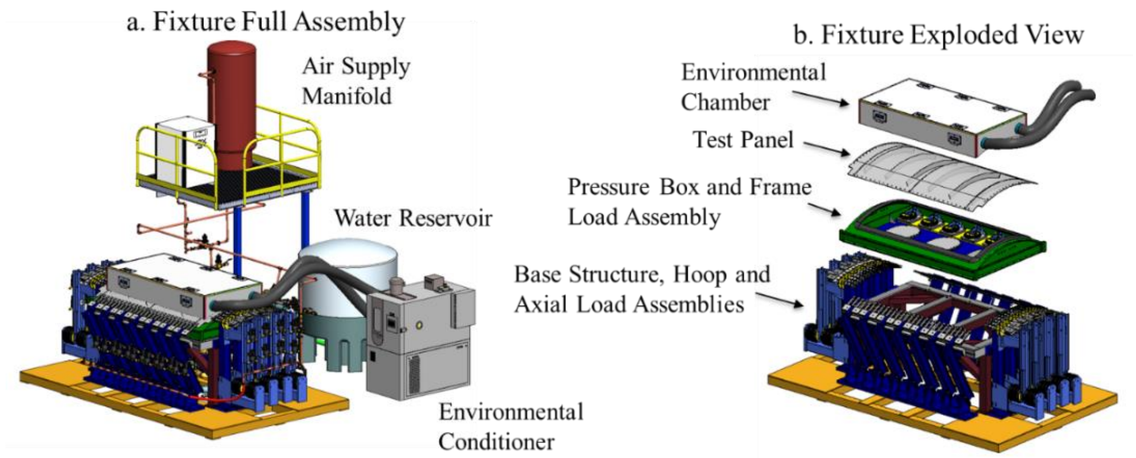


Figure 2: FAA FASTER fixture assembly and major components [10].

EXPERIMENTAL PROCEDURES

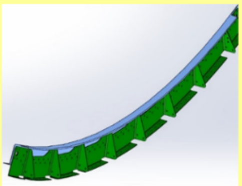
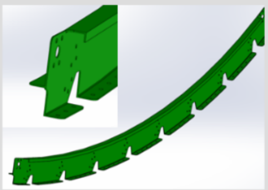
Testing for this program was conducted using the FAA's FASTER facility. A description of the test panel, test phases, applied loads, and inspection and monitoring methods are outlined in this section.

Target Application and Panel Description

For this study, the fuselage panels were built to be similar to a typical single-aisle airplane fuselage panel, such as that found on either Boeing 737 or Airbus A320 model airplanes. The location of the fuselage panel is assumed to be on the crown, and just forward of the wing, where the major modes of loading are pressurization and vertical bending due to flight and landing loads. LMI Aerospace was contracted by Arconic to fabricate panels 1-3 using standard aerospace manufacturing practices, including, but not limited to, forming, chemical milling, surface treatment, and joining technologies. This paper only presents the results of panels 1-3 since Panel 4 and 5 have yet to be built and tested. Therefore, they are not discussed hereinafter.

A description of Panels 1-3 is provided in Table 1. Baseline Panel 1 was constructed using 2524-T3 aluminum skin reinforced with 7000 series aluminum stringers and frames. For Panels 2 and 3, metallic skins consisted of 2060-T8 Al-Li and 2029-T3 clad aluminum, respectively, both reinforced with Al-Li 2055 stringers, and Al-Li 2099 integral frames. All panels were 3175 mm by 1854 mm with a radius of 1880 mm, as shown in Figure 3. Reinforcing doublers were installed along the outer perimeter of the skin and to the frame ends for load attachment points of the fixture. Holes of 12.7 mm diameter were drilled along the reinforced doubler edge of the panel for load introductions in the axial and hoop directions. The stringer spacing is 178 mm and frame spacing is 508 mm for all three panels.

Table 1. Material and configuration of skin, stringer and frame.

		Panel 1 Baseline	Panel 2 Advanced Density Reduction	Panel 3 High strength, corrosion-resistant
Component	Skin	2524-T3 sheet, 1.5mm	2060 - T8E30 Al-Li, 1.27mm	2029 – T3, 1.35mm
	Stringer	7150-T77511 extrusions, riveted	2055 -T84 Al-Li extrusions, riveted	2055 -T84 Al-Li extrusions, riveted
	Frame	7075-T62 – floating frame, shear tied, extruded, riveted 	2099 - T83 Al-Li integral frame and shear tie extrusions, riveted 	2099 - T83 Al-Li integral frame and shear tie extrusions, riveted 

Test Phases and Damage Scenarios

A phased approach will be used to study the two damage scenarios for all three panels in this program. Phase 1 was completed using circumferential damage, and results for the first three panels are reported in this paper. Phase 2 efforts are under way with longitudinal cracking as damage scenario.

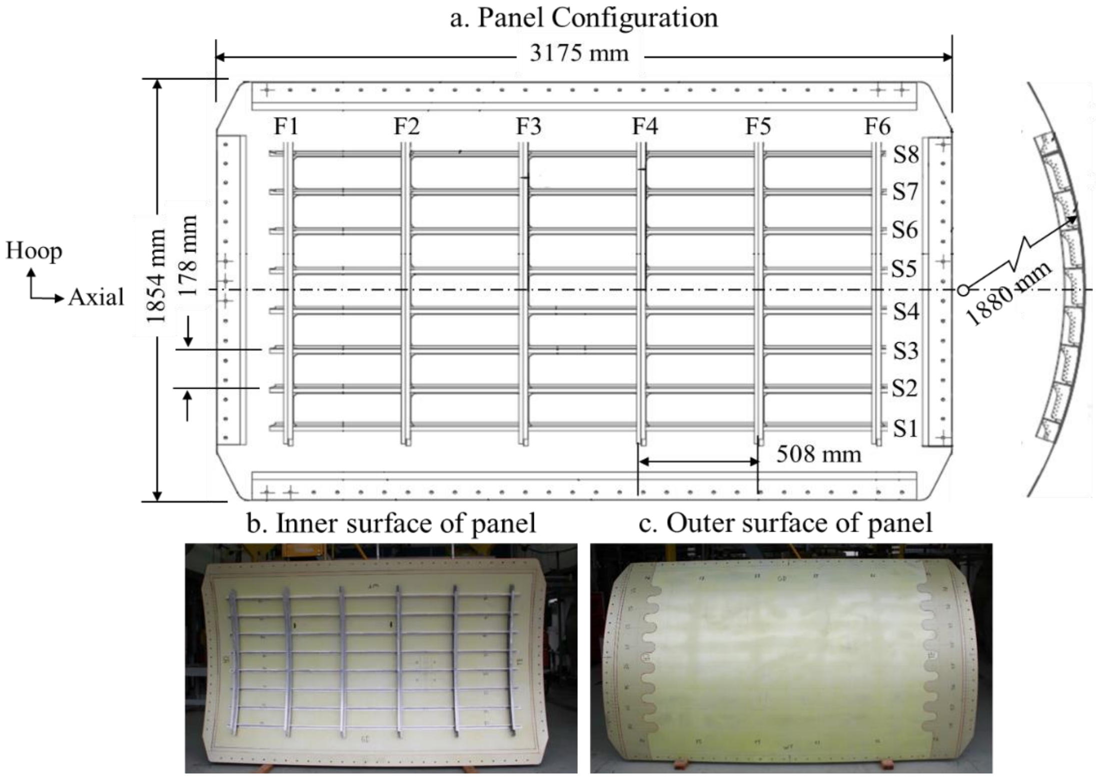


Figure 3: Baseline panel configuration and views of the internal and external surface.

Phase 1: The initial damage is comprised of a two-bay circumferential skin crack having a total length of 33 mm between frames F2 and F3, with the central stringer S4 severed (see Figurea). Strain surveys were conducted to ensure proper load introduction. The panel was then fatigue tested under loads representing pressure, flight maneuver, and gust accelerations, and landing loads in the forward crown section of a single-aisle aircraft. Fatigue testing was conducted until the crack extended to a final total length of 287 mm. Afterwards, a limit load test was conducted, during which the panel was subjected to approximately 2.5G axial load while holding the pressure constant under operational conditions. The panel was then repaired for the next phase of the program.

Phase 2: The initial damage is comprised of a two-bay longitudinal skin crack having a total length of 82 mm between stringers S6 and S7, with the central frame/shear tie F4 severed, (see Figureb). Strain surveys were conducted to ensure proper load introduction and to verify no effects from the repair made in Phase 1. The panel was then fatigue tested, simulating pressure-only operational conditions until the crack extended to a final length of approximately 406 mm. Finally, a residual strength test will be conducted to failure measuring the load-carrying capacity of the panels for this final damage condition.

Inspection and Monitoring Methods

During all phases of testing, several NDI methods were used to monitor and record the formation and growth of cracks. Visual inspections were made on the inner and outer surfaces of the skin using high-magnification cameras that could be remotely controlled during the test. High-frequency eddy current was used on the outer surface of the skin. Along with these inspection methods, the panels were instrumented with up to 200 strain gages and a DIC system to monitor strains throughout the tests. In addition, a commercial piezoelectric-based structural health monitoring system was used to collect data and to assess its capabilities to monitor FCG.

Applied Mechanical Loads

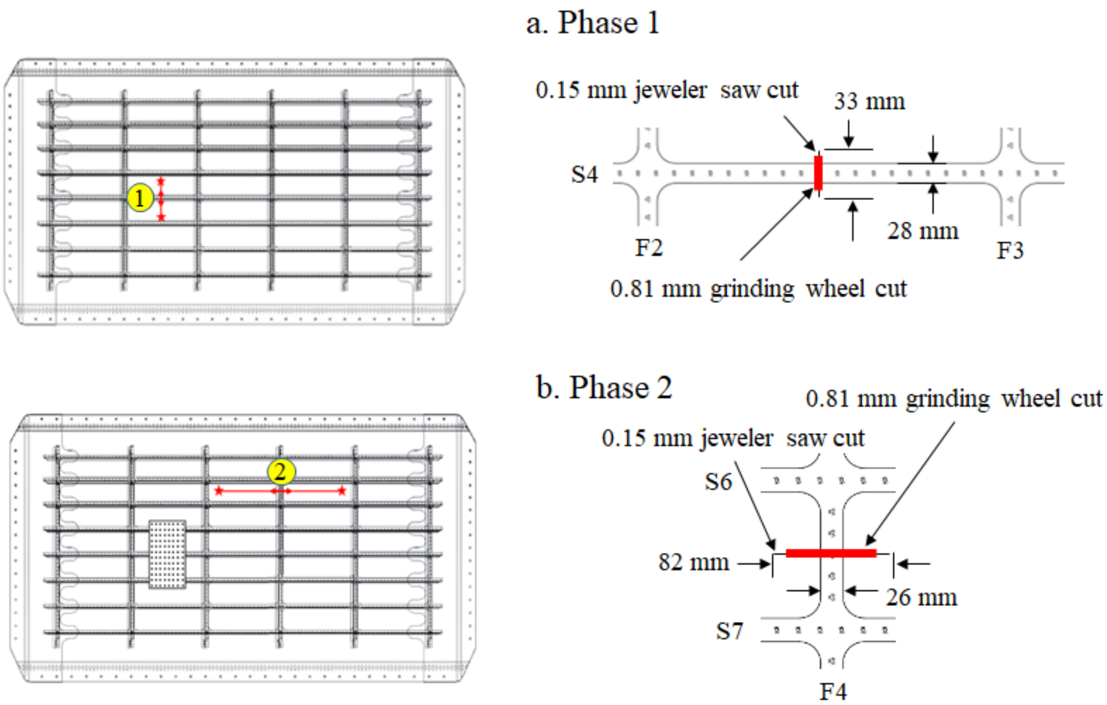


Figure 4: Initial damage scenarios used in the two test phases.

The loads on the crown of the fuselage forward of the wing are primarily due to pressure and bending from flight and landing loads. An elevated fuselage pressure of 68.3 kPa (9.9 psi) was assumed as the potential operational condition for an aircraft equipped with EMST. It was shown by Steadman [14], and assumed in this study, that flight loads measured in typical single-aisle aircraft, such as the Boeing 727 and Boeing 737, resemble the mini-TWIST spectrum [15] if the acceleration excursions are reduced by a factor of 2.0. In a hierarchical finite element approach [11], hoop and axial stresses applied to a panel located on the crown of an aircraft were determined, as shown in Figure a. Hoop stresses are assumed to be due to pressurize cycles only, and axial stresses are assumed to be due to pressure, flight, and landing loads, as shown schematically in Figure b.

For spectrum loading of the skin, the flight axial stresses, S_F , and the landing axial stress, S_L ,are given by:

$$S_F = S_{Press} + (1+\Delta n)S_{1g} \text{ and } S_L = \alpha S_{1g} \qquad (1)$$

respectively, where S_{Press} is the axial stress due to pressure, S_{1g} is the 1G axial bending stress for the panel location in the crown, Δn is the sequence of flight excursion load factors representing the 50% Mini-TWIST spectrum, and α is a landing stress parameter which was limited to be α = -0.6 to prevent specimen buckling. Finite element analyses were conducted to calculate the skin pressure and bending stresses in the panel [11].

Though the FASTER fixture is capable of executing complex variable amplitude spectrum loading that represents fuselage down-bending loads, it is not practical to run a full-scale fatigue test program under such conditions. Instead, an equivalency approach was used to determine a constant amplitude load applied in the axial direction, designated by S_{eq},, used in the panel test as shown in Figure c. This equivalent constant amplitude loads would provide an equivalent crack growth as the complex flight loads, as shown in Figure d. This was done experimentally using M(T) spectrum loaded and constant amplitude specimens [12].

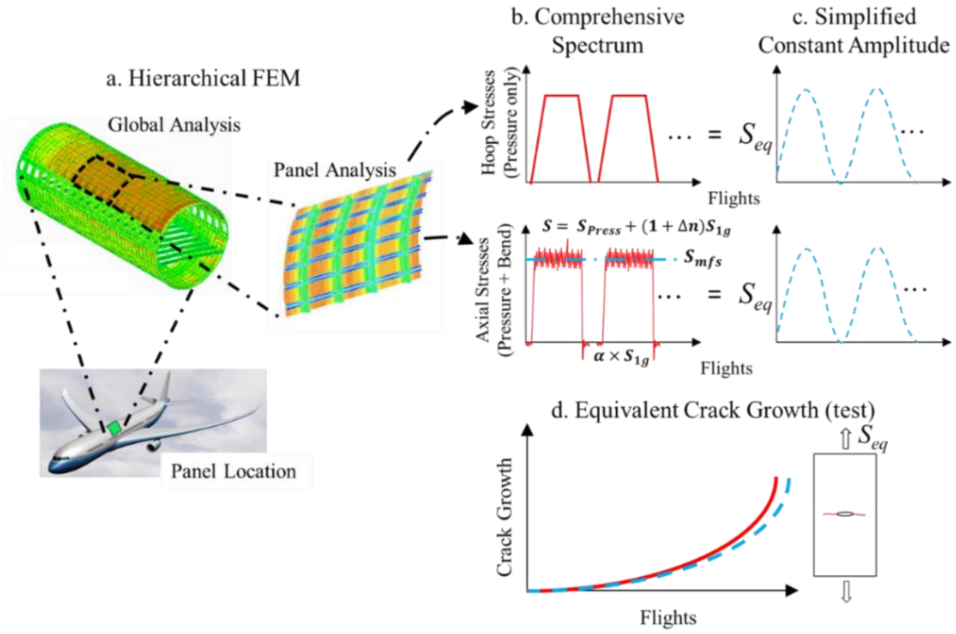


Figure 5: Determination of panel hoop and axial stresses and conversion to equivalent constant amplitude loads used in full-scale test.

The original goal of this program was to conduct different fuselage panel tests under the same loading conditions at a key location in the crown of a generic, single aisle commercial aircraft, where the loads are primarily due to pressure and bending from flight and landing loads. However, the Panel 2 and Panel 3 skin was approximately 18% and 10% thinner than Panel 1skin respectively, due to small manufacturing tolerance difference in the chem-mill pocket thickness. This would have resulted in a significant difference in crack growth life using the same load conditions based on the stress intensity factor solution differences alone. The skin thickness difference in the as-built Panel 2 and Panel 3 were accounted for by testing these panels to approximately the same stress intensity factors as baseline Panel 1 based on finite element analysis (FEA), as shown in Figure . Using this approach, an apples-to-apples comparison of damage tolerance performance can be made between the different panels. The applied actuator loads (hoop, axial, and frame loaders) used for each panel, as shown in Figure 7, yielding similar stress intensity factors. Strain surveys were conducted at 75% of the fatigue loads. Fatigue loading was conducted using R=0.05 and a frequency of 0.03 Hz. All testing was done under lab ambient conditions.

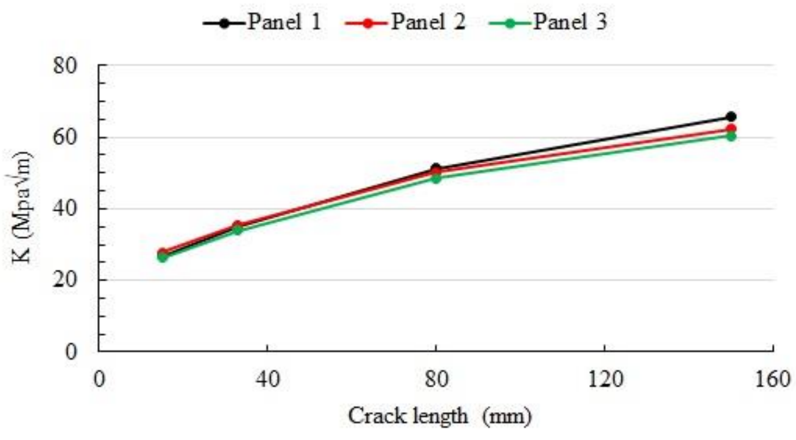


Figure 6: Stress intensity factor (Ktotal) comparison between panels.

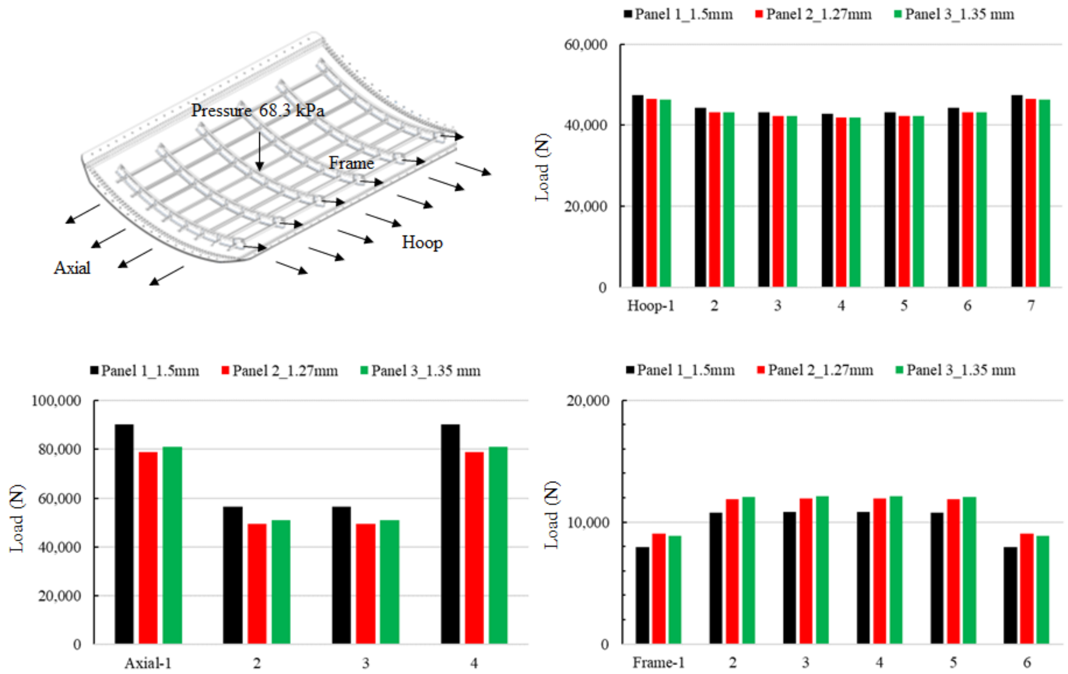


Figure 7: Test loads comparison between panels yielding similar stress intensity factors.

RESULTS AND DISCUSSION

Tests and analyses were performed to determine the fatigue and damage-tolerance performance of the first three panels in this program (see Table 1 for panel description). Representative results focus on the Phase 1 circumferential crack test for each of the three panels.

Initial strain surveys verified proper load introduction to the panels and correlated with the FEA results. Representative results shown in Figure 8 reveal that axial strains, measured using both strain gages and DIC near the original notch-tip areas, were in good agreement with FEA. The panels were then fatigue tested under the constant amplitude equivalent load which simulated flight load conditions, during which the skin crack extended across two stringer bays to a final length of approximately 287 mm as shown in Figure 9: Panel 2 Phase 1 results indicate transition points of fracture surface morphology where FCG rates change. In general, slow and stable crack growth was observed during fatigue for all three panels. The crack surface morphology changes from V (valley) to S (slant) were observed in all three panels, and the morphology changes associated with distinct transition points. Preliminary results indicate that crack-growth rates changed at these transition points similar to that observed in coupon tests conducted on M(T) specimens [12]. The cracks extended to a final length of about 287 mm after 33,600, 48,850 and 46,900 cycles in Panels 1, 2 and 3, respectively. Panels 2 and 3 exhibited better fatigue crack growth performance compared to Panel 1, as shown in Figure 10.

Afterwards, the panels were subjected to approximately 2.5G axial load in a limit load test holding the pressure constant at 68.3 kPa. Strain gages located on the outboard flange of the stringer ahead of the fatigue crack revealed that the stresses were below the yield strength of the stringer material, as shown in Figure 11. The intact stringers ahead of the crack were effective in containing the damage. In addition, limited stable tearing and crack-tip plasticity was observed from each crack tip, as shown in Figure 9 and in Figure 12 using DIC measurements.

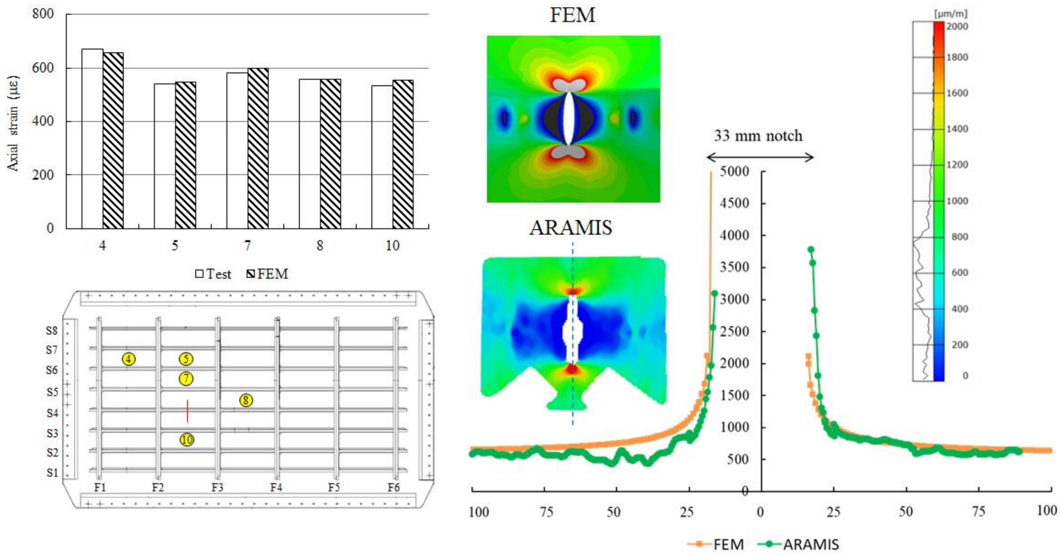


Figure 8: Panel 2 phase 1 strain survey results verify FEA and applied loads.

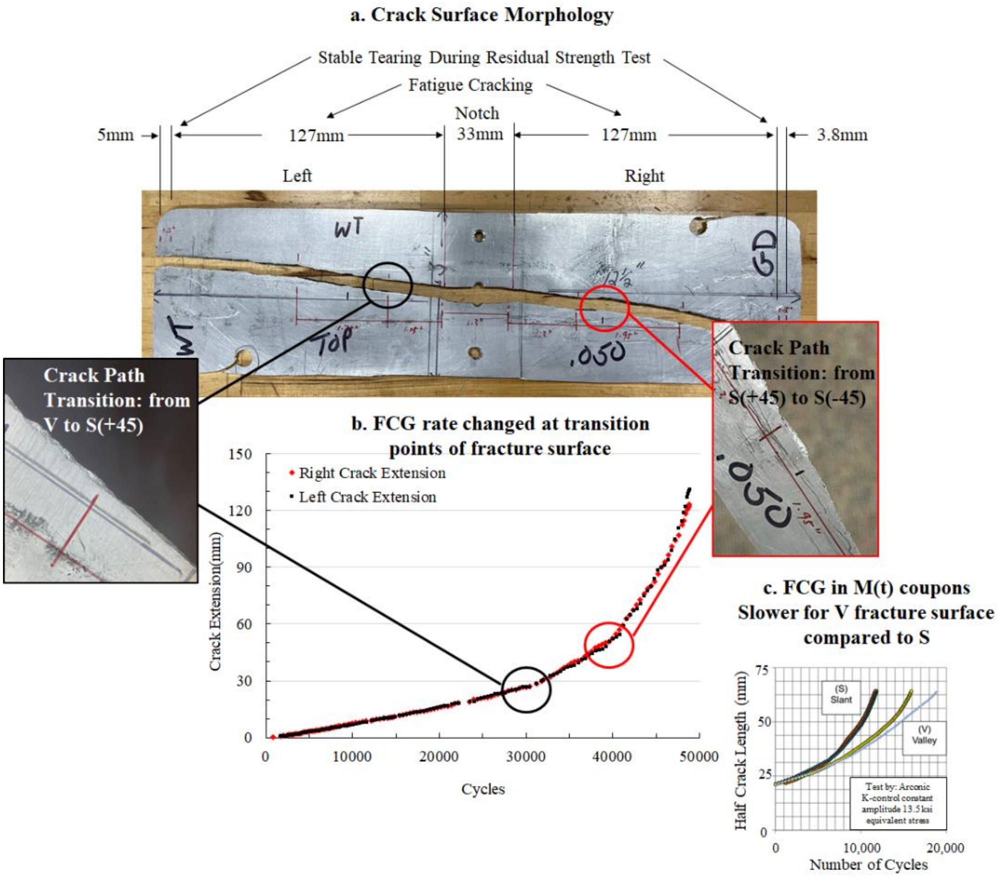


Figure 9: Panel 2 Phase 1 results indicate transition points of fracture surface morphology where FCG rates change.

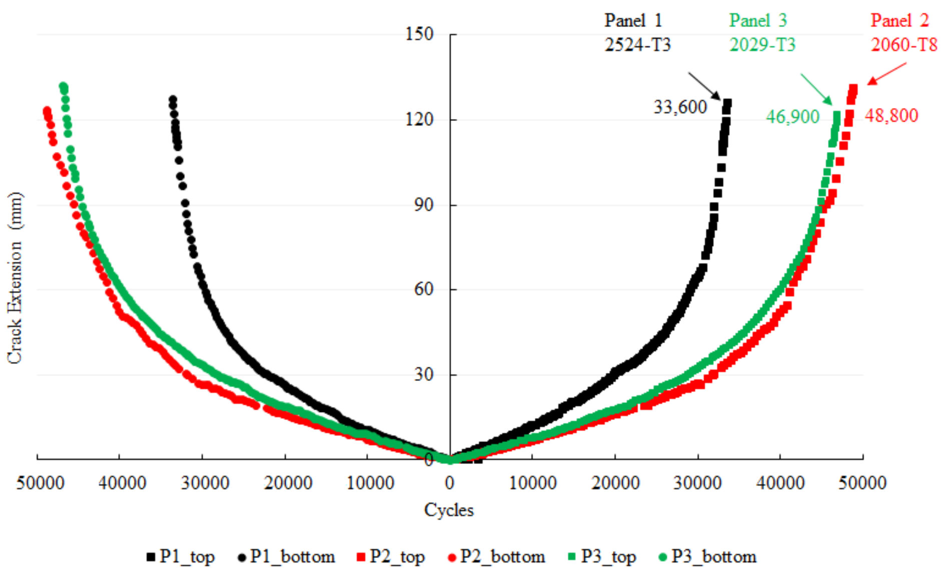


Figure 10. Circumferential crack growth comparison between Panels 1-3.

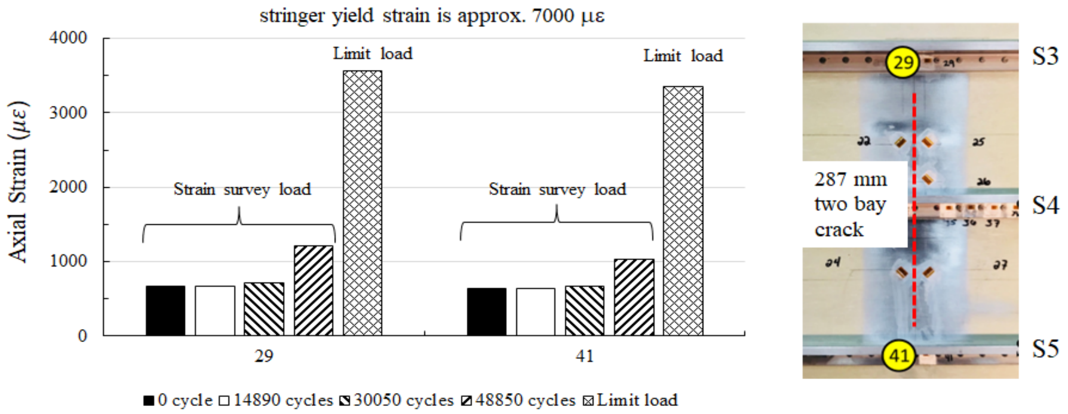


Figure 11. Panel 2 stringer strain during the limit load test.

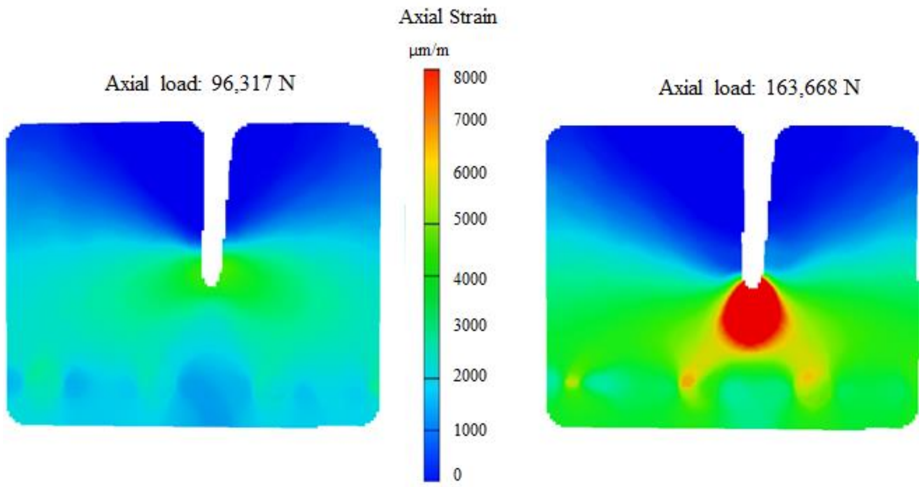


Figure 12. Panel 2 DIC result during the circumferential crack limit load test.

SUMMARY

In a collaborative effort, the FAA, Arconic, and Embraer are assessing emerging metallic structures technologies (EMST) for fuselage applications through full-scale test and analysis. Several technologies are being considered, including advanced aluminum-lithium alloys and selective reinforcement using fiber metal laminate (FML). Data from this study will be used to verify that EMST provides an improvement in damage tolerance performance and structural safety as compared to the current fuselage structure constructed with conventional materials and fabrication processes, and to enhance regulatory guidance on EMST. Recent efforts focused on baseline Panel 1 constructed using conventional materials and Panels 2 and 3 constructed with advanced Al-Li alloys and next generation clad alloy. Results and other major findings include:

- Phase 1: A two-bay circumferential skin crack having a total length of 33 mm was inserted with the central stringer severed. The panel was then subjected to fatigue cycles under loads developed by matching the stress intensity factors, during which slow and stable crack growth occurred to a final length of 287 mm.
- The change of crack surface morphology from V (valley) to S (slant) was observed on all three panels. In general, the crack surfaces start in V shape and switch to S at longer crack length, and crack morphology changes associated with crack-growth rate changes.
- Panels 2 and 3 exhibited approximately 40% improvement in fatigue life compared to baseline panel 1 for circumferential damage scenario.

Future efforts include the completion of the remaining phase 2 efforts for Panel 3, and the assessment of Panels 4 and 5 with an advanced damage containment design concept incorporating FML reinforcement.

REFERENCES

[1] Prasad, N., Gokhale, A., Wanhill, R. (Eds.) (2014). “Aluminum-Lithium Alloys, Processing, Properties and Applications,” *Elsevier*.

[2] Stonaker, K., Bakuckas, Won, I., and Freisthler, M. (2015). “Material Characterization of Aluminum Lithium Alloys Used in Aerospace Applications,” *Proceedings of the 28th ICAF Symposium*, June 3–5, 2015, Helsinki, Finland.

[3] Bertoni, M., Fernandez, F., Miyazaki, M. (2014). “Fuselage Technology Demonstrator,” *Proceedings of AeroMat 2014*, June 16–19, 2014, Orlando, FL.

[4] Chaves, C. (2017). “A Review of Aeronautical Fatigue Investigations in Brazil,” *Proceedings of the 35th ICAF Conference*, June 5–6, 2017, Nagoya, Japan.

[5] Kok, L., Poston, K., Moore, G. (2011). “Bombardiar Aerspace FSW Demonstrator,” *Proceedings of the 26th ICAF Symposium*, June 1–3, 2011, Montreal, Canada.

[6] Schmidt, H. (2005). “Damage Tolerance Technology For Current and Future Aircraft Structure,” *Proceedings of the 23rd ICAF Symposium*, June 8–10, 2005, Hamburg, Germany.

[7] Beumler, T. (2014). “Development of Thin-Walled FML-Structures,” *Proceedings of AeroMat 2014*, June 16–19, 2014, Orlando, FL.

[8] Heinimann, M., Kulak, M., Bucci, R., James, M., Wilson, G., Brockenbrough, J., Zonker, H., and Sklyut, H. (2007). “Validations of Advanced Metallic Hyrid Concepts with Improved Damage Tolerance Capabilities for Next Generation Lower Wing and Fuselage Applications,” *Proceedings of the 24th ICAF Symposium*, May 16–18, 2007, Naples, Italy.

[9] Silva, D., Cruz, M., Mendonca, W., Brandao, F., Sakata, A., Silva, G., and Kulak, M. (2017). "Manufacturing of a Fiber Metal Laminate Lower Wing Cover Demonstrator," *Proceedings of AeroMat 2017*, April 10–12, 2017, Charleston, SC.

[10] Tian, Y., and Bakuckas, J. (2019). "Full-Scale Aircraft Structural Test Evaluation and Research (FASTER) Fixture—Capabilities Description and User Manual," *DOT/FAA/TC-TN19/6*.

[11] Kulak, M., Bakuckas, J. G., Tian, Y., Stanley, D. (2023). "Design, Analysis and Test Development of Full-Scale Fuselage Test Panels to Assess Emerging Metallic Structures Technologies," [Unpublished manuscript], *DOT/FAA/TC-TNxx/xx*.

[12] Stonaker, K., Bakuckas, J., Stanley, D., Kulak, M., Chang, P., and Freisthler, M. (2019). "Assessment of Fatigue Behavior of Advanced Aluminum Alloys Under Complex Variable Amplitude Loading," *Proceedings of the 30th ICAF Symposium*, June 5–7, 2019, Krakow, Poland.

[13] Tian, Y., Stanley, D. and Bakuckas, J. G., (2021). Assessment of Emerging Metallic Structures Technologies through Test and Analysis of Fuselage Structure: Test Panel 1. DOT/FAA/TC-21/25.

[14] Steadman, D. (2007), "Destructive Evaluation and Extended Fatigue Testing of Retired Transport Aircraft, Volume 5: Data Analysis Report," *DOT/FAA/AR-07/22,V5*.

[15] Lowak, H., de Jonge, J., Franz, T., and Schutz, D. (1979) "MINITWIST—A shortened version of TWIST," NLR *Report No. TB-146.*