Research Article

Assessment of emerging metallic structures technologies through test and analysis of fuselage structure

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Received: 9 August 2019 / Accepted: 4 October 2019 / Published online: 31 October 2019

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 facility. In this collaborative effort, full-scale fuselage panel test data will be obtained to assess the effect of EMST fuselage concepts on damage tolerance performance as compared to the current baseline aluminum fuselage structures located on the crown of a typical single-aisle aircraft forward of the wing. Several technologies will be considered in the scope of the project, including advanced aluminum–lithium alloys, selective reinforcement using fiber metal laminates, and advanced joining processes, such as friction stir welding. Data from this study will be used to verify improved weight and structural safety performance of EMST and to assess the adequacy of existing airworthiness standards and guidance needed for the implementation of arising technologies and their impact on future designs. Results from the first baseline panel test are presented in this paper and will be compared to future tests on advanced panels containing varying EMST to assess the damage-tolerance performance.

Keywords Advanced technologies · Damage tolerance · Crack growth · Residual strength · Aircraft fuselage structure

1 Introduction

The aircraft industry is striving to both improve 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 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 is needed to fully vet and properly assess a new technology and understand potential certification and continued airworthiness issues. This includes the assessment for continued relevance of existing regulations and potential development of additional safety standards and regulatory guidance, if needed, with the end goal of maintaining or enhancing the current level of safety afforded by the existing airworthiness standards.

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SN Applied Sciences (2019) 1:1521 | https://doi.org/10.1007/s42452-019-1471-7

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 compared with the current baseline aluminum fuselage located on the crown of a typical single-aisle aircraft forward of the wing spar. Several EMST are being considered, including integral frames, friction stir welded skin joints, new metallic alloys, bonded stringers, and selective reinforcement using fiber metal laminates. Several panels with various EMST are planned to be tested using the FAA's Full-Scale Aircraft Structural Test Evaluation and Research (FASTER) facility designed for testing fuselage panels capable of simulating aircraft service load conditions through synchronous application of mechanical and environmental load conditions [10], Fig. 1.

A phased approach is being undertaken to study three damage scenarios: (1) a two-bay skin crack, along the hoop direction, with central stringer severed; (2) a mill-line crack parallel to stringer, located near the edge of the milled section of a skin bay, and; (3) a two-bay skin crack, along the axial 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 load history of a fuselage panel located on the crown of the aircraft, forward of the wing [11, 12]. To demonstrate potential improvements in operational usage when considering aircraft equipped with EMST, an elevated fuselage pressure differential was used in the load sequence, which is approximately 15% higher than that used in a typical single-aisle transport category aircraft, such as the B737 and A320. The final stage of testing will be a residual strength test to limit load conditions. Data from this program will be used to demonstrate the improvement in damage tolerance and structural safety potential of EMST and to assess the adequacy of existing regulations when considering EMST.

Recent initial efforts have focused on the first baseline panel consisting of 2524-T3 skin and conventional 7000-series aluminum substructure assembled through riveting. The baseline panel was subjected to three phases of testing and accumulated over 84,000 simulated flights over a 10-month period. 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 the baseline panel test are summarized below and will be compared with advanced panels containing varying EMST to assess the damage tolerance performance:

- *Phase 1* A two-bay hoop skin crack having a total length (tip-to-tip) of 33 mm was inserted with the central stringer severed. The panel was then fatigue tested under simulated flight load conditions for 33,600 cycles in which slow and stable crack growth occurred to a final total length of 287 mm. During the subsequent residual strength test conducted up to a 2.5G axial limit load, local stable tearing extension occurred from each crack-tip.

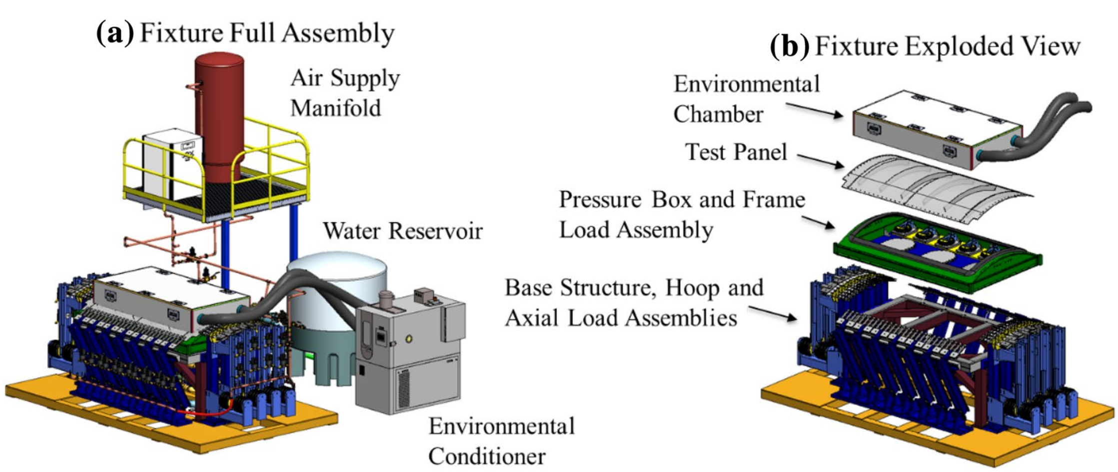


Fig. 1 FAA FASTER fixture assembly and major components

- *Phase 2* A mill-line crack having a total length of 152 mm was inserted in a skin mid-bay parallel to a stringer and then subjected to 7500 fatigue cycles. The crack extended approximately 50 mm from each notch-tip and displayed intermittent periods of slow/no crack growth because of crack binding.
- *Phase 3* A two-bay axial skin crack having a total length of 38 mm was inserted with the central frame severed and then fatigue tested to 43,600 cycles. During fatigue, the crack extended across two frame bays to a final length of 406 mm. Afterwards, a residual strength test was conducted during which the panel failed at an applied pressure of 117 kPa. Approximately 26 mm of stable tearing was observed from each crack tip prior to failure of the panel.

2 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.

2.1 Target application and panel description

In this study, the target aircraft considered is a typical single-aisle airplane, such as the B737 of A320. The location of the fuselage panel is assumed to be the crown 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 the baseline panel using standard aerospace manufacturing practices, including but not limited to forming, chemical milling, surface treatment, and joining technologies. The final panel dimensions were 3175 mm by 1854 mm with a radius of 1880 mm, as shown in Fig. 2. The skin material was 2524-T3 with a pocketed construction where the skin thickness was 1.4 mm in the mid-bay regions and 1.6 mm in the pad-up regions under the frames and stringers. The substructure included eight stringers made from 7150-T77511 extruded in a Z-section with a 178-mm spacing and six 7075-T62 frames connected using 7075-T62 shear ties with a 508-mm spacing. A two-piece floating Z-section frame and L-section shear tie construction was used. 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. Loads were also introduced into each frame.

2.2 Test phases and damage scenarios

A phased approach was undertaken to study the three damage scenarios summarized as follows.

Phase 1 Initial damage consisted of a two-bay hoop skin crack having a total length of 33 mm between frames F2 and F3, with the central stringer S4 severed (see Fig. 3a).

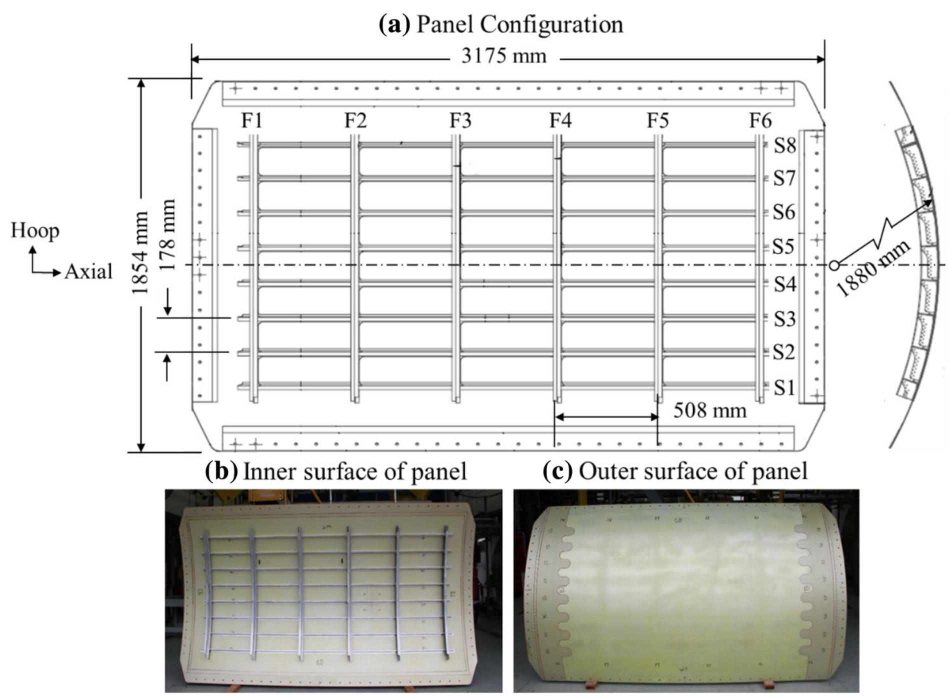


Fig. 2 Baseline panel configuration and views of the internal and external surface

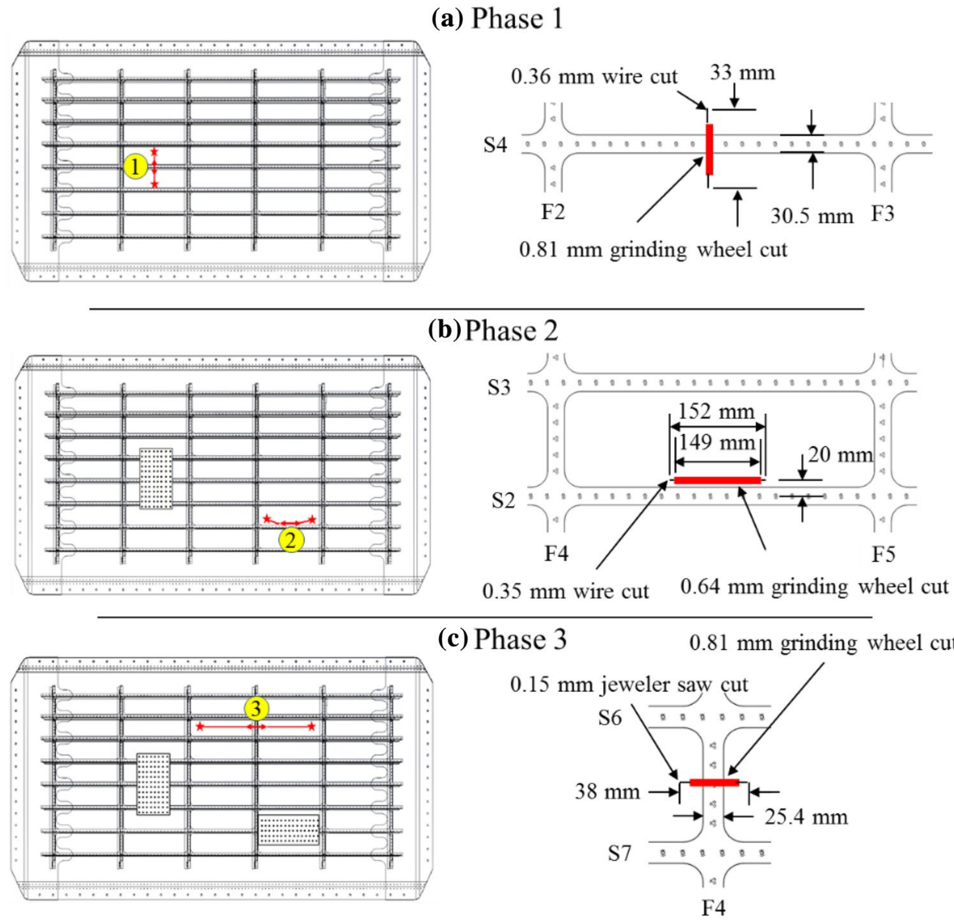


Fig. 3 Initial damage scenarios used in the three test phases

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 a 2.5G axial load while holding the pressure constant under operational conditions. The panel was then repaired for follow-on phases.

Phase 2 To study crack turning phenomena, the initial damage consisted of a mill-line crack having a total length of 152 mm inserted in the skin parallel to stringer S2 midway between frames F4 and F5, as shown in Fig. 3b. 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 flight-load conditions to monitor the direction and rate of crack growth. Afterwards, the panel was repaired for follow-on phases.

Phase 3 Initial damage consisted of a two-bay axial skin crack having a total length of 38 mm between stringers S6 and S7, with the central frame/shear tie F4 severed, as shown in Fig. 3c. Strain surveys were conducted to ensure proper load introduction and to verify no effects from the repair made in Phases 1 and 2. The panel was then fatigue tested, simulating pressure-only operational conditions until the crack extended to a final length of approximately 406 mm. A residual strength test was finally conducted to failure measuring the load-carrying capacity of the panel.

2.3 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 baseline panel was instrumented with over 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.

2.4 Applied mechanical loads

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 [13], and assumed in this study, that flight loads measured in typical single-aisle aircraft, such as the B727 and B737, resemble the mini-TWIST spectrum [14] 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 Fig. 4a. 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 Fig. 4b.

The axial stresses during flight and landing are given by:

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

respectively, where S_{Press} is the stress due to pressure, S_{1g} is the bending stress under 1-g, S_{mfs} is the mean flight stress, Δn is the incremental load factor from the 50% Mini-TWIST spectrum, and $\alpha = -0.6$ is the landing stress parameter. Finite element analyses were conducted to calculate the pressure and bending stresses [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, donated by S_{eq}, used in the panel test as shown in Fig. 4c. This equivalent constant amplitude loads would provide

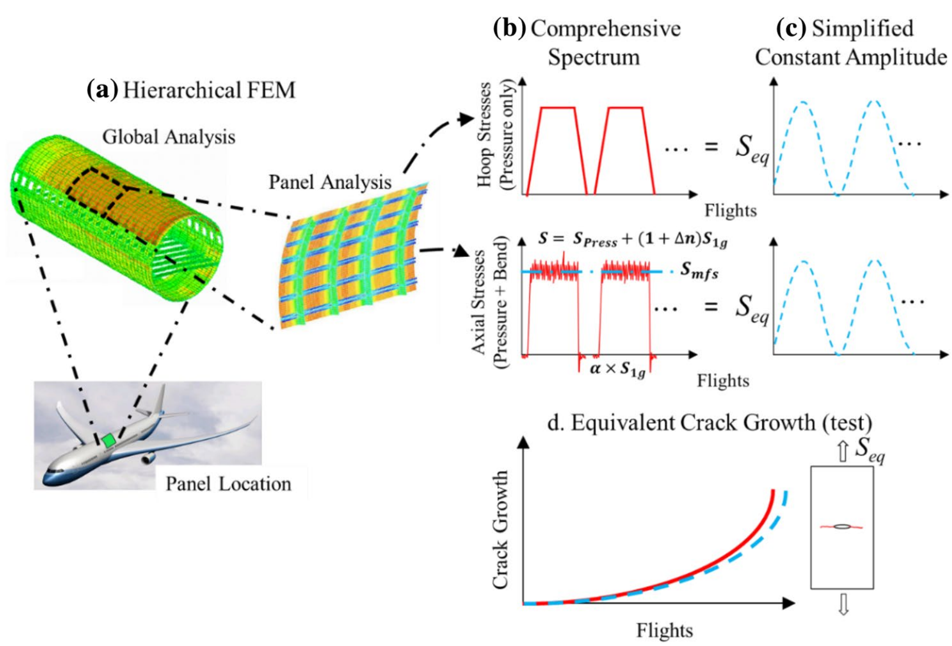


Fig. 4 Determination of panel hoop and axial stresses and conversion to equivalent constant amplitude loads used in full-scale test

Table 1 Summary of applied loads

Phase	Test type	Pressure (kPa)	Axial load			
			S_{Eq} (MPa)	S_{Press} (MPa)	S_{1g} (MPa)	Δn
1	Strain survey	51.2	67.2	28.8	36.9	0.04
	Fatigue	68.3	89.6	38.4	49.2	0.04
	Limit load	68.3	158.6	38.4	49.2	0.04
2	Strain survey	51.2	28.8	28.8	0	0
	Fatigue	68.3	38.4	38.4	0	0
3	Strain survey	51.2	28.8	28.8	0	0
	Fatigue	68.3	38.4	38.4	0	0
	Residual strength	117.2	65.9	65.9	0	0

an equivalent crack growth as the complex flight loads, as shown in Fig. 4d. This was done experimentally using M(T) specimens, as described by Stonaker et al. [14].

The applied loads used in each test phase are shown in Table 1. 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.

3 Analysis

Finite element analysis (FEA) for this program was conducted by Arconic. As shown in Fig. 5, a hierarchical approach using global and panel models provided:

- Actuator loads for the FASTER fixture to provide stresses in the test section that match stresses in the global model of the idealized fuselage.
- Pre-test predictions of the stress and strain fields.
- Stress-intensity factors used in the equivalent constant amplitude stress testing, fatigue crack growth analysis, and residual strength calculations.
- Other fracture parameters, including δ_5 for comparison of stable tearing measurements.

A details of the analysis approaches used in this program are provided by Kulak et al. [11].

4 Results and discussion

Tests and analyses were performed to determine the fatigue and damage-tolerance performance of the baseline fuselage panel, which was constructed using conventional materials and fabrications processes. Comparisons will be made to advanced fuselage panels with varying EMST conducted in future test. Representative results focus on the baseline panel test for each of the three phases.

4.1 Phase 1: Two-bay hoop crack with central stringer severed

Initial strain surveys verified proper load introduction to the baseline panel and validated the FEA model. Representative results shown in Fig. 6 reveal that axial strains measured at gages near the original notch-tip were in

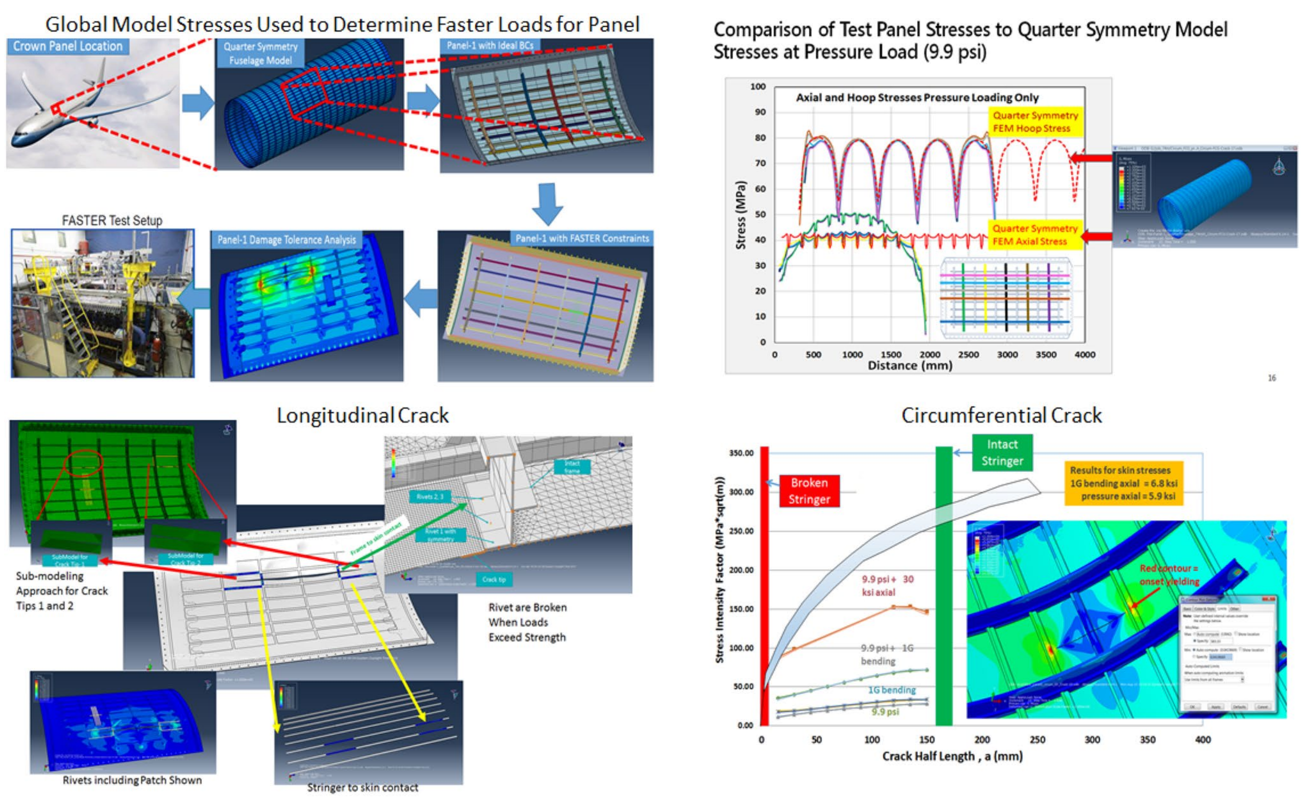


Fig. 5 Hierarchical FEA approach used in program

Fig. 6 Phase 1 strain survey results verify FEA and applied loads

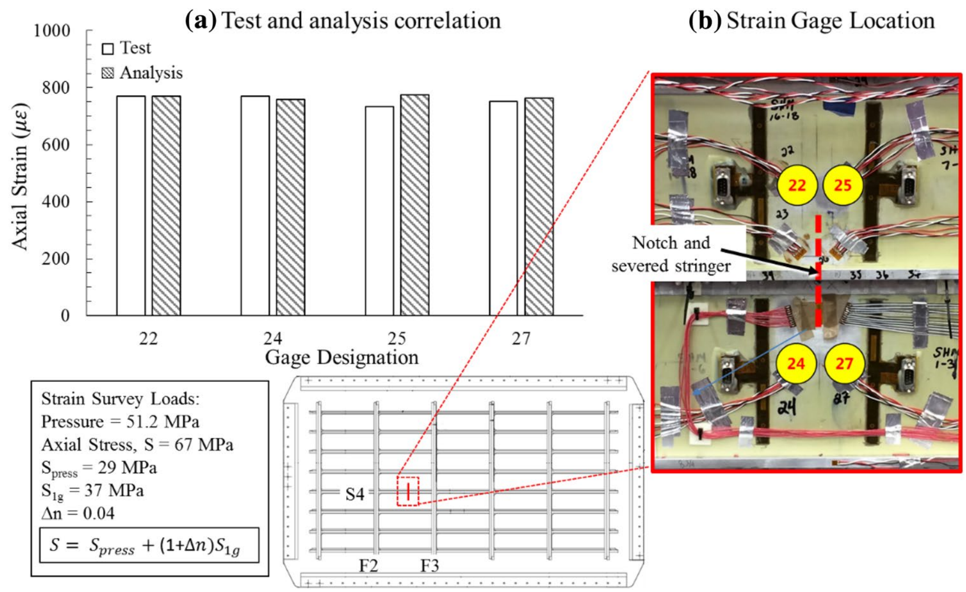
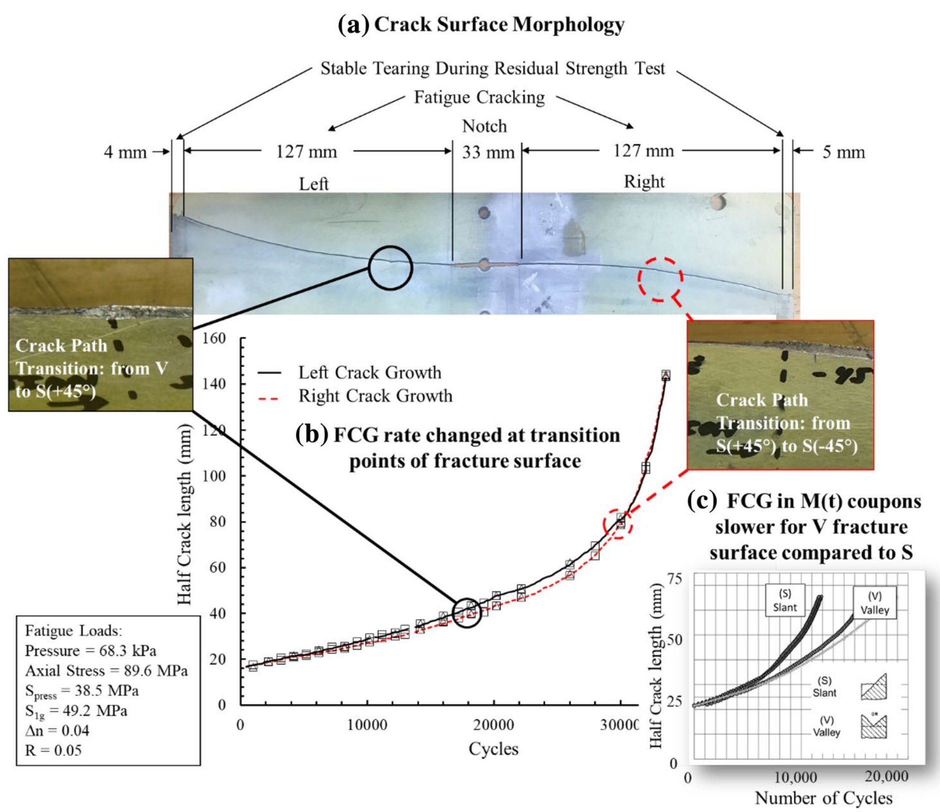


Fig. 7 Phase 1 results indicate transition points of fracture surface morphology where FCG rates change



good agreement with FEA. The panel was then fatigue tested under simulated flight load conditions for 33,600 cycles, during which the skin crack extended across two stringer bays to a final length of approximately 287 mm, as shown in Fig. 7. In general, slow and stable crack growth was observed during fatigue. The crack surface morphology had distinct transition points where, on the left side, the surfaces changed from V (valley) to S (slant) fracture and on the right side transitioned from a + 45° to – 45° slant fracture. Preliminary results indicate that crack-growth rates changed at these transition points similar to that observed in coupon tests conducted on M(t) specimens [14]. Afterwards, the panel was subjected to a 2.5G axial load in a limit load test holding the pressure constant at 68.3 kPa. Limited stable tearing extension was observed from each crack tip. The panel was then repaired for follow-on phases.

4.2 Phase 2: Mid-bay mill-line crack parallel to stringer

In this phase, the panel was fatigue tested under pressure load conditions for 7500 cycles, in which the crack extended approximately 50-mm from each notch-tip. Representative results are shown in Figs. 8 and 9. In general, natural cracks developed at 18° and 15° angles from the left and right notch-tips, respectively, which agreed

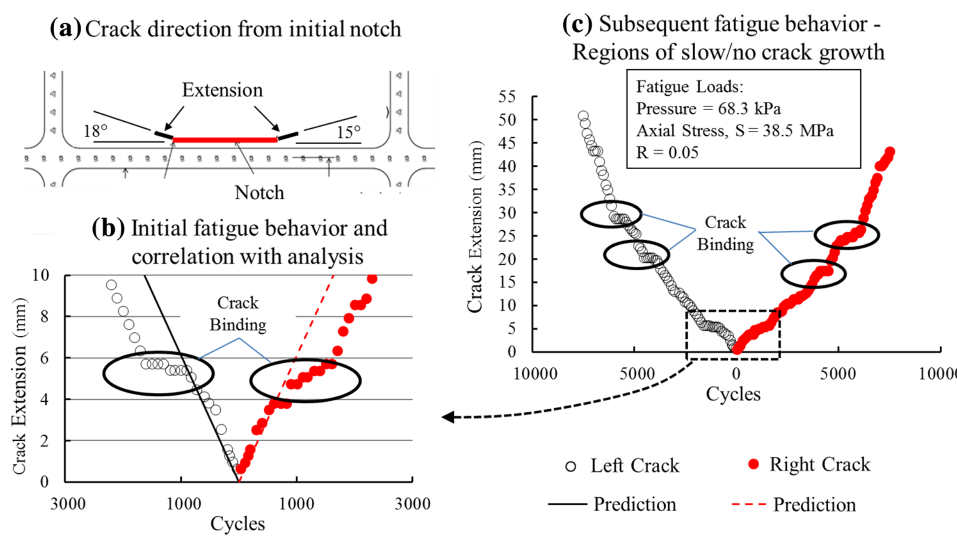


Fig. 8 Phase 2 FCG results revealed regions of slow/no growth due to crack binding. Good agreement with analysis at initial stages of fatigue

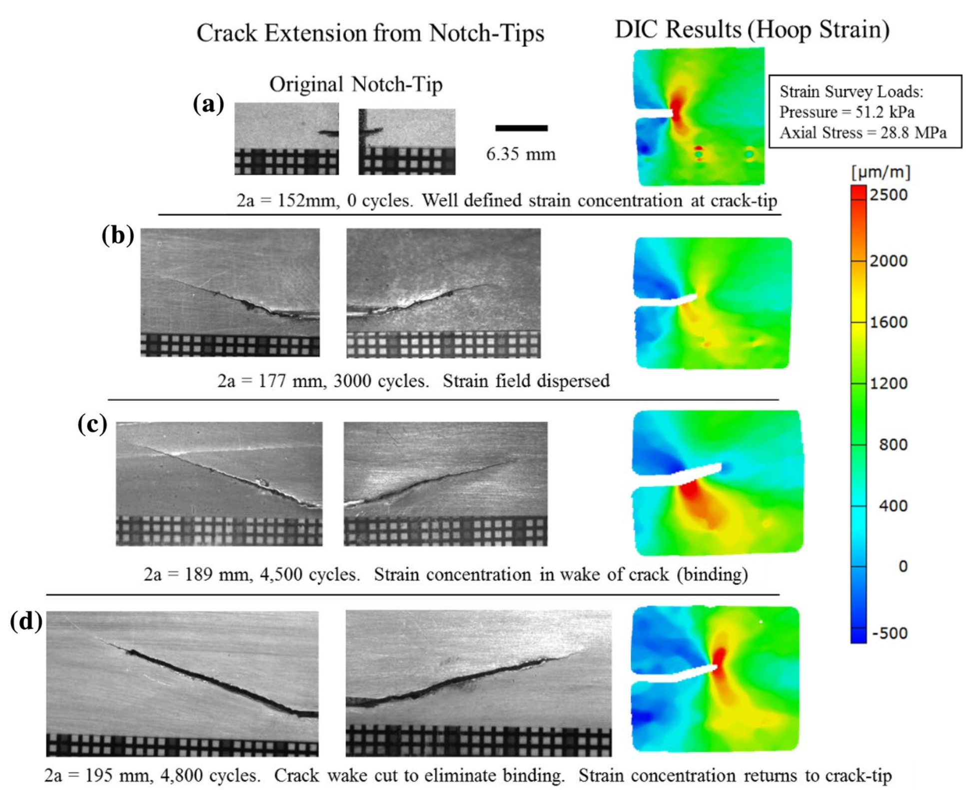


Fig. 9 Phase 2 FCG results reveal high strains in the crack wake due to binding

with FEA predictions, as shown in Fig. 8a. The initial crack growth increased steadily and correlated with analysis up to approximately 1000 cycles where crack extension reduced precipitously, Fig. 8b. Subsequently, the fatigue behavior displayed additional slow/no growth intervals at 4500 and 6000 cycles, at which the crack wake surfaces were notched with 0.35-mm diamond wire, leaving the natural crack-tip (see Fig. 8c). Results from DIC revealed crack binding where the highest tensile strains were measured in the wake of the crack during slow/no crack growth interval, as shown in Fig. 9c. After notching the crack wake, the high tensile strain region transitioned back to the crack-tip (see Fig. 9d). Upon completing the fatigue test, the panel was repaired for the final phase of testing.

4.3 Phase 3: Two-bay axial crack with central frame severed

For the final phase, the panel was fatigue tested under pressure load conditions for 43,600 cycles, during which the skin crack extended across two frame bays to a final total length of approximately 406 mm. Representative results are shown in Fig. 10. The crack growth was quite slow in the initial stages of fatigue from the initial total notch length of 38 mm. Consequently, the notch was extended twice because of unexpected slow crack growth to lengths of 51 mm and 83 mm after 6000 cycles and 12,500 cycles, respectively. Local effects from the severed frame end suppress crack growth and cause binding for the shorter notch lengths (less than 83 mm).

DIC results shown in Fig. 10a revealed dispersed and asymmetric crack-tip strains for the shorter cracks which transitioned to the classical kidney bean strain field as the crack-tip extended from the central severed frame. Subsequent to the second notch inserted at 12,500 cycles, the fatigue crack growth was symmetric, stable and continuous as shown in Fig. 10b. After approximately 41,500 cycles, rapid, but stable, crack extension was observed.

A residual strength test was then conducted under pressurize loading applied quasi-statically. Representative results are shown in Fig. 11. Both the crack extension and fracture parameter, δ_5, were measured as a function of applied pressure (see

Figure 11a). As shown, initial crack extension was measured at an applied pressure of 75 kPa. Approximately 26 mm of stable tearing was observed from each crack tip prior to reaching the maximum applied

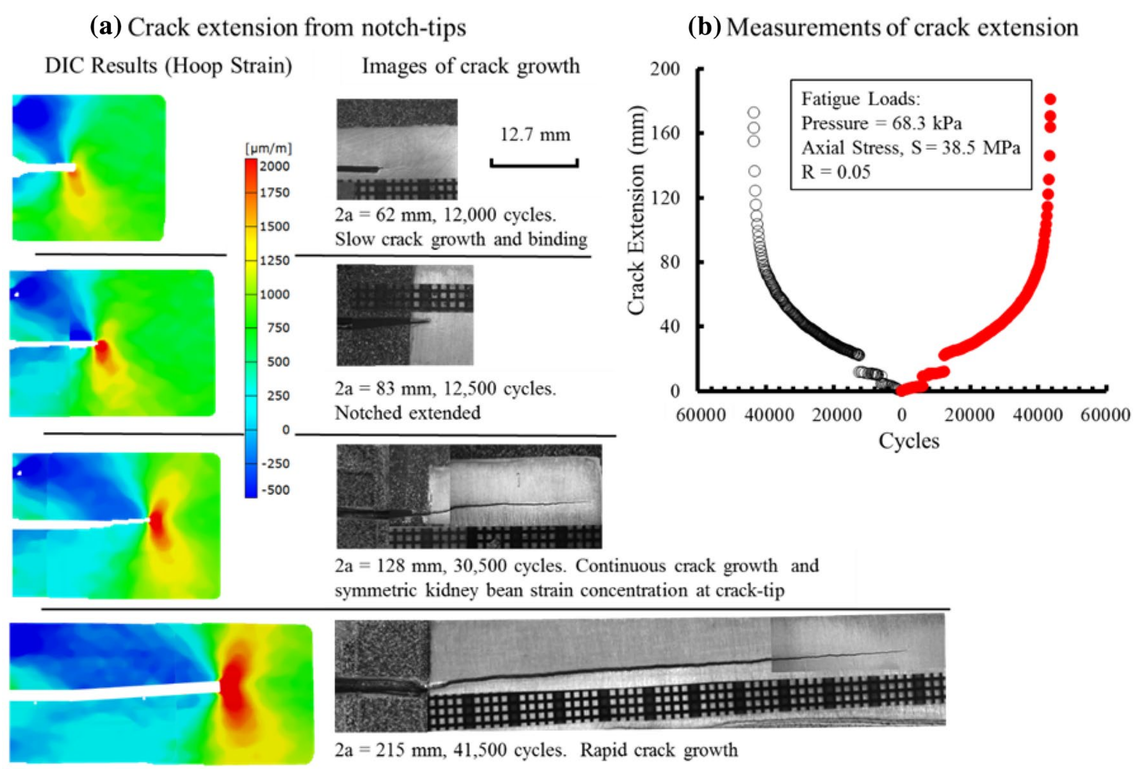


Fig. 10 Phase 3 results show slow FCG for short crack lengths due to local effects of severed frame and binding. Continuous FCG occurred for longer crack lengths

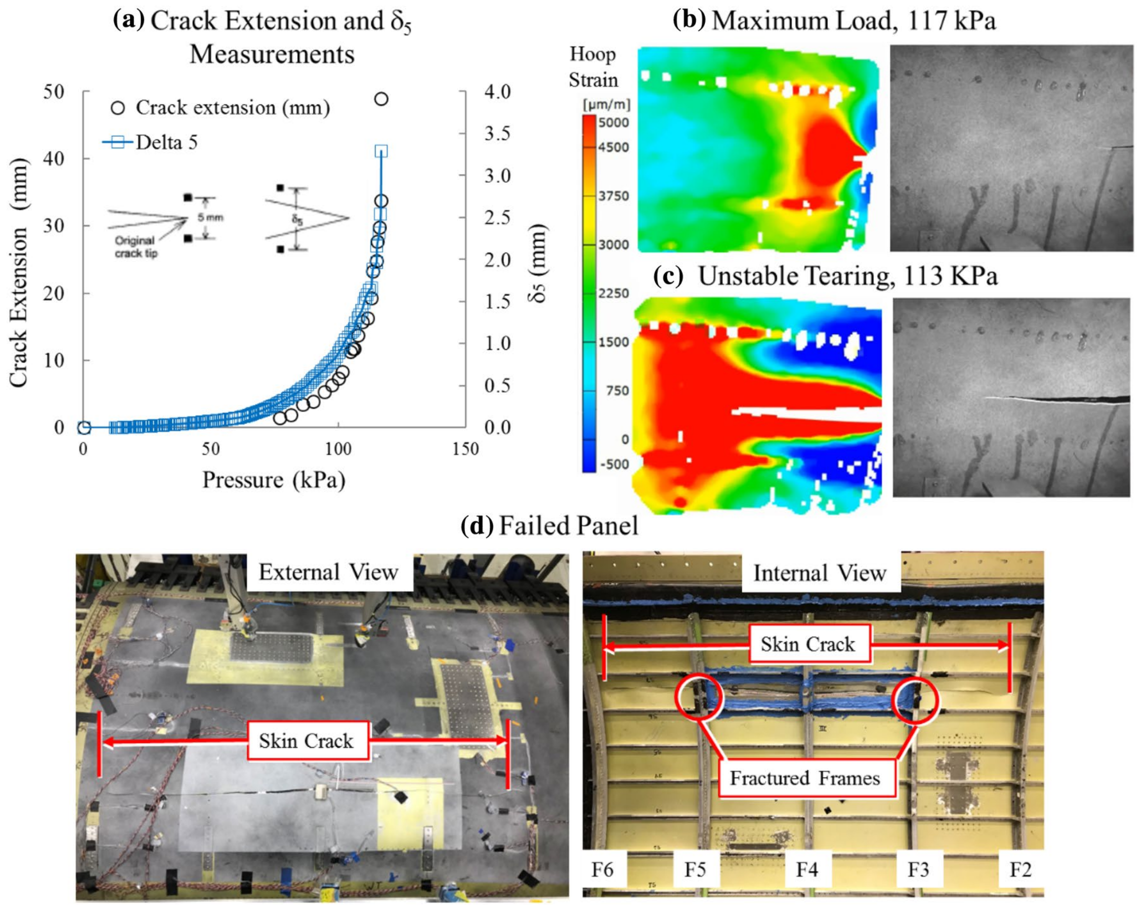


Fig. 11 Phase 3 residual strength test showing measurements of crack extension and δ_5, progressive tearing, and final state of failure of the panel

pressure of 117 kPa, as shown in Fig. 11b. Unstable tearing then occurred, resulting in failure of the panel (see Fig. 11c). Extensive damage occurred to the panel where the skin crack extended to a total length of 1955 mm and severed two intact frames, as shown in Fig. 11d. The pressure at failure exceeded the residual strength damage tolerance requirements in Title 14 Code of Federal Regulations 25.571.

5 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, selective reinforcement using fiber metal laminates, and advanced joining processes, such as friction stir welding. Data from this study will be used to verify potential improved damage tolerance performance that EMST offer compared to the current fuselage structure constructed with conventional materials and fabrication processes. In addition, unique damage mechanisms and damage-tolerance behavior associated with EMST will be identified to assess the relevance of existing regulations and to inform whether additional safety standards and regulatory guidance should be developed in implementing EMST. Recent efforts focused on the baseline panel consisting of 2524-T3 skin and 7000-series aluminum substructure assembled through riveting. The baseline panel was subjected to several phases of testing and accumulated more than 84,000 simulated flights during a 10-month period. Results from the first baseline panel test will be compared to future tests on advanced panels containing varying EMST. Results and other major findings include:

- *Phase 1* A two-bay hoop skin crack having a total length of 33 mm was inserted with the central stringer severed. The panel was then subjected to 33,600 fatigue cycles, during which slow and stable crack growth occurred to a final length of 287 mm. Afterwards, a residual strength test was conducted in which the panel was subjected to a 2.5G axial load under a constant operational pressure level. Limited stable tearing extension was observed from each crack-tip.

- *Phase 2* A mill-line crack having a total length 152 mm inserted in a skin mid-bay parallel to a stringer. During subsequent fatigue test to 7500 cycles, the crack extended approximately 50 mm from each notch-tip, displaying intermittent periods of slow/no growth due to crack binding.
- *Phase 3* A two-bay axial skin crack having a total length of 38 mm was inserted with the central frame severed. After 43,600 cycles of fatigue testing, the skin crack extended approximately to a total length of 406 mm. During the subsequent residual strength test, approximately 26 mm of stable tearing was observed from each crack tip prior to failure of the panel at 117 kPa pressure. This pressure exceeded the residual strength damage tolerance requirements defined in Title 14 Code of Federal Regulations 25.571.

Compliance with ethical standards

Conflict of interest The authors declare that they have no conflict of interest.

References

1. Prasad N, Gokhale A, Wanhill R (eds) (2014) Aluminum-lithium alloys, processing, properties and applications. Elsevier, Amsterdam
2. Stonaker K, Bakuckas Won I, Freisthler M (2015) Material characterization of aluminum lithium alloys used in aerospace applications. In: Proceedings of the 28th ICAF symposium, June 3–5, 2015, Helsinki, Finland
3. Bertoni M, Fernandez F, Miyazaki M (2014) Fuselage technology demonstrator. In: Proceedings of AeroMat 2014, June 16–19, 2014, Orlando, FL
4. Chaves C (2017) A review of aeronautical fatigue investigations in Brazil. In: Proceedings of the 35th ICAF conference, June 5–6, 2017, Nagoya, Japan
5. Kok L, Poston K, Moore G (2011) Bombardiar aerspace FSW demonstrator. In: 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. In: Proceedings of the 23rd ICAF symposium, June 8–10, 2005, Hamburg, Germany
7. Beumler T (2014) Development of thin-walled FML-structures. In: 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, Sklyut H (2007) Validations of advanced metallic hybrid concepts with improved damage tolerance capabilities for next generation lower wing and fuselage applications. In: 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, Kulak M (2017) Manufacturing of a fiber metal laminate lower wing cover demonstrator. In: Proceedings of AeroMat 2017, April 10–12, 2017, Charleston, SC
10. Tian Y, 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, Chang P, Sklyut H, Heinimann M (2019) Design, analysis and test development of full-scale fuselage test panels to assess emerging metallic structures technologies. DOT/FAA/TC-TN19/5
12. Stonaker K, Bakuckas J, Stanley D, Kulak M, Chang P, Freisthler M (2019) Assessment of fatigue behavior of advanced aluminum alloys under complex variable amplitude loading. In: Proceedings of the 30th ICAF symposium, June 5–7, 2019, Krakow, Poland
13. Steadman D (2007) Destructive evaluation and extended fatigue testing of retired transport aircraft, vol 5. Data analysis report. DOT/FAA/AR-07/22,V5
14. Lowak H, de Jonge J, Franz T, Schutz D (1979) MINITWIST—a shortened version of TWIST. NLR report no. TB-146