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Results From FAA Program to Validate Bonded Composite Doublers for Commercial Aviation Use

Dennis P. Roach

FAA Airworthiness Assurance NDI Validation Center

Sandia National Laboratories

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ABSTRACT

The number of commercial airframes exceeding twenty years of service continues to grow. In addition, Service Life Extension Programs are attempting to extend the "economic" service life of commercial airframes to thirty years. The use of bonded composites may offer the airframe manufacturers and aircraft maintenance facilities a cost effective method to extend the lives of their aircraft. The Federal Aviation Administration has sponsored a project at its *Airworthiness Assurance NDI Validation Center* (AANC) to validate the use of bonded composite doublers on commercial aircraft.

A specific application was chosen - reinforcement of an L-1011 door frame - in order to provide the proof-of-concept driving force behind this test and analysis project. However, the data stemming from this study serves as a comprehensive evaluation of bonded composite doublers for general use. The associated documentation package provides guidance regarding the design, analysis, installation, damage tolerance, and nondestructive inspection of these doublers. An industry team consisting of an Original Equipment Manufacturer (Lockheed-Martin), an airline (Delta Air Lines), and the AANC was formed. FAA oversight was provided through the Atlanta Aircraft Certification Office and the FAA's William J. Hughes Technical Center. Textron Systems Division provided their expertise in doubler installation and conducted a training class for Delta personnel.

This paper provides an overview of the FAA project and details the design, analysis, and test activities which were conducted in order to gain FAA approval for composite doubler use on commercial aircraft. Structural tests evaluated the damage tolerance and fatigue performance of composite doublers while finite element models were generated to study doubler design issues. Nondestructive inspection procedures were developed and validated using full-scale test articles. Finally, installation dry-runs demonstrated the viability of applying composite doublers in hangar environments. The first use of the project's documentation package was to support the installation of an FAA-approved Boron-Epoxy composite repair on a Lockheed L-1011 aircraft. This represents the first (non-decal) use of a bonded composite doubler on a U.S. commercial aircraft. A second important product of the results cited above is a Lockheed Service Bulletin which allows the door corner composite doubler to be installed on all L-1011 aircraft.

INTRODUCTION

One of the major thrusts established under the FAA's National Aging Aircraft Research Program is to foster new technologies associated with civil aircraft maintenance. A typical aircraft can experience over 2,000 fatigue cycles (cabin pressurizations) and even greater flight hours in a single year. The unavoidable by-product of this use is that crack and corrosion flaws develop throughout the aircraft's skin and substructure elements. These flaws must be repaired before an aircraft is returned to service. Most often, the repair consists of an engineered array of reinforcing plates which are fastened to the structure surrounding the damaged area. Optimization of this repair process is key to the continued safe and revenue-producing use of the aircraft. Composite doublers offer enhanced safety through improved aircraft fatigue life and corrosion resistance. Cost savings associated with their use (time savings in installation) is a desirable by-product which will accelerate their introduction into routine use.

Other advantages over mechanically fastened repairs include: 1) adhesive bonding eliminates stress concentrations caused by additional fastener holes, 2) composites are readily formed into complex shapes permitting the repair of irregular components, 3) composite doublers can be tailored to meet specific anisotropy needs thus eliminating the undesirable stiffening of a structure in directions other than those required, and 4) a high strength-to-weight ratio (reduced drag and weight).

Following is a brief overview which covers the array of tasks which were carried out to validate bonded composite doubler technology for commercial aircraft applications. Specific activities addressing the L-1011 door corner application are also presented. This project

produced a number of industry and FAA approval documents which will streamline future composite doubler applications. Follow-on projects have already been initiated which will take advantage of the foundation built by this technology validation effort.

TYPICAL COMPOSITE DOUBLER INSTALLATION - Figure 1 shows a typical bonded composite doubler repair over a cracked parent aluminum structure. The number of plies and fiber orientation is determined by the nature of the reinforcement required (i.e. stress field and configuration of original structure). Surface preparation is the most critical aspect of the doubler installation. This consists of paint removal, solvent clean, scotch-brite abrasion and chemical treatment to assure proper adhesion. Since the doubler must be installed in the field, vacuum bag pressure and thermal heat blankets, commonly used on in-situ honeycomb repairs, are used to cure the composite laminate and adhesive layer [1]. The taper at the edge of the doubler is used to achieve a uniform stress field in the area of maximum load transition. Finally, a top ply of fiberglass is installed to supply mechanical and environmental protection for the installation.

DAMAGE TOLERANCE ASSESSMENT

This test series utilized fatigue specimens to establish the damage tolerance of composite doublers bonded to aluminum skin. Each specimen consisted of an aluminum "parent" plate, representing the original aircraft skin, with a bonded composite doubler. The specimens contained engineered flaws in both the parent aluminum material and bonded doubler. The flaw scenarios included worst-case combinations of aluminum fatigue cracks, doubler impact damage, hot/wet conditioning and disbonds between the doubler and the skin. Tension-tension fatigue tests were used to: 1) assess the potential for interply delaminations and disbonds between the aluminum and the laminate, and 2) determine load transfer and crack mitigation capabilities of composite doublers in the presence of severe defects.

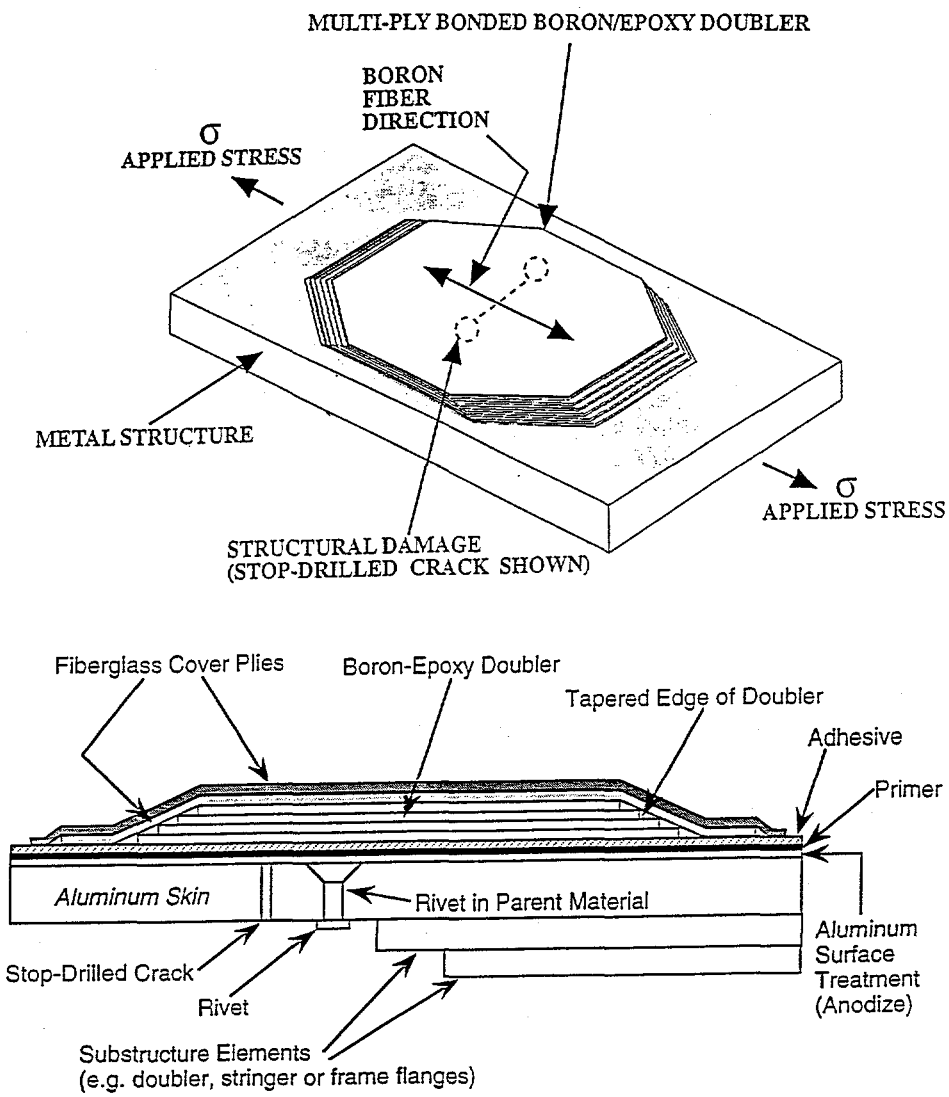

Figure 1: Schematic of Bonded Composite Doubler Installation on an Aluminum Skin

It was demonstrated that even in the presence of extensive damage in the original structure (cracks, material loss) and in spite of non-optimum installations (adhesive disbonds), the composite doubler allowed the structure to survive more than 144,000 cycles of fatigue loading [2]. Installation flaws in the composite laminate did not propagate over 216,000 fatigue cycles. Comparisons with control specimens which did not have composite doubler reinforcement showed that the fatigue lifetime was extended by a factor of 20. Furthermore, the added impediments of impact - severe enough to deform the parent aluminum skin - hot-wet exposure, and disbonds, representing almost 30% of the axial load transfer perimeter, did not effect the doubler's performance.

All specimens showed aluminum-to-composite load transfer values of 35% - 50%. This value remained constant over four fatigue lifetimes indicating that there was no deterioration in the bond strength. Post-fatigue ultimate strength tests showed that the doubler was able to restore the damaged aluminum skin to its original load carrying capacity [2-4]. Since the tests were conducting using extreme combinations of flaw scenarios (sizes and collocation) and excessive fatigue load spectrums, the performance parameters were arrived at in a conservative manner.

TECHNOLOGY EVALUATION THROUGH FULL-SCALE STRUCTURAL TESTS

FUSELAGE TEST ARTICLE AND BIAXIAL TEST FACILITY - This phase of the project's test series utilized a complete L-1011 door surround structure to verify key doubler design and analysis parameters. The full-scale tests studied load transfer around the composite doublers using pressure and axial loads. The data was also used to validate Finite Element Models (FEM) of the L-1011 application [1]. Nondestructive Inspections (NDI) were interjected throughout the test series in order to: 1) evaluate the capabilities of each technique in a "real-life" environment, and 2) develop NDI procedures which can be used by airline technicians on aircraft in the field. Finally, the door surround structure tests evaluated the effectiveness of a composite doubler in reducing the stresses around an actual L-1011 door frame corner. The L-1011 application contained many extreme engineering challenges in terms of number of plies, geometry complexity, the presence of cut-outs, high shear loads (lack of taper), and footprint size. Thus, the full-scale tests produced data to support a large array of doubler applications and designs.

The fuselage door surround structure test article (see Figure 2) was cut from a retired L-1011 aircraft. It was approximately 141" H (151" arc length in the hoop direction) X 114.75" W and included all of the substructure elements. The specimen contained eight circumferential frames and six longitudinal stringers, as well as, the upper and lower longeron around the door cut-out. The Lockheed-designed composite doubler, also shown in Figure 2, was installed by Delta Air Lines technicians at Delta's Atlanta maintenance facility in accordance with Ref. [1]. The doubler was fabricated in accordance with the Lockheed design drawing LCC-7622-378 [5]. The perimeter of the door surround structure was reinforced with bonded and bolted doubler plates to accommodate the tension rams and clevises in the axial direction and the turnbuckle restraints in the hoop direction.

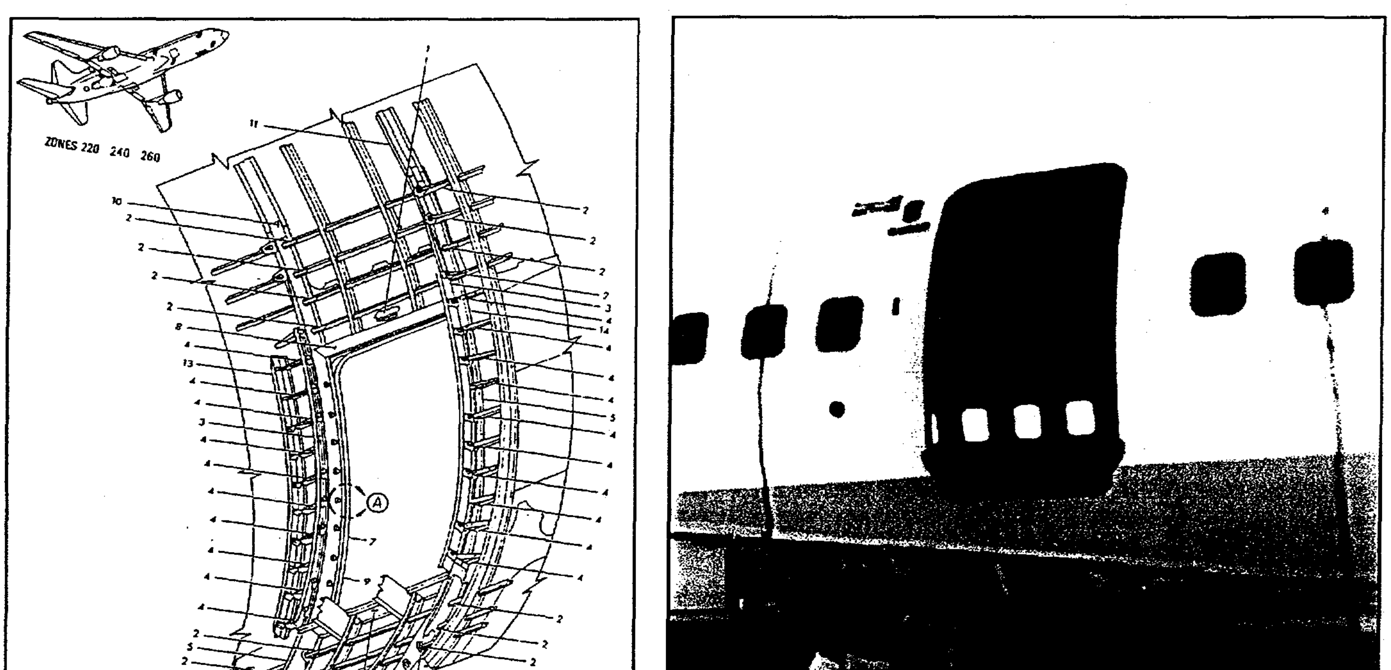

Figure 2: Widebody Fuselage Test Article (L-1011 Door Surround Structure)

The door surround structure test article was subjected to a combined load environment of external vacuum (note: external vacuum is used to simulate the internal pressure which generates the primary hoop stresses in the airplane) and axial, or longitudinal, stress. The differential cabin pressure was generated using a custom vacuum chamber while the axial loads were applied by hydraulic rams. The applied biaxial tension loads approximated the stresses induced by normal flight pressure loads.

Figure 3 shows a schematic of the door surround structure mounted in the biaxial test facility. It shows the vacuum chamber and the restraints in the hoop direction and the load trains in the axial direction. Turnbuckles in the hoop direction allowed adjustments to the boundary conditions in order to assure a uniform load across the width of the door surround structure. As the vacuum load was applied, hoop and axial loads were developed in the test article. In order to maintain proper control over the desired ratio between hoop and axial loads, twelve tension rams were used to supplement and or relieve axial loads during the tests. The loads applied by the hydraulic tension rams or restrained by the turnbuckle load trains were monitored by load cells.

The entire strain field on and around the composite doubler was monitored using a series of biaxial and Rosette strain gages. Data was acquired to study the following issues: 1) load transfer through the doubler, 2) stresses in the doubler and parent structure, and 3) effect of the doublers on the fuselage stresses adjacent to the doubler. The strain gage locations are shown in Figure 4. The locations shown correspond to high strain areas, potential crack initiation sites, and important load transition regions along the outer perimeter of the doubler.

FUSELAGE TEST RESULTS - Reference [6] is a comprehensive report on the test series results. Some of the important results will be highlighted here. The door surround structure was tested before and after the doubler was installed to evaluate strain field improvements.

The maximum principal stresses in the doubler-affected region are listed in Table 1. The largest stress in the aluminum skin was found at R-7 (inside skin) at the door corner radius. The magnitude was 10.6 KSI, down from the 15.3 KSI recorded at that same location before the doubler was installed. The largest Boron-Epoxy stress was 19.3 KSI at the R-8 Rosette gage. This is less than the 22.4 KSI measured at that same location before the doubler was installed. Similar comparisons of aluminum stresses before the doubler versus aluminum/boron stresses at the same locations after the doubler revealed the stress reductions summarized in Table 1. It can be seen that the stresses were reduced by as much as 53% through the application of the reinforcing doubler. Figure 5 plots the reduction in several principal stresses after the doubler was installed.

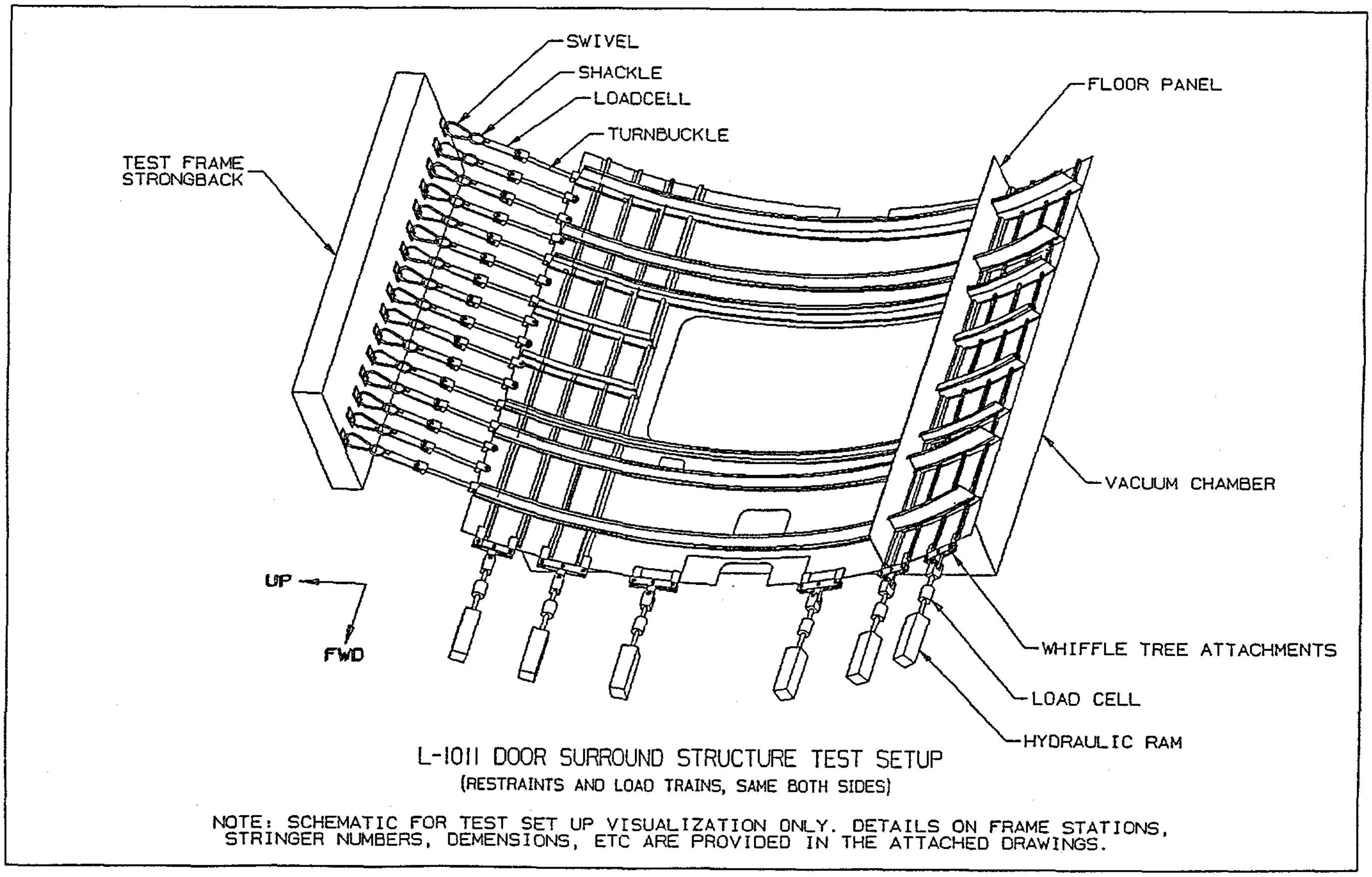

Figure 3: Biaxial Test Facility Showing Fuselage Structure Mounted in Vacuum Chamber and Hoop Restraint, Axial Load Application Hardware

Rosette Gage Number	Stress Before Doubler (KSI)	Stress After Doubler (KSI)	Percent Reduction in Stress
R-2	12.7	6.9	45% *
R-4	9.6	6.2	35% *
R-5	7.5	4.9	35%
R-6	15.3	12.9	16% *
R-7	15.3	10.6	31%
R-8	22.4	19.3	14% *
R-10	13.2	6.2	53% *

* Indicates aluminum stress before vs. boron stress in same location after doubler was installed

Table 1: Principal Stress Reductions Observed Within the Doubler Footprint

Common locations inside and outside the test article were gaged to obtain information regarding out-of-plane bending in critical regions. Bending strains, an important consideration in fuselage skin doublers, accounted for 30% to 50% of the total strain in the structures being monitored. The load transfer into the doubler - and away from the aluminum - was approximately constant at each location regardless of the applied loads. The load transfer was consistently in the 40 - 60 % range. These load transfer levels agree with the results obtained in the damage tolerance tests described above.

The stress reductions produced by the composite doubler, along with the fact that the doubler did not induce any undesirable stress risers in the door corner area, quantify the realization of the basic doubler design goals: 1) to uniformly reinforce the area (global reduction in door corner strains), and 2) achieve the proper stiffness ratio between the parent skin and composite laminate (excessive doubler stiffness would attract loads and produce stress risers).

ADDITIONAL TESTS TO EVALUATE DOUBLER RESPONSE IN EXTREME CONDITIONS - Other full-scale test articles were designed and tested to assess the response and ultimate strength of composite doublers in extreme load scenarios [3]. The primary issues to be addressed were doubler delamination and disbond especially in high peel stress regions. Tension,

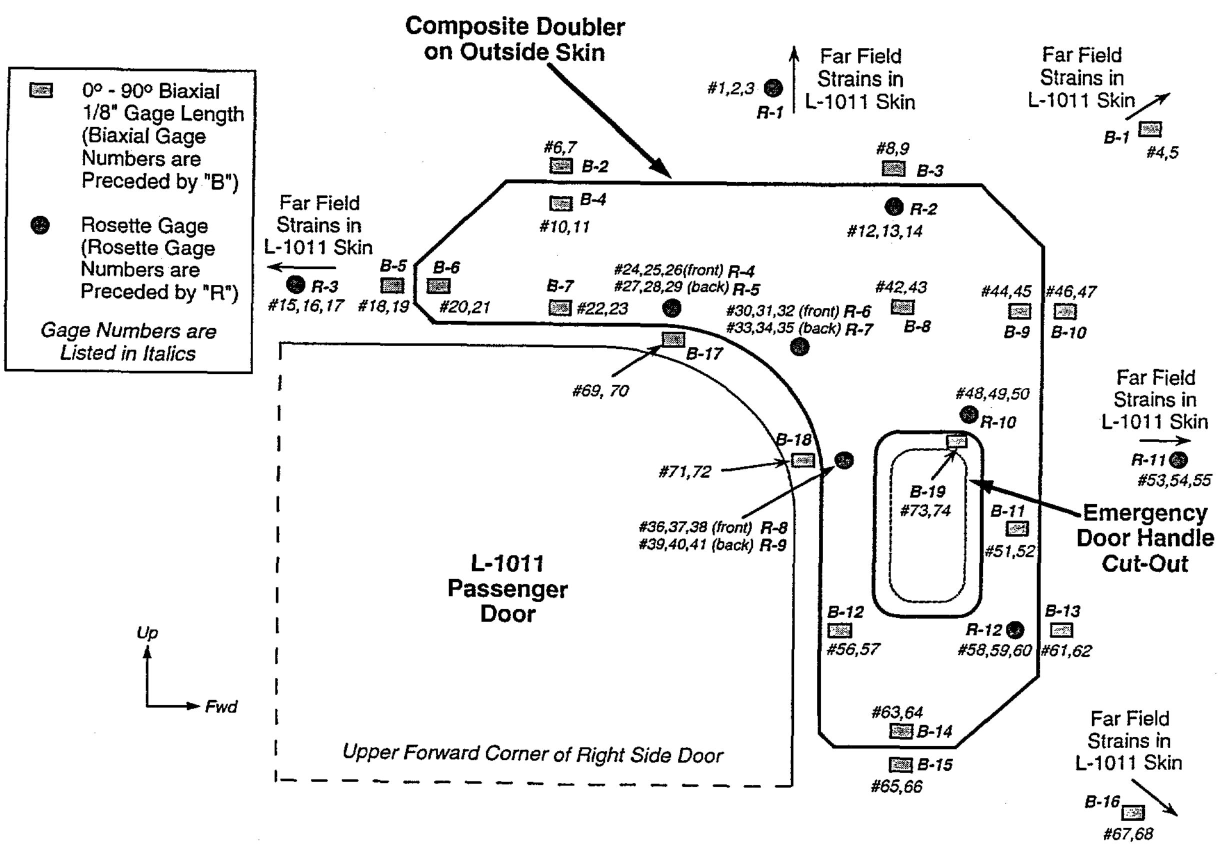

Figure 4: Gage Layout to Monitor Strain Field in Composite Doubler and Door Structure

shear, and bending loads were applied until failure occurred in extremely thick doublers (72 plies = 0.41" th.) which had no thickness taper (ply drop-offs) around the perimeter. Important test results were: 1) the bond is able to transfer plastic strains into the doubler, 2) the aluminum skin must experience yield strains before any damage to the doubler will occur, and 3) a properly designed doubler can produce limit load safety factors in excess of 2.5 (minimum repair design goals are around 1.5) while improving the fatigue life of the parent structure.

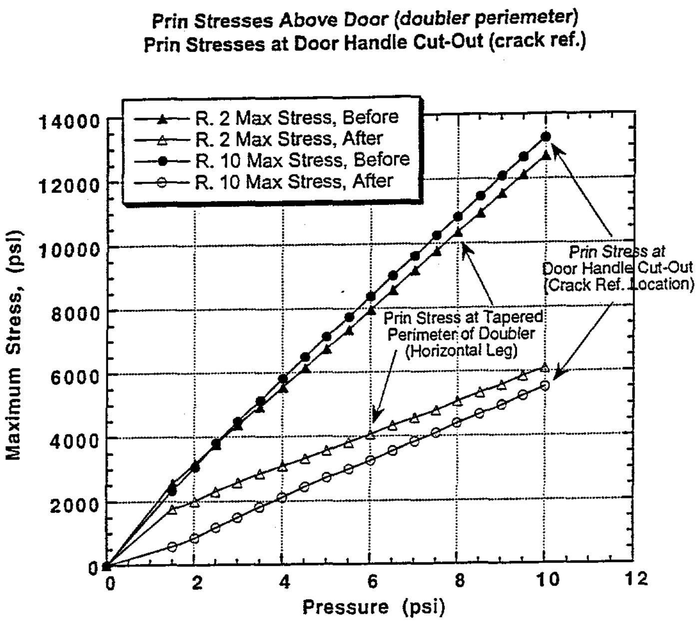

Figure 5: Comparison of Principal Stresses Before and After Doubler Installation

DESIGN AND ANALYSIS (FEM EVALUATION)

The performance of the 72 ply composite doubler was analyzed using a finite element model (FEM) of the fuselage structure containing the passenger door [7]. The fuselage tests described above were used to assess the general performance of the doubler and to validate the analytical model. Results from the validated FEM were then used to predict the composite doubler and aluminum skin stresses during maximum flight load scenarios. The analysis determined the strength of the doubler and adhesive joint and checked the potential for crack growth in the parent structure. Overall safety margins were established for the installation.

The FEM of the door region included half of the fuselage between fuselage stations FS769 and FS963. Figure 6 shows the FEM of the P-3 door region. The finer mesh, to improve the model's resolution in the area of the composite doubler (note door handle cut-out forward of the door), is highlighted in the Figure 6 inset.

Validation of the FEM consisted of comparisons between the analytical results and strain gage measurements obtained during the full-scale fuselage tests. A comparison between the principal strain values determined from the test data and the corresponding FEM strains is shown in Figure 7. The FEM predictions were particularly good in the area of greatest concern around the door corner radius and near the upper forward corner of the emergency door handle release cut-out. The FEM results were primarily within 10% of the experimental data.

These comparisons indicated that the FEM was able to accurately determine the stress field, damage tolerance and crack mitigation capabilities of the bonded composite doubler. The crack growth analysis revealed that the doubler increases the safety-limit of the fuselage structure by a factor of 2.8. The damage tolerance analysis showed that the doubler exhibits sufficient strength to provide adequate fatigue enhancement over the full spectrum of environmental conditions.

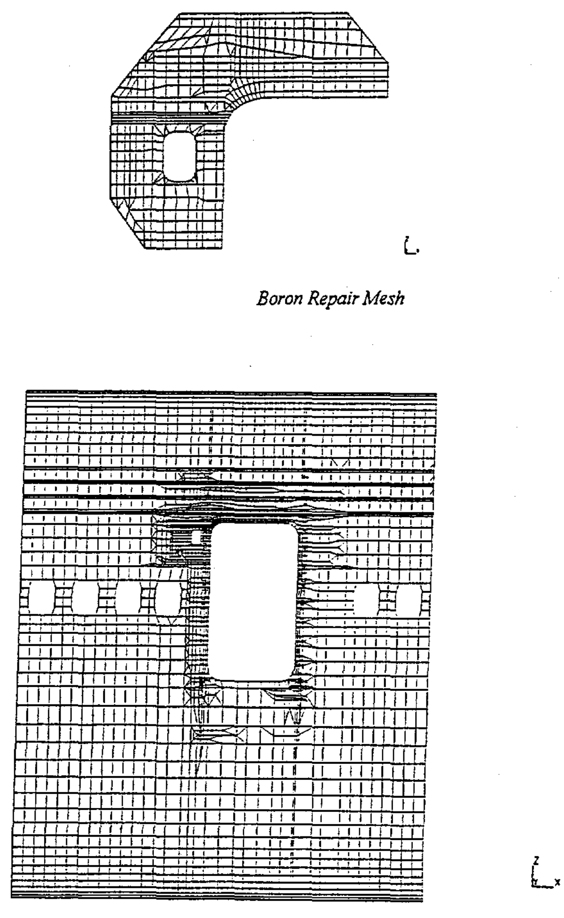

Figure 6: Finite Element Mesh of Fuselage Section and Composite Doubler

NONDESTRUCTIVE INSPECTION

Periodic inspections of the composite doubler for disbonds and delaminations (from fabrication, fatigue, or impact damage) is essential to assuring the successful operation of the doubler over time. Nondestructive inspection is affected by the geometry and material properties of the doubler installation. The thickness of the doubler creates lift-off effects during eddy current (EC) inspections and signal attenuation during ultrasonic (UT) examinations. Eddy current and X-ray inspections performed on the door surround structure test article did not reveal any cracks in the parent structure. After the doubler was installed and tested in the biaxial load facility, X-ray, eddy current, and ultrasonic inspections did not show any disbond/delamination flaws (composite laminate) or cracks (aluminum) in test article. References [8-10] describe the applicable NDI techniques in detail. The important features of applicable NDI techniques are summarized below.

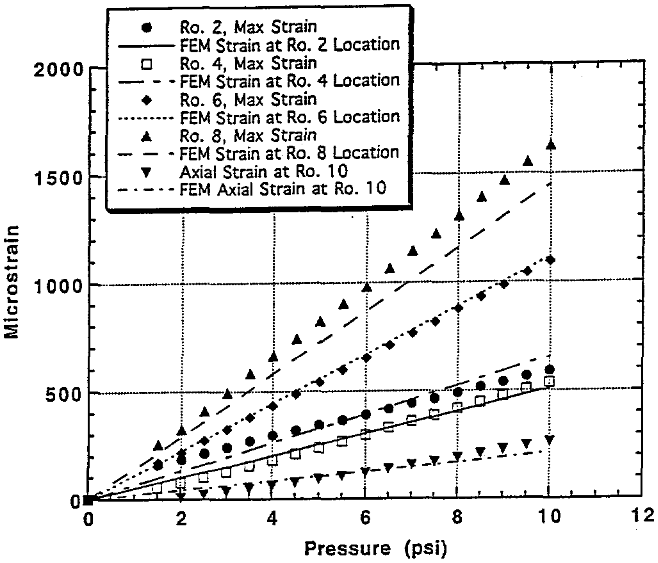

Figure 7: Comparison of Experimental and Analytical Principal Strains in the Doubler

ULTRASONIC INSPECTIONS - The AANC has used ultrasonics to detect both interply delaminations as well as disbonds at the laminate-to-aluminum interface. In conventional Pulse-Echo Ultrasonics (PE UT), pulses of high frequency sound waves are introduced into a structure being inspected. A-Scan signals represent the response of the stress waves, in amplitude and time, as they travel through the material. As the waves interact with defects or flaw interfaces within the solid and portions of the pulse's energy are reflected back to the transducer, signal anomalies (flaws) are detected, amplified and displayed on a CRT screen.

Improvements in disbond detection can be achieved by taking the A-Scan signals and transforming them into a single C-Scan image of the part being inspected. C-Scan technology uses information from single point A-Scan waveforms to produce an area mapping of the inspection surface. These 2-D images are produced by digitizing point-by-point signal variations of an interrogating sensor while it is scanned over a surface. C-Scan area views provide the inspector with easier-to-use and more reliable data with which to recognize flaw patterns. Specific emphasis can be placed on the UT signal - and highlighted in the color-mapped C-Scan - based on user specified amplitude gates and time-of-flight values.

The automated (motorized) C-Scan system used by the AANC to inspect bonded composite doublers is shown in Figure 8. The scanner is inspecting the composite doubler on the L-1011 fuselage test article. The echo signal is recorded, versus its X-Y position on the test piece, and a color coded image is produced from the relative characteristics of the sum total of signals received. A UT inspection procedure was developed by the AANC and approved for use on L-1011 aircraft by Lockheed-Martin [8]. This procedure was applied to the door corner doubler after it was installed on a Delta L-1011 aircraft. Figure 9 shows a sample C-scan from an ultrasonic inspection of a composite doubler with engineered flaws (reference standard). Disbond and delamination flaws are revealed by continuous and distinct signal loss areas (bright spots on color map).

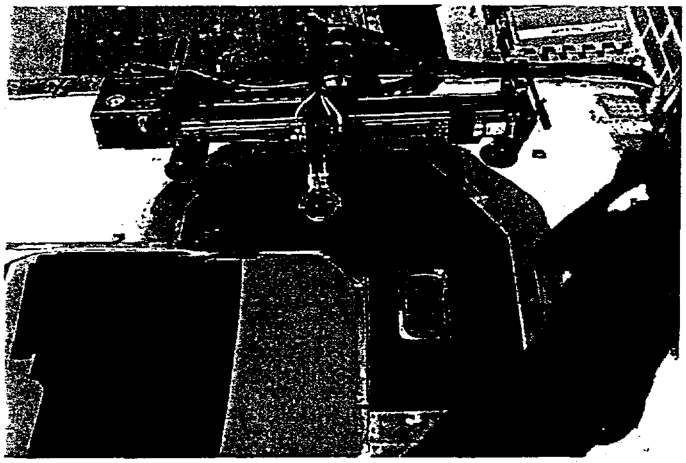

Figure 8: Automated Ultrasonic Scanner Inspecting the Composite Doubler on the Fuselage Test Article

Extensive testing has shown that the two-dimensional, color coded images produced by manual and automated scanners are able to reliably detect disbond and delamination flaws on the order of 0.50" in diameter. Time savings, human factors issues, and repeatability are some of the main advantages associated with C-Scan ultrasonics. Key to implementing this NDI technique is the use of representative calibration standards which allow for accurate equipment settings (amplifier gains and signal gates) over the full range of laminate thicknesses [10].

X-RAY INSPECTIONS - The AANC conducted a study to: 1) demonstrate that composite doublers do not interfere with the ability to perform X-ray inspections for cracks in aluminum, and 2) identify proper exposure time and power settings to optimize the sensitivity of X-ray technique when inspecting through extremely thick composite doublers [10]. Radiography was demonstrated to be a very effective inspection method to detect cracks in the parent, aluminum material covered by the composite doubler. The Boron-Epoxy material does not impede the X-ray inspections. Radiography is as effective as before a doubler is installed. Power and exposure times can be adjusted to accommodate the presence of the doubler and achieve the required film density and resolution.

Several fatigue crack specimens and the L-1011 fuselage section were inspected through a 72 ply composite doubler (0.4" th). To form a basis of comparison, X-rays were also taken without the doubler placed over the cracked specimens. The specimens included 1st layer, 2nd layer, EDM notch and fatigue cracks ranging from 0.05" to 1.0" in length. The required damage detection threshold for cracks under the doubler is 1.0". Fatigue cracks on the order of 0.38" in length were found under 0.40" thick Boron-Epoxy doublers. Image Quality Indicators (IQI), inserted into the field of view, verified the resolution and sensitivity of the radiographic technique. IQI lines with widths of 0.010" and dots with diameters of 0.10" were clearly imaged on the X-ray film.

EDDY CURRENT INSPECTIONS - Eddy Current (EC) inspection uses the principles of electromagnetic induction to identify or differentiate structural conditions in conductive metals. It was applied to numerous bonded composite doubler installations in order to assess the ability of EC to detect cracks in aluminum skin beneath a composite laminate. The presence of a crack is indicated by changes in the flow of eddy currents in the skin. Initial testing conducted by the AANC on fatigue crack specimens established the following limits of crack detectability through composite doublers: 1) a 0.060" long first layer (surface) crack can be detected in the aluminum through a 0.310" thick doubler, 2) a 0.15" length surface crack can be detected through a 0.5" thick laminate, and 3) a 0.15" long subsurface crack (0.040" th. surface plate) can be detected through a 0.310" thick doubler [10].

FAA DOCUMENTATION AND INDUSTRY APPROVAL

The data stemming from this study serves as a comprehensive evaluation of bonded composite doublers for general use. The associated documentation package provides guidance regarding the design, analysis, installation, damage tolerance, and nondestructive inspection of these doublers. Although an initial aircraft application was pursued in parallel to this investigation, the overall goal was to provide results that are pertinent to any use of Boron-Epoxy doublers for commercial aircraft reinforcement or repair. In order to streamline the use of composite doublers in other applications, the documentation package for this validation effort resides in the public domain. The FAA's Atlanta Aircraft Certification Office maintains the documents under the FAA project number SP1798AT-Q. The documentation package is summarized in Table 2.

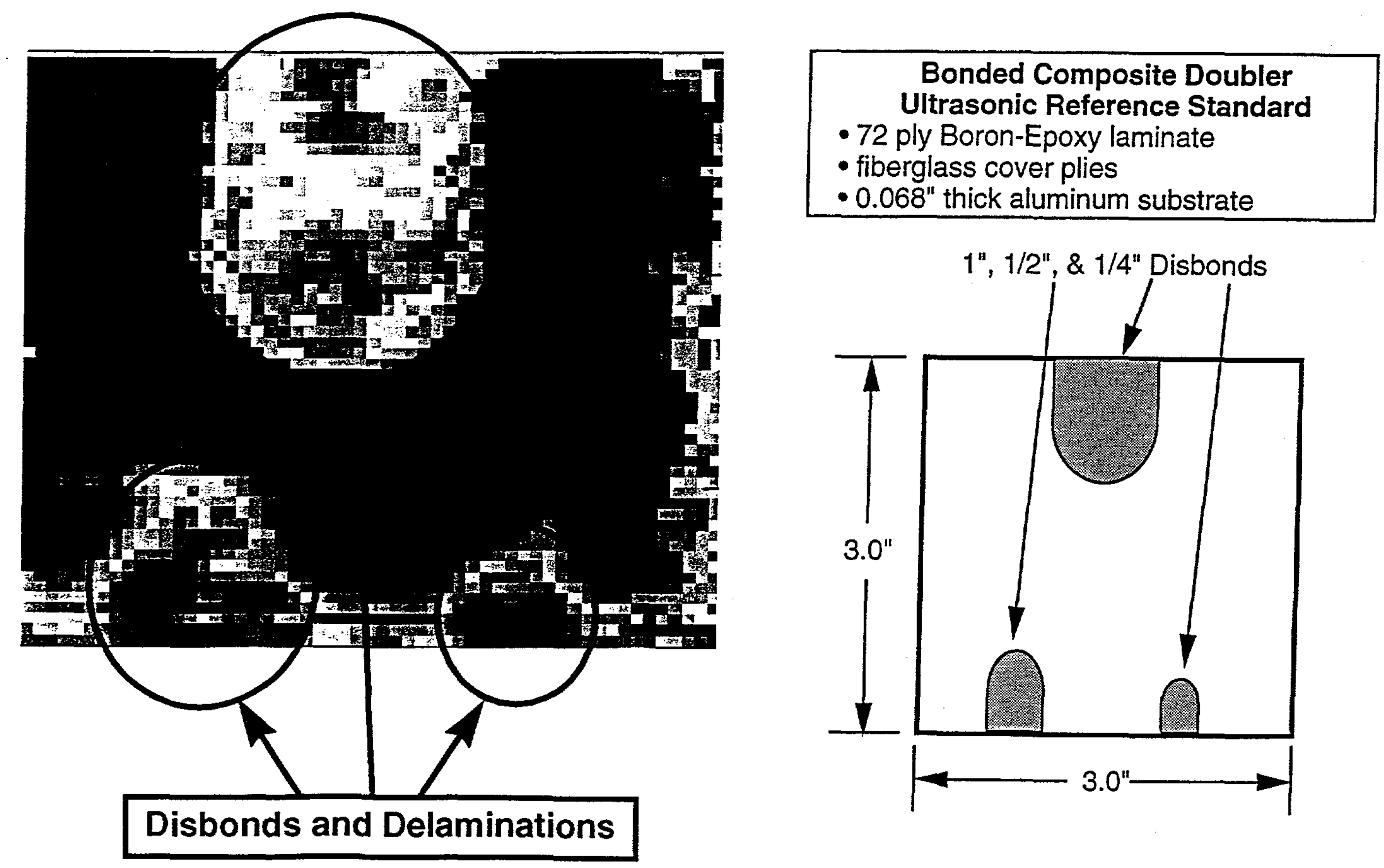

Figure 9: Sample C-Scan Image from Ultrasonic Inspection of 72-Ply Reference Standard Containing Engineered Flaws

Item	Report/ Information Type	FAA Report Number
1	Boron-Epoxy & Adhesive Material Allowables	LG95ER0193
2	Damage Tolerance Assessment of Composite Doublers	SNL96ER0189
3	Full-Scale Structural and NDI Testing	SNL96ER0006
4	Boron-Epoxy Doubler Installation Process Specification	TSM 2000,008-001
5	Design and Analysis of Bonded Composite Doubler Repairs	LG95ER0157
6	Nondestructive Inspection Procedures	AANC-PEUT-Comp-5521/4-004

Table 2: Public Domain Documentation on Composite Doubler Technology

INDUSTRY RECOGNITION OF COMPOSITE DOUBLER REPAIR TECHNOLOGY - The other important outcome from the activities cited above are the series of industry and FAA approvals which: 1) indicate the industry's endorsement of the technology, and 2) allow for expanded use of bonded composite doublers on commercial aircraft. A Lockheed-Martin Service Bulletin (093-53-278) is being issued. It will allow the door corner composite doubler to be installed on all L-1011 aircraft. In addition to obtaining approval from a Lockheed Designated Engineering Representative (DER), the FAA also approved all data and procedures stemming from this project. An Alternate Means of Compliance (AMOC) was granted by the FAA and allows for the deviation from the original, metallic repair Service Bulletin (093-53-237) and the FAA Aging Aircraft Airworthiness Directive (94-05-1).

COMPOSITE DOUBLER INSTALLATION ON A COMMERCIAL AIRCRAFT

The best proof that the composite doubler repair technique is viable on commercial aircraft was provided by the recent installation of a doubler on a Delta Air Lines L-1011. The Sandia Lab's AANC, Textron Specialty Materials and Delta Air Lines performed the installation and inspection of the L-1011 door corner composite doubler. It was installed in lieu of the standard repair of four, riveted metallic plates. Figure 10 shows the Delta aircraft and the doubler installation staging area at the right side, P3 passenger door. Figure 11 shows the aircraft skin section which was cleaned and anodized - via a phosphoric acid non-tank anodize (PANTA) process - to support the doubler bonding process [1]. In order to optimize the uniformity and stability of the temperature field, the 190°F, six-hour cure cycle was used. Three heat blankets were used to provide the computer-controlled temperature profile and heat lamps were positioned on the internal fuselage structure to minimize the effects of the various heat sinks. The completed composite doubler installation is shown in Figure 12. A fiberglass environmental protection ply and a copper mesh lightning protection ply were installed on top of the Boron-Epoxy laminate. The contour around the door corner radius, the cut-out to accommodate the emergency door handle access port, and the tapered edges around the perimeter are evident.

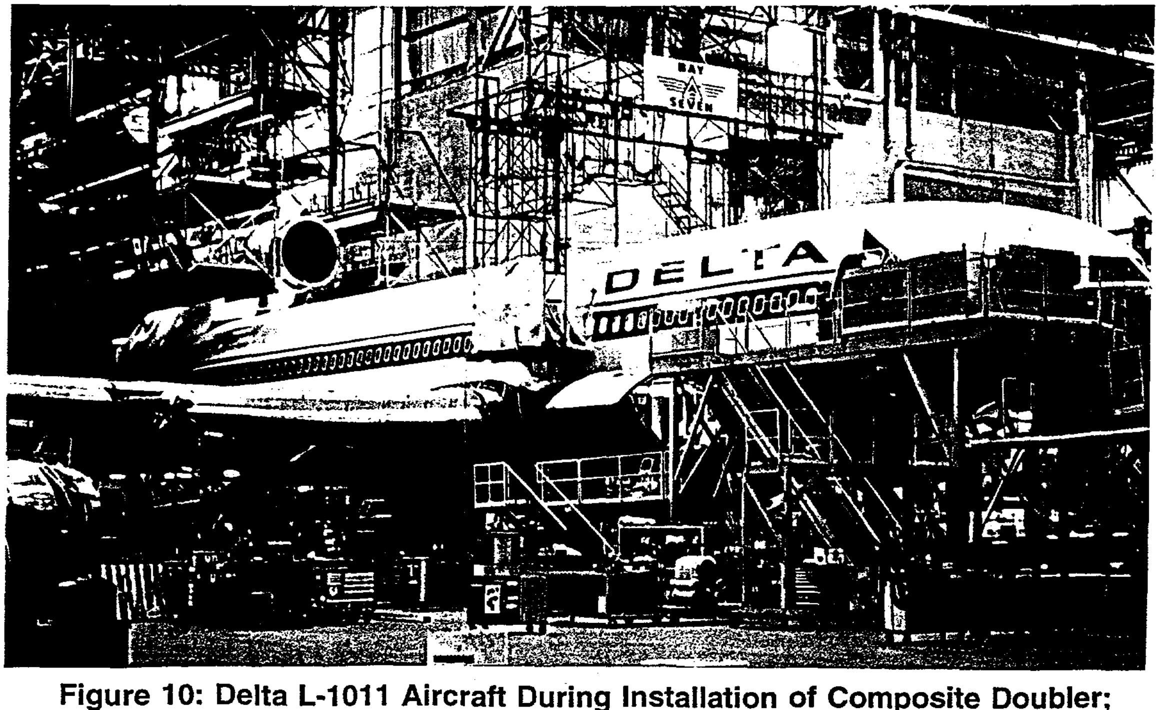

Figure 10: Delta L-1011 Aircraft During Installation of Composite Doubler; Work Stand is Positioned Next to Right Side Passenger Door

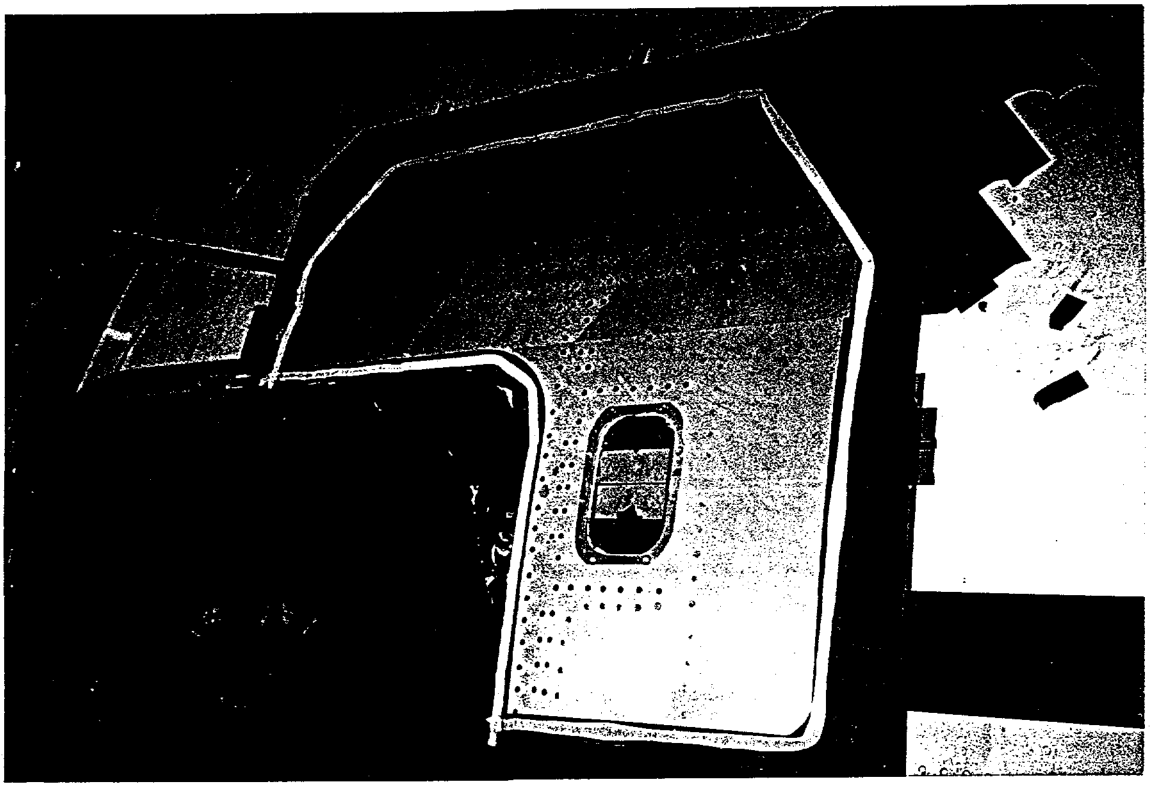

Figure 11: Anodized Door Corner Prepared for Doubler Bonding Process

Figure 12: Finished Product - Composite Doubler Following Cure Process

The doubler was inspected using pulse-echo ultrasonics (UT). Most of the flaw detection effort was focused on the critical 1.5" wide strip around the perimeter of the doubler. The allowable flaw size in this load transfer region is a 0.5" diameter. The inspection did not reveal any flaws in the L-1011 doubler. A series of quality assurance measures were included in the process to assure sufficient strength in the adhesive layer and Boron-Epoxy laminate. Wedge tests showed that the surface preparation was good and that the full adhesive strength was achieved. Shear and bending tests demonstrated that the laminate had sufficient strength. The installation was considered a complete success and the aircraft has returned to service in the Delta Air Lines fleet.

From an engineering standpoint, the door corner application provided a good showcase for composite doubler capabilities. The design, fabrication, and installation challenges included large heat sinks, severe bending loads (shear stresses), a cut-out in the center of the doubler, a complex geometry, multiple taper directions, and an extremely thick (72 ply, 0.040" th) doubler.

In addition to the engineering benefits noted above, bonded composite doublers can provide economic advantages. Due to the engineering challenges discussed above, this L-1011 door corner application does not provide the optimal cost-benefit case study. However, the composite doubler installation, although not streamlined by repetition, produced an approximate 50% reduction in man-hours versus the conventional metallic repair.

CONCLUSION

The unavoidable by-product of aircraft use is that crack and corrosion flaws develop throughout the aircraft's skin and substructure elements. Economic barriers to the purchase of new aircraft have created an aging aircraft fleet and placed even greater demands on efficient and safe repair methods. The composite doubler repair technique provides a safe and cost-effective solution to aircraft repair challenges. In many applications, engineering benefits make composite doublers an attractive alternative to metallic repairs.

This program demonstrated the successful performance of composite doubler repairs when exposed to the complex environments on an operating aircraft. It established a technical foundation for general composite doubler repairs on commercial aircraft. Furthermore, appropriate FAA and industry approvals for the use of composite doublers on commercial aircraft were produced. Critical elements of this program included:

1. Design and analysis methodology - determined material allowables and groundrules for an acceptable composite doubler design
2. Installation procedures - established process necessary for a successful installation
3. Damage tolerance assessment - demonstrated proper performance of composite doublers in worst-case flaw and load scenarios
4. Full-scale testing - used aircraft structure and custom test facilities to assess composite doubler repair technology in real-life environments
5. Nondestructive inspection - developed techniques and certified procedures necessary to assure the continued safe operation of composite doublers
6. FAA approvals - produced a comprehensive documentation package and associated FAA approvals necessary to offer this technology to industry.

The entire aviation industry can receive the engineering and economic benefits provided by this repair technology. Technical advantages include: 1) improved fatigue life, 2) increased strength, 3) decreased weight, 4) eliminates introduction of crack initiation sites (i.e. fastener holes), 5) does not corrode, and 6) improves aerodynamics. Economic benefits include: 1) cost savings through reduction in man-hours required to install a repair, and 2) reduced aircraft downtime.

The aviation industry and the FAA are continuously searching for ways to improve aircraft maintenance practices. Enhanced safety is the primary goal while cost reduction is necessary to our nation's competitiveness in the global air transportation market. Composite doubler repairs successfully address both of these concerns.

ACKNOWLEDGMENTS

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