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Validation of Bonded Composite Doubler Technology Through Application Oriented Structural Testing

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ABSTRACT

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. Recent DOD and other government developments in the use of bonded composite patches on metal structures has supported the need for research and validation of such doubler applications on U.S. certificated airplanes. Composite patching is a rapidly maturing technology which shows promise of cost savings on aging aircraft. While there have been numerous studies and military aircraft installations of composite doublers, certain information gaps and a slow acceptance of this new technology on the part of the aviation industry has suppressed the extension of composite doublers into commercial applications. Sandia Labs is conducting a proof-of-concept project with Delta Air Lines, Lockheed Martin, Textron, and the FAA which seeks to remove any remaining obstacles to the approved use of composite doublers. By focusing on a specific commercial aircraft application - reinforcement of the L-1011 door frame - and encompassing all "cradle-to-grave" tasks such as design, analysis, installation, and inspection, this program is designed to prove the capabilities of composite doublers. This paper reports on a series of structural tests which have been conducted on coupons and subsize test articles. Tension-tension fatigue and residual strength tests attempted to grow engineered flaws in coupons with composite doublers bonded to aluminum skin. Also, structures which modeled key aspects of the door corner installation were subjected to extreme tension, shear, and bending loads. In this manner it was possible to study strain fields in and around the Lockheed-designed composite doubler using realistic aircraft load scenarios and to assess the potential for interply delaminations and disbonds between the aluminum and the laminate. The data acquired was also used to validate finite element models (FEM) and associated Damage Tolerance Analyses.

INTRODUCTION

Limited commercial aircraft demonstrations and operational testing have confirmed that under proper conditions, composite doublers can provide a long lasting and effective repair or structural reinforcement [1-5]. Repairs and reinforcing doublers using bonded composites have been reported to have numerous advantages over mechanically fastened repairs. Adhesive bonding eliminates stress concentrations caused by additional fastener holes. Composites are readily formed into complex shapes permitting the repair of irregular components. Further, composite doublers can be tailored to meet specific anisotropy needs thus eliminating the undesirable stiffening of the structure in directions other than those required. Numerous articles have addressed the myriad of concerns associated with repairing aircraft structures. Reference [6] presents the results of a program which demonstrated the successful application of externally bonded composite repairs while Ref. [7] highlights how riveted metallic repairs can degrade the fatigue initiation life and damage tolerance capabilities of aircraft structures.

The number of commercial airframes exceeding twenty years of service continues to grow. In addition, Service Life Extension Programs are becoming more prevalent and test and evaluation programs are presently being conducted to extend the "economic" service life of commercial airframes to thirty years. The use of bonded composites may offer the airframe manufacturers and airline maintenance facilities a cost effective technique to extend the lives of their aircraft.

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DAMAGE TOLERANCE FATIGUE AND STATIC ULTIMATE TESTS

Test Objectives - This test series utilized small-scale fatigue specimens to assess the strength and stability of composite doublers bonded to aluminum skin. An array of design parameters, including various flaw scenarios, the effects of surface impact, and other "off-design" conditions, were studied. Tension-tension fatigue tests: 1) attempted to grow engineered flaws, and 2) determine load transfer capabilities of composite doublers in the presence of defects. Several specimens which survived the fatigue tests were subjected to static ultimate tension tests in order to determine ultimate strength and failure modes.

Basic Specimen Design - The test series included nine different specimen configurations. Each specimen consisted of an aluminum "parent" plate, representing the original aircraft skin, with a bonded composite doubler. The doubler was bonded over a flaw in the parent aluminum. Table I summarizes the engineered specimen flaws which included fatigue cracks (unabated and stop-drilled), aluminum cut-out regions, impact damage, hot/wet conditioning and disbond combinations. Figure 1 shows one of the most severe flaw scenarios (Specimen BE-4) in which an unabated fatigue crack has a co-located disbond (i.e. no adhesion between doubler and parent aluminum plate) as well as two, large, 0.75" diameter disbonds in the critical load transfer region of the doubler perimeter. The aluminum plate was 0.070" thick, 2024-T3 to match the L-1011 fuselage skin around the door frame.

The boron-epoxy composite doublers were a symmetric, multi-axial lay-up of 13 plies: $[0, +45, -45, 90]_3$ with a 0° cover ply on top. The plies were cut to different lengths in both in-plane directions in order to taper the thickness of the resulting doubler edges. This produces a more gradual load transfer between the aluminum and the doubler (i.e. reduces the stress concentration in the bondline around the perimeter). The number of plies and fiber orientations were chosen such that the crossectional stiffness ratio of boron/epoxy to aluminum was 1.4:1 $\{(Et)_{Al} = 1.4\ (Et)_{BE}\}$.

Test Environment - Tension-tension fatigue tests on the coupon specimens used baseline stress levels of 3 KSI to 20 KSI (850 - 5600 lbs. load). The lower stress limit, or test pre-load, was used to eliminate the residual curvature in the test specimen which results from the different coefficients of thermal expansion between the aluminum and boron-epoxy materials. The upper stress limit was based on the maximum hoop stresses observed in the L-1011 skin (cabin pressure plus flight loads). Load transfer through the composite doubler and stress risers around the defects was monitored using strain gage layouts similar to the one shown in Figure 1. 72,000 cycles corresponds to two design lifetimes for the L-1011 aircraft. The fatigue tests continued until unstable flaw growth occurred or until a maximum of 144,00 cycles (4X the design objective) were reached. The flaw profiles were monitored using eddy current and ultrasonic inspection techniques [8]. Several of the specimens which survived the fatigue tests were subjected to static ultimate tension tests in order to determine ultimate strength (residual strength on flawed specimens) and failure modes.

DAMAGE TOLERANCE TEST RESULTS

Fatigue Tests

Fatigue tests have been completed on specimens BE-1 through BE-6. The results are summarized in Table I and shown graphically in Figure 2. The main items of note are as follows:

1. Stop-Drilled Cracks with Composite Doubler Reinforcement - Specimens BE-2 and BE-3 showed that crack growth could be eliminated for a number of fatigue lifetimes using this configuration (note delay of crack reinitiation until 72 K and 126 K cycles in the Fig. 2 BE-2 and BE-3 curves). This was true in spite of the performance reducing impediment of adhesive disbonds between the doubler and the aluminum plate. Because of this initial crack growth arrest, Specimens BE-2 and BE-3 experienced total crack growths of less than 1.75" up through 144 K fatigue cycles.

Configuration	Description	Fatigue Test Results
BE-1	unabated 0.5" crack in aluminum; no engineered flaws in composite doubler installation	• crack propagated 1.78" in 144K cycles
BE-2	stop-drilled 0.5" crack in aluminum with co-located adhesive disbond; 0.75" dia. disbonds in edge of doubler	• stop-drilled crack reinitiated after 126 K cycles • crack propagated 0.875" in 144 K cycles
BE-3	stop-drilled 0.5" crack in aluminum with co-located adhesive disbond; 1.0" dia. disbonds in edge of doubler	• stop-drilled crack reinitiated after 72 K cycles (small burr in stop-drilled hole acted as starter notch) • crack propagated 1.71" in 144 K cycles
BE-4	unabated 0.5" crack in aluminum with co-located adhesive disbond; 0.75" dia. disbonds in edge of doubler	• crack propagated 2.21" in 144 K cycles • fatigue test was extended until specimen failure occurred at 182 K cycles
BE-5	1" dia. hole in parent aluminum plate; no engineered flaws in composite doubler installation	• 144 K fatigue cycles applied - no fatigue cracks generated
BE-6	Aluminum Control Specimen: unabated 0.5" crack in aluminum plate; plate has no reinforceing doubler	• crack propagated until specimen failed at 9 K cycles • duplicate specimen failed at 12 K cycles
BE-7	Doubler Baseline Specimen: composite doubler installed without any engineered flaws; no flaws in parent aluminum plate	• fatigue testing underway
BE-8	stop-drilled 0.5" sawcut edge crack with collocated impact damage on doubler; 160°F hot, wet conditioned	• fatigue testing underway
BE-9	unabated 0.5" fatigue edge crack with collocated impact damage on doubler; impact damage on edge of doubler; 160°F hot, wet conditioned	• fatigue testing underway

Table 1: Composite Coupon Specimen Matrix And Fatigue Test Summary

2. Fatigue Cracks With No Abatement - Specimens BE-1 and BE-4 survived 144 K fatigue cycles with crack growths of 2" or less. Specimen BE-1 had a good doubler bond along the length of the fatigue crack while specimen BE-4 had the added detriment of a disbond co-located with the fatigue crack (see Figure 1). As a result, the initial stage of crack growth was quicker in specimen BE-4, however, the two crack growth curves blended into a single propagation rate at a crack length (a) equal to 1.75". In fact, Figure 2 shows that in spite of the initial flaw scenario engineered into the test specimen, all of the flaw growth curves tend to blend into the same outcome as the crack propagates beyond 2" in length. This is because all of the specimens have the same configuration at this point.

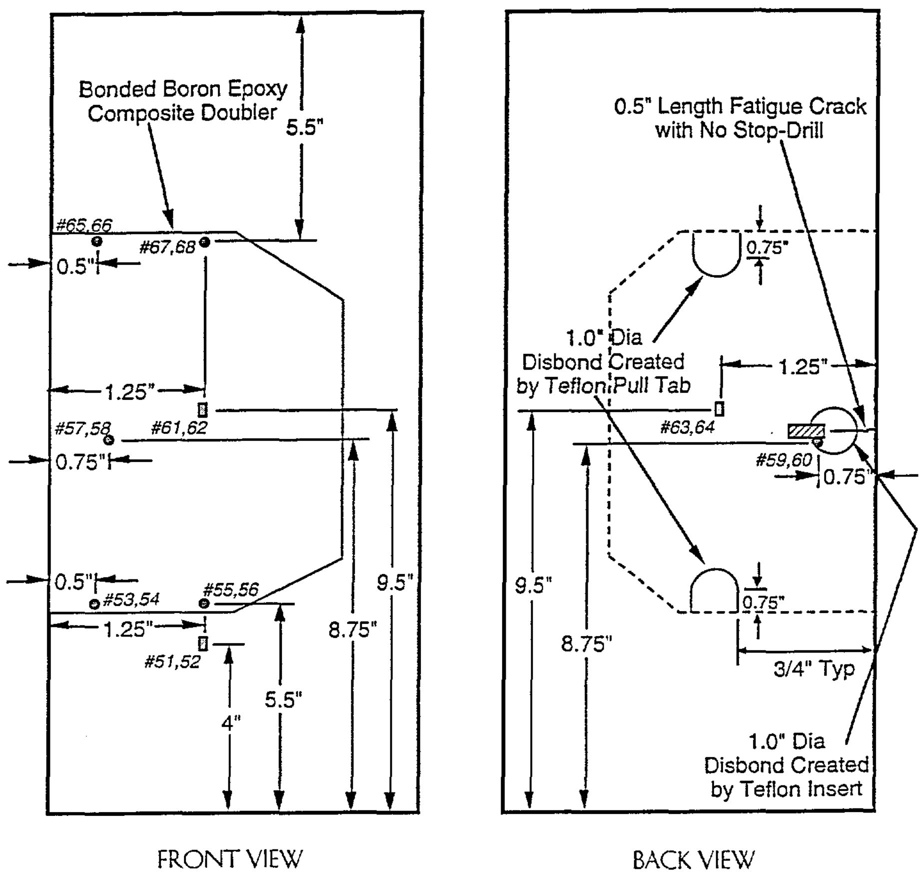

FIGURE 1 : Composite Doubler Fatigue Specimen with an Unabated Crack and Co-Located Disbond (Specimen BE-4); Numbers in Italics Represent Biaxial Strain Gage Channels

3. Material Removed from Parent Plate and Composite Doubler Reinforcement - Specimen BE-5 had a 1" diameter hole simulating the removal of damage (e.g. crack or corrosion) in the parent structure. It could also be considered a stop-drill hole with a very generous radius. A crack did not propagate in this specimen after 144 K fatigue cycles. The bonded composite doubler picked up load immediately adjacent to the cut-out so this type of material removal enhanced the overall performance of the installation.

4. Propagation of Adhesive Disbonds - One of the concerns that has hindered the expansion of composite doubler technology into commercial aviation is the potential for disbonds between the composite doubler and the aluminum skin. It has been shown in related studies that the load transfer region which is key to the doubler's performance is around its

perimeter. The purpose of the disbonds in specimens BE-2 through BE-4 was to demonstrate the capabilities of composite doublers when large disbonds exist in the critical load transfer region as well as around the cracks which the doublers are intended to arrest. Large disbonds of 0.75" and 1.0" diameter were engineered into the test specimens. Inspections performed at 1,2, 3, and 4 fatigue lifetime intervals revealed that there was no growth in any of the disbonds. Comparisons between the BE-1 (no disbonds) and the BE-2, 3, and 4 (engineered disbonds) fatigue curves in Figure 2 show that the large disbonds did not decrease the composite doubler's performance.

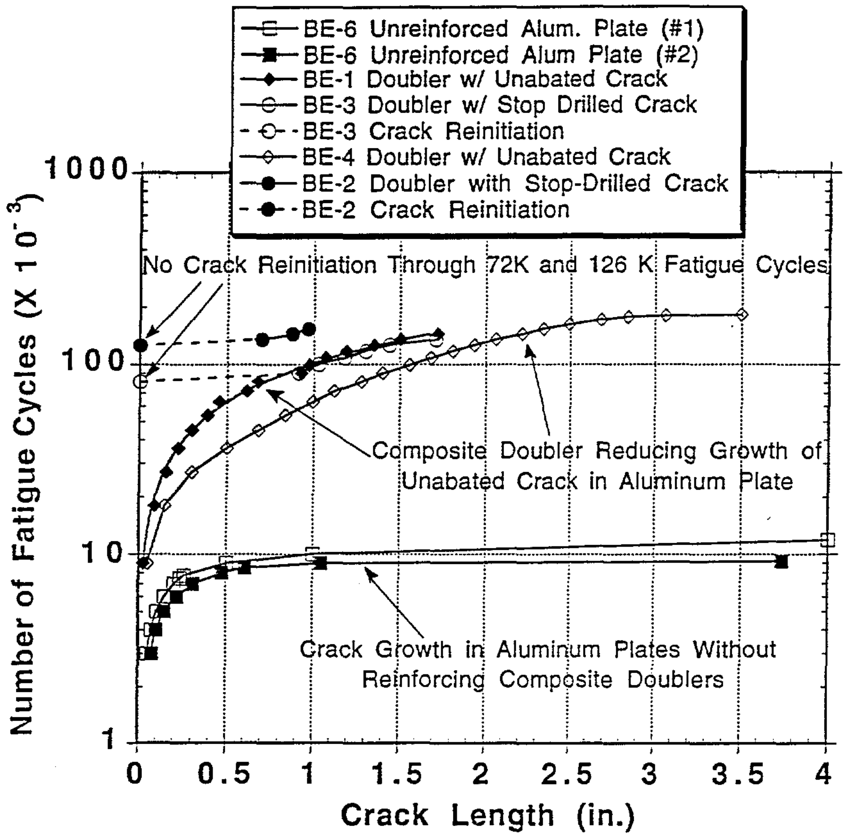

FIGURE 2 : Fatigue Crack Growth in 2024-T3 Plates With and Without Reinforcing Composite Doublers

5. Control Specimens and Comparison of Crack Growth Rates - Two tests were conducted on aluminum control specimens which were not reinforced by composite doublers. In these tests the fatigue cracks propagated through the width of the specimens after 9 K and 12 K cycles. By comparison, specimen BE-4, which had a composite doubler, failed after 182 K cycles. Thus, the fatigue lifetime as defined in the test coupons, was extended by a factor of approximately 20 through the use of composite doublers. In Figure 2, the unreinforced panels asymptotically approach 10 K cycles-to-failure while the plates reinforced by composite doublers asymptotically approach 100 K to 200 K fatigue cycles.

Strain Field Measurements in Damage Tolerance Coupons

The maximum doubler strains were found in the load transfer region around the perimeter (taper region) of the doubler. In all five doubler specimens (BE-1 through BE-5), the strains monitored in this area were 48% - 52% of the total strain in the aluminum plate (e.g. Channel 52 in Figure 1). This value remained constant over four fatigue lifetimes indicating that there was no deterioration in the bond strength. The strain gages were also able to show the effects of disbonds in the installation. For example, gages 56, 58, and 68 registered very little strain since they were mounted over disbonds (see Fig. 1) which produced strain relief in the doubler.

Although the strains remained constant in the critical load transfer region, Figure 3 shows that there were several changes in the strain fields as the fatigue tests progressed. These changes were due to the propagation of the crack through the aluminum plate. At N= 0 cycles, the strains at the center of the doublers amounted to 30% of the total strain in the aluminum plates. At N = 144 K cycles, however, the same strain gages registered 60% to 70% of the total strain in the plate. The N = 0 and N = 144 K cycles curves in Figure 3 show how the doubler picks up more load as the crack propagates and the plate relieves its load. A complimentary load reduction occurs in the aluminum skin; the plate strains are reduced as the crack propagates.

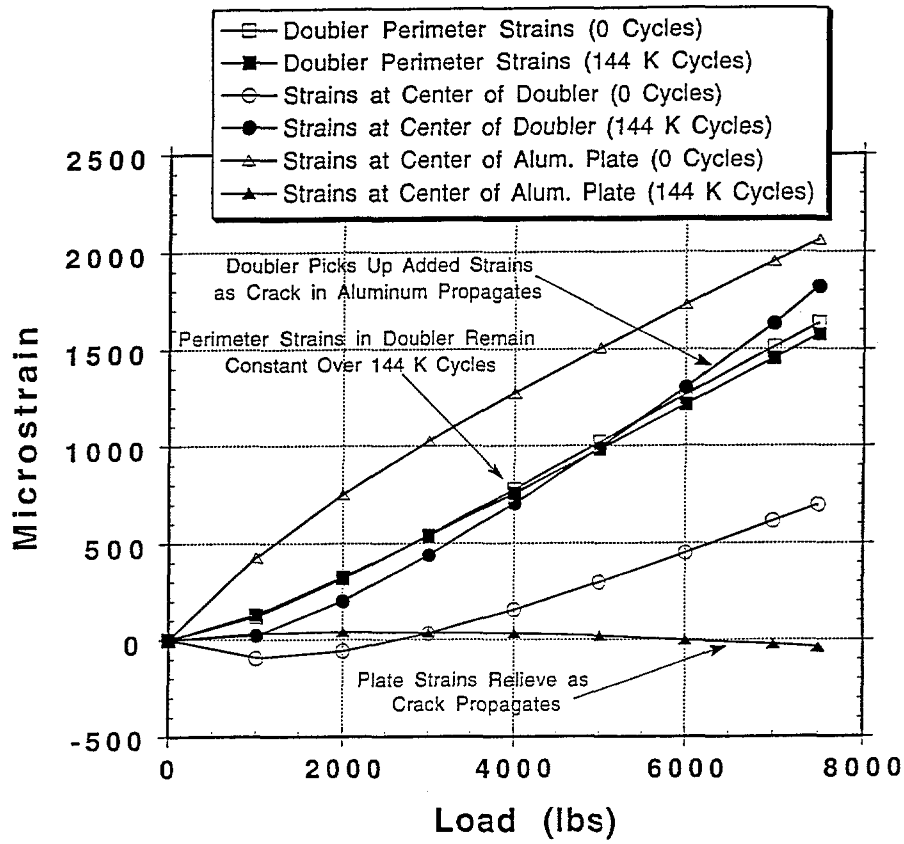

FIGURE 3 : Strain Field Changes as a Result of Crack Propagation (Specimen BE-4)

Static Tension Ultimate Tests - Residual Strength

Two of the specimens which were subjected to 144,000 fatigue cycles (four L-1011 lifetimes) were subsequently tested to determine their static ultimate tensile strength. Since the specimens were tested after cracks were grown, these tests were actually *residual strength tests*. By using the maximum load at failure and the original crossection area at the start of the static ultimate test, the resulting "ultimate tensile strength" numbers should be conservative.

Both specimens, BE-1 and BE-2, had plate crack reinitiation during the course of their fatigue tests. Their failure modes were identical: cohesive bond failure and crack propagation through the aluminum plate. The doubler separated from the aluminum plate through a cohesive fracture of the adhesive. Thus, there was no disbond growth and adhesive was found on both the aluminum and composite laminate. In calculating the ultimate tensile stress, the cross-sectional dimensions of the aluminum and the bonded doubler were used.

1. Specimen BE-1: fatigue testing propagated the unabated crack to 2.25" in length; measured static ultimate tensile strength was 103 KSI.
2. Specimen BE-2: fatigue testing propagated the stop-drilled crack (crack reinitiation at 126,000 cycles) to 1.625" in length; measured static ultimate tensile strength was 88 KSI.

Even in the presence of severe worst case installations (disbonds) and extensive damage growth (fatigue cracks extending through 50% of the specimen width), it was seen that the doubler-reinforced-plates were able to achieve static ultimate tensile strengths in excess of the 70 ksi Mil handbook listing for 2024-T3 material.

SUBSIZE DOOR CORNER TEST ("C" SECTION TEST)

This test specimen was a subsize mock-up of a skin cut-out with a corner reinforced by a composite doubler. The important geometric variables, including the composite doubler footprint and lay-up, matched the configuration to be deployed on the L-1011 aircraft. Figure 4 shows the basic "C" section geometry with one corner, around the door corner radius, containing a bonded, Boron-Epoxy doubler. Specimen geometry and load conditions were engineered to exceed design strains corresponding to the worst-case L-1011 flight condition of fuselage downbending and pressurization. The primary issues to be addressed were doubler delamination and disbond especially in the high peel stress regions.

The aluminum substrate was 0.25" thick 7075-T6 and the composite doubler was 0.41" thick (72 Boron-Epoxy plies). NDI was applied between each of three tests to monitor any flaw growth [8]. The three structural tests conducted were: 1) Limit Load Test to 10,300 lbs., 2) 150% Limit Load Test to 15,600 lbs., and 3) Ultimate Failure Test. Failure was defined as an inability to sustain an increasing load (i.e. a "knee" in load vs. displacement curve).

DOOR CORNER TEST RESULTS

The test specimen survived the first two load scenarios, up to 15,600 lbs. (150% Limit Load) without failure. There were slight nonlinearities observed in the high strain areas and the onset of yielding occurred at 12,900 lbs. Emphasis here will be on ultimate failure tests which experimentally determined the design margin for this doubler.

Strains in the Aluminum Substrate

Figure 5 shows the strain gage locations used to monitor the composite and aluminum material responses. During the Ultimate Failure Test, several areas in the aluminum plate yielded and exhibited nonlinear strain behavior. Figure 6 shows that channels 5 (45°), 7, 8 (both 67°) and 16 (90°) all exceeded material yield levels. Note that the onset of yielding during the 150% Limit Load Test was found to be at approximately 5700 $\mu\varepsilon$ while in this test, yielding began at approximately 7000 $\mu\varepsilon$. This is because the material was strain hardened during the 150% Limit Load Test. Subsequent yielding did not occur until the strain levels corresponding to the previous 15,600 lb. test load were exceeded. This phenomena is clearly labeled in Figure 6.

Strains in the Composite Doubler

The strain field in the composite doubler was similar to that of the parent aluminum plate with nonlinear behavior noted in the gages located around the radius of the door cut-out (Channels 6 and 9). Figure 7 shows the strains measured in the Boron-Epoxy doubler up to failure of the test article. Since Boron-Epoxy material does not normally exhibit the nonlinear stress-strain response or yielding as shown in the figure, the nonlinearities in the curves probably stem from the yielding in the aluminum substrate and the corresponding load transfer into the composite doubler. This conclusion is supported by the fact that the onset of the composite's nonlinear response (at the 15,600 lbs., 150% Limit Load level) matches the one noted for the aluminum plate.

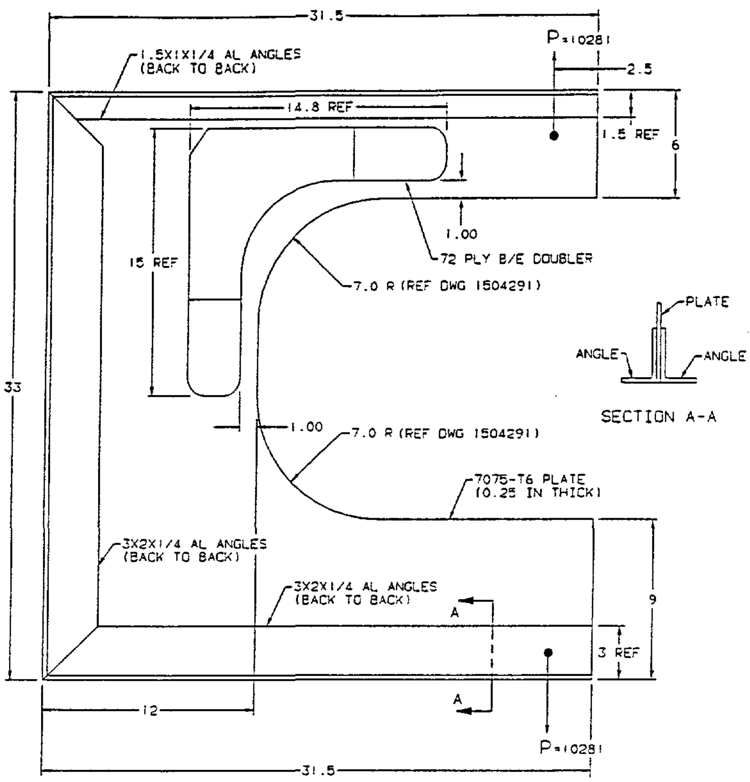

FIGURE 4 : Subsize Door Corner ("C" Section) Test Article

The largest strain occurred at the 67° location around the doubler's radius (Channel 9) and had a maximum value of 6979 $\mu\varepsilon$ at 25,060 lbs. load. Table 2 shows that the maximum strains in the composite are roughly 35-50% of the maximum strains in the adjacent aluminum plate. This result agrees with the load transfer values observed in the Damage Tolerance tests. Strain levels along the perimeter of the door corner (Channel 9) were much larger than the strains at the center of the doubler (-2,000 $\mu\varepsilon$; Channel 13). This supports the design assumption and analyses for bonded doublers which indicate that the load transfer takes place in the perimeter (first 1" - 2" around edge) of the doubler's footprint. Thus, disbonds have their greatest effect in this area and are less detrimental when present in the "interior" of the doubler's footprint.

Channel	Location	Maximum Strain at 25,060 lbs.	Percent of Strain from Adjacent Alum. Plate
6	45° on Corner - Composite	3,635 $\mu\varepsilon$	35.6% (vs. Ch. 5)
9	67° on Corner - Composite	6,979 $\mu\varepsilon$	49.3% (vs. Ch. 8)
18	Horizontal Leg - Composite	2,169 $\mu\varepsilon$	34.2% (vs. Ch. 17)

TABLE 2: Summary of Maximum Strains Measured in the Composite Doubler for the Ultimate Failure Test

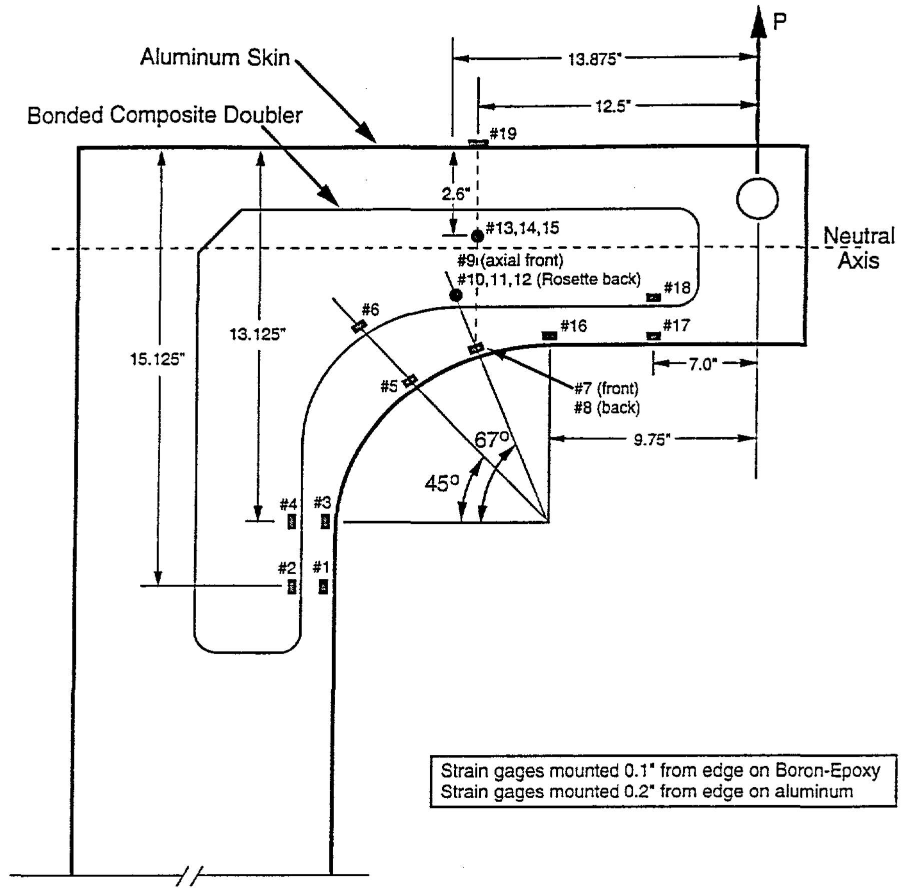

FIGURE 5 : Strain Gage Locations for Subsize Door Corner Test

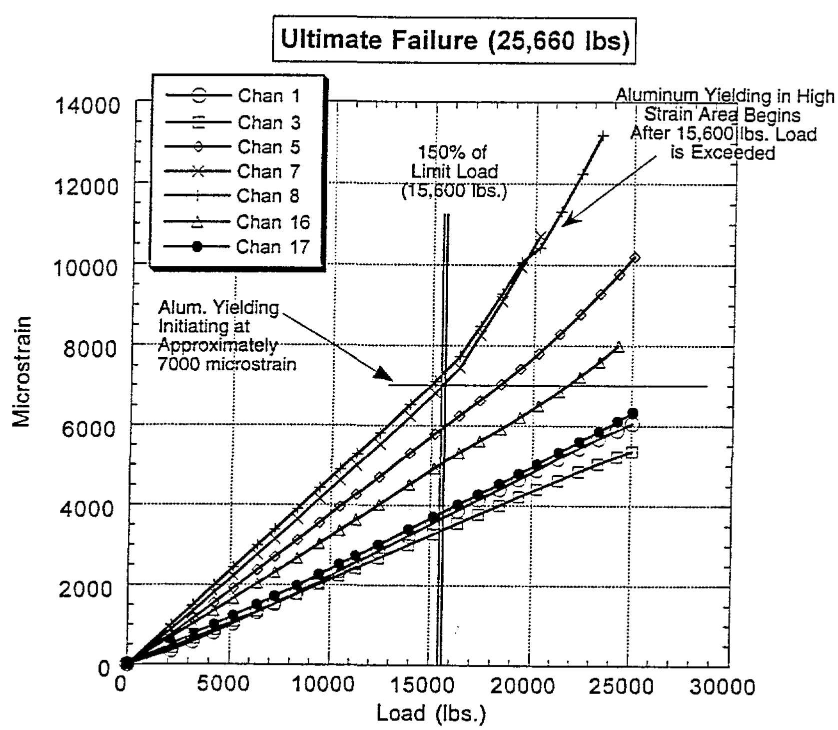

FIGURE 6 : Strains in the Aluminum Substrate Up to Specimen Failure

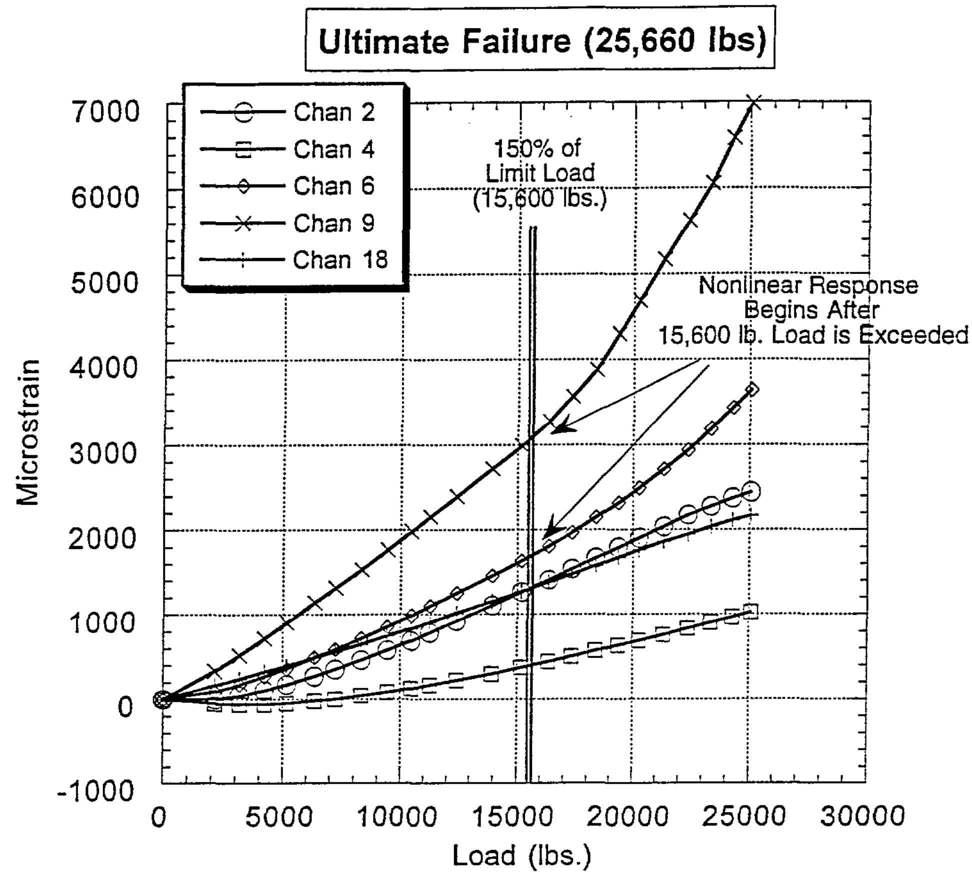

FIGURE 7 : Strains in the Composite Doubler Up to Specimen Failure

Specimen Ultimate Failure

The Subsize Door Corner Test Article withstood a maximum load, or failure load, of 25,660 lbs. The failure load was 2.5 times the limit load for this composite application. Thus, the safety factor for this doubler design exceeded the minimum goal of 1.5.

The test specimen's failure mode can be described as follows. As the load was increased beyond the 1.5 Limit Load (15,600 lbs.) mark, the aluminum substrate experienced significant yielding. This, in turn, accelerated the load transfer into the composite doubler and produced nonlinear behavior in the doubler's strain field. A disbond, between the composite laminate and the aluminum substrate, began to grow at the point of maximum strain: 67° around the door corner radius. The flaw in the adhesive layer propagated along the horizontal leg of the doubler and around the door corner. Also, a second phenomenon occurred simultaneous with the disbond growth. Because the doubler created a shift in the neutral axis, secondary (in-plane) bending was induced throughout the test. The secondary bending produced out-of-plane deformations, or bowing, of the test specimen. This out-of-plane deformation, coupled with the growing disbond, created a global loss in the structures ability to carry the test load. During specimen failure, much of the load was transferred to the composite doubler. Since the doubler now had a substantial area in which it was unsupported by the aluminum (area of disbond), the composite material fractured along the line of maximum strain.

Comparisons with Finite Element Model

A finite element model (FEM) was produced for the Subsize Door Corner test article in order to: 1) determine if and when strain levels could be achieved to simulate design loads from fuselage downbending and internal pressure, 2) establish the location for critical strain gages, and 3) compare analytical and experimental results for the structure's response up to failure.

Figure 8 compares the strain gage measurements with the strain field predicted by the FEM. Analytical strains in both the aluminum substrate and composite doubler compare quite well with the experimentally measured strains. The FEM predictions were particularly good in the area of greatest concern around the door corner radius (Channels 5, 7, 8, 9). Table 3 summarizes the comparison between the FEM analytical predictions and the strains measured during the load tests. The comparisons are made for the maximum load (13,000 lbs.) prior to yielding in the aluminum plate.

Channel	Finite Element Model Analytical Strains	Strain Gage Measurements Experimental Strains	Percent Difference
5	5,311 με	5,067 με	4.8 %
7	6,153 με	6,045 με	1.8 %
8	6,153 με	6,305 με	2.4 %
9	2,912 με	2,595 με	12.2 %
16	5,225 με	4,215 με	24 %
19	-2,874 με	-3,351 με	14.2 %

TABLE 3: Summary of Comparison Between Finite Element Model and Experimental Strains at P = 13,000 lbs.

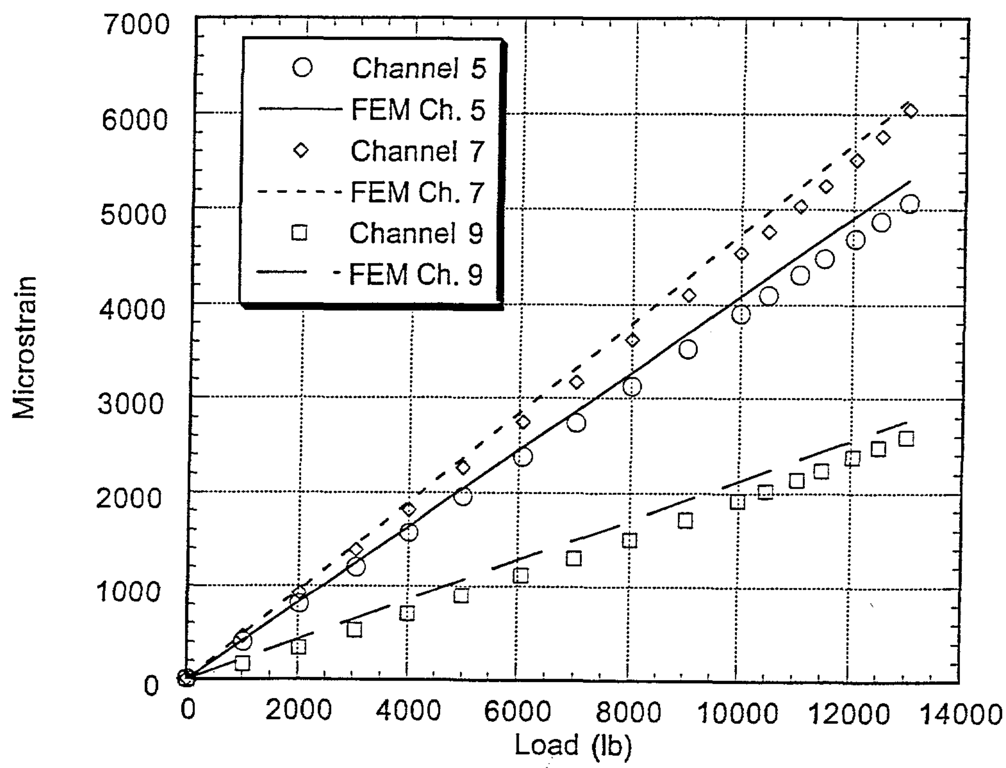

FIGURE 8 : Comparison Between Aluminum/Composite Strains and Values Predicted by the Finite Element Model

FOUR POINT BENDING TESTS

In this test series, the specimen was a beam comprised of a 72 ply Boron-Epoxy laminate (lay-up as per the Lockheed design drawing) bonded to an aluminum substrate. The beam was 13" long and 1" wide. The composite laminate was 0.41" thick and the 7075-T6 aluminum plate was 0.25" thick. This configuration matches the installation on the L-1011 aircraft. Figure 9 shows the test specimen. The purpose of this test was to experimentally assess the potential for delaminations and disbonds in the L-1011 composite installation when subjected to pure bending stresses. Because bending of the fuselage skin is an important consideration in the door corner installation, this test focuses on worst case conditions for pure bending loads. The beam bending flexure test was modeled after the ASTM 53-600, method 330 for four point bending. The load was uniformly and continuously increased until structural failure occurred.

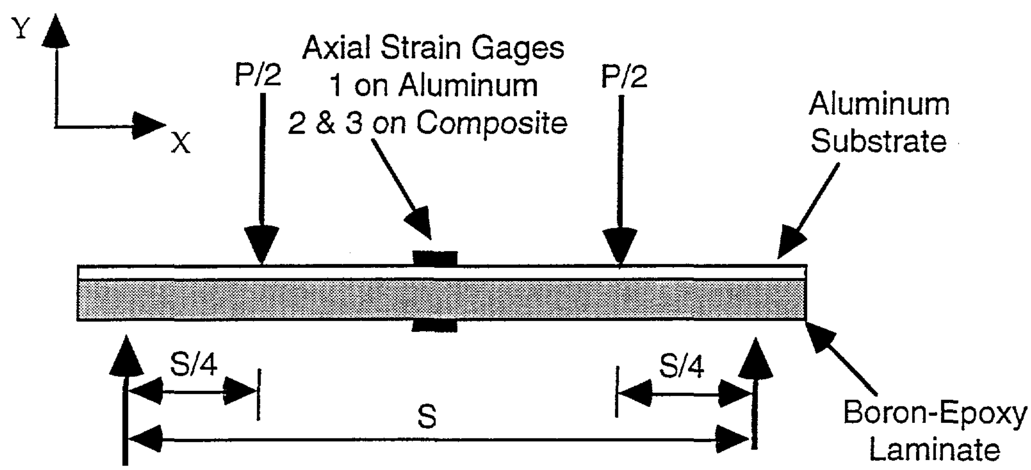

FIGURE 9: Four Point Bending Test Specimen

Strains and Failure in the Four Point Bending Test Specimen

The four point bending specimen was tested with the composite laminate on the bottom. Thus, the plies in the laminate experienced tension loads. Figure 10 plots the strains measured in the specimen. The strains at the top (aluminum) and bottom (composite), although opposite in direction, are equal in magnitude. The maximum strains reached 8,000 $\mu\varepsilon$ before the specimen failed at P = 4,058 lbs. These strains were larger than the strains experienced by the Subsize Door Corner Test Article during its 150% Limit Load Test. Thus, the resistance to delamination for this bending direction exceeds the desired design margins for the door installation. The strain responses were linear up through failure indicating that there was no general yielding in the beam.

The failure mode was interply delamination, at assorted locations, followed immediately by fracture of the laminate. The laminate fractured at the center of the beam, at the point of maximum flexure, and below one of the load application points. In order to demonstrate repeatability, a similar specimen was tested. All aspects of the test - failure load and mode - were similar. The maximum strains reached 8,000 $\mu\varepsilon$ before the second specimen failed at P = 4,300 lbs.

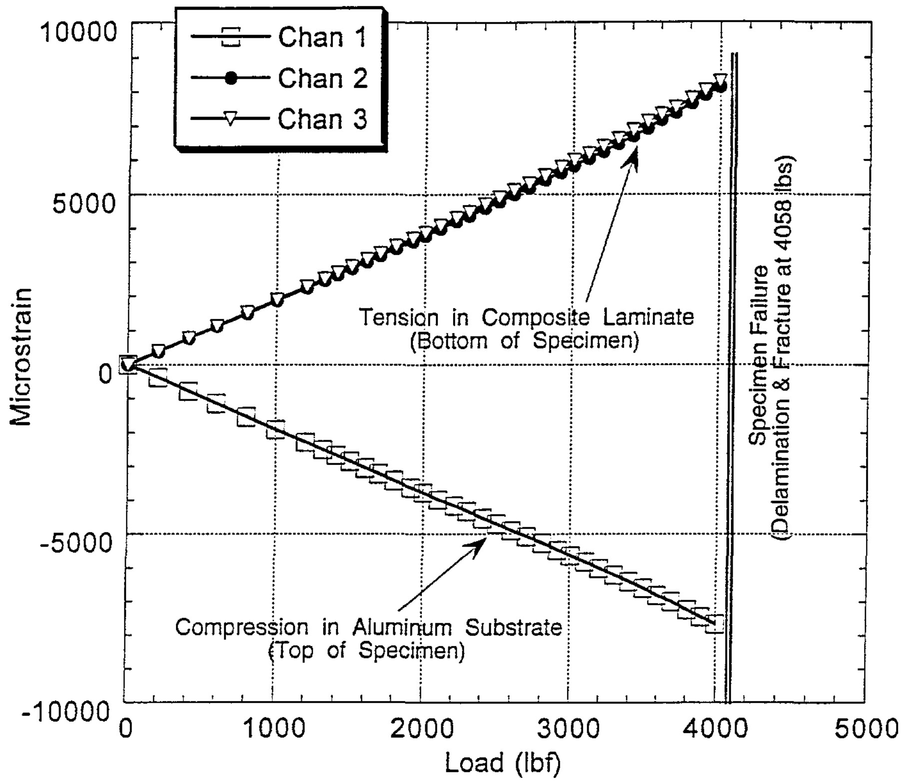

FIGURE 10 : Bending Strains for Composite Laminate -Aluminum Specimen

CONCLUSIONS

At this time, bonded composite doublers have not been certified for use on the U.S. commercial aircraft fleet. Most of the concerns surrounding composite doubler technology pertain to long-term survivability, especially in the presence of non-optimum installations, and the validation of appropriate inspection procedures. The program presented here intends to introduce composite doubler technology to the U.S. commercial aircraft fleet after resolving any remaining uncertainties.

The structural tests recently completed by the AANC were designed to supplement the current database on composite doubler performance by focusing on a particular commercial aircraft application. These tests validated the Boron-Epoxy doubler design for the L-1011 passenger door corner reinforcement. Worst case load conditions were applied to support Lockheed's analysis which predicted that the doubler would be able to survive the peel and bending stresses induced in this installation. The test articles all survived strain levels which exceeded the design allowable values. Strain gage results on the Subsize Door Corner specimen indicated a large plastic region in the corner of the aluminum plate. This plastic zone caused the strain rate in the doubler to increase. Thus, the bond is able to transfer plastic strains into the doubler. It also demonstrated that the aluminum skin must experience yield strains before any damage to the doubler will occur.

Damage tolerance tests established the performance of boron/epoxy composite doublers in the presence of compounding flaw scenarios. <u>Fatigue:</u> In this test series, relatively severe installation flaws were engineered into the test specimens in order to evaluate boron/epoxy doubler performance under worst case, off-design conditions. 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 was able to help the structure survive more than four design lifetimes of fatigue loading. Comparisons with control specimens which did not have composite doubler reinforcement showed that the fatigue lifetime was extended by a factor of 20. Adhesive Disbonds: Despite the fact that the disbonds were placed above fatigue cracks and in critical load transfer areas, it was observed that there was no growth in the disbonds over four fatigue lifetimes. Further, it was demonstrated that the large disbonds, representing almost 30% of the axial load transfer perimeter, did not decrease the overall composite doubler performance. Strain Fields: The maximum doubler strains were found in the load transfer region around the perimeter (taper region) of the doubler. 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. Residual Strength: The presence of disbonds and fatigue cracks did not prevent the doubler-reinforced-plates from achieving static ultimate tensile strengths in excess of the 70 ksi Mil handbook listing for 2024-T3 material. Thus, the composite doubler was able to restore the structure to its original load carrying capability.

All tests were performed in extreme combinations of flaw scenarios (sizes and combinations) and excessive fatigue load spectrums so performance parameters presented here were arrived at in a conservative manner. A companion publication from the AANC will discuss nondestructive inspections of boron/epoxy composite doubler installations in light of the damage tolerance observed in this study.

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