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PERFORMANCE ANALYSIS OF BONDED COMPOSITE DOUBLERS ON AIRCRAFT STRUCTURES

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ABSTRACT

Researchers in both private and government agencies contend that composite repairs (or structural reinforcement doublers) offer numerous advantages over metallic patches including corrosion resistance, light weight, high strength, elimination of rivets, and time savings in installation. Their use in commercial aviation has been stifled by uncertainties surrounding their application, subsequent inspection and long-term endurance. Because of the rapidly increasing use of composites on commercial airplanes, coupled with the potential for economic savings associated with their use in aircraft structures, it appears that the demand for composite patching procedures will increase.

The process of repairing or reinforcing airplane structures is time consuming and the design is dependent upon an accompanying stress and fatigue analysis. A repair that is too stiff may result in a loss of fatigue life, continued growth of the crack being repaired, and the initiation of a new flaw in the undesirable high stress field around the patch. Uncertainties in load spectrums used to design repairs exacerbates these problems as does the use of rivets to apply conventional doublers. Many of these repair or structural reinforcement difficulties can be addressed through the use of composite doublers; however, any remaining unknowns associated with the use of this material must be eliminated. Primary among these unknown entities are the effects of non-optimum installations and the certification of adequate inspection procedures.

This paper presents on overview of a program intended to introduce composite doubler technology to the U.S. commercial aircraft fleet. In this project, a specific composite application has been chosen on an L-1011 aircraft in order to focus the tasks on application and operation issues. Through the use of laboratory test structures and flight demonstrations on an in-service L-1011 airplane, this study is investigating composite doubler design, fabrication, installation, structural integrity, and non-destructive evaluation. In addition to providing an overview of the L-1011 project, this paper focuses on a series of fatigue and strength tests which have been conducted in order to study the damage tolerance of composite doublers. Test results to-date are presented.

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BACKGROUND

The Aging Aircraft NDI Validation Center (AANC) was established by the Federal Aviation Administration Technical Center (FAATC) at Sandia National Laboratories in August of

DISCLAIMER

1991. The goal of the AANC is to support development and validation of inspection technologies for aging commuter and transport aircraft. The Center also supports other activities related to general airworthiness assurance and improved maintenance procedures. 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 doublers 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. 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 [1-4].

This FAA-based project is designed to compliment the existing data on composite doublers in order to produce guidance which will assure the airworthiness of composite repairs and structural reinforcements. It introduces structural reinforcement concepts to the L-1011 fleet using structurally bonded, high modulus boron/epoxy composites. The parties participating in this project are the AANC (Sandia National Labs), Lockheed Aeronautical Systems, Delta Air Lines, Warner Robins Air Force Base, Textron Specialty Materials, Inc., and the FAA.

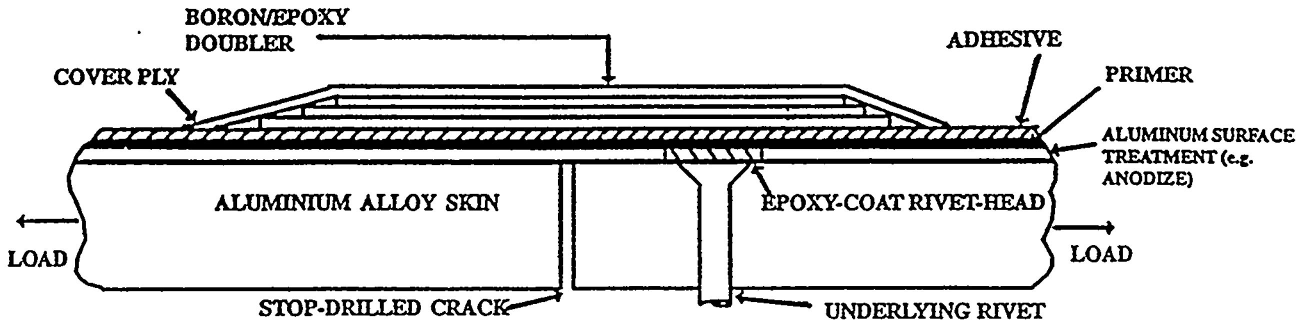
Advantages of Composite Doublers in Light of Structural Integrity Concerns

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. For a cracked structure, a bonded repair significantly reduces the stress intensity factor and, as a result, may reduce crack growth. It has been shown that in many cases, crack growth can be completely eliminated [4]. Figure 1 shows a typical bonded composite doubler repair over a cracked parent aluminum structure.

At present, there is a concern that when repairing multi-site or wide-spread damage using conventional methods, the close proximity of a large number of mechanically fastened repairs may lead to a compromise in the damage tolerance of the structure. Numerous articles have addressed the myriad of concerns associated with repairing aircraft structures. Reference [5] presents the results of a program which demonstrated the successful application of externally bonded composite repairs while Ref. [6] 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.

The trend toward operating aircraft approaching or exceeding their original design life has been reflected in an increased number of structurally significant defects. Corrosion



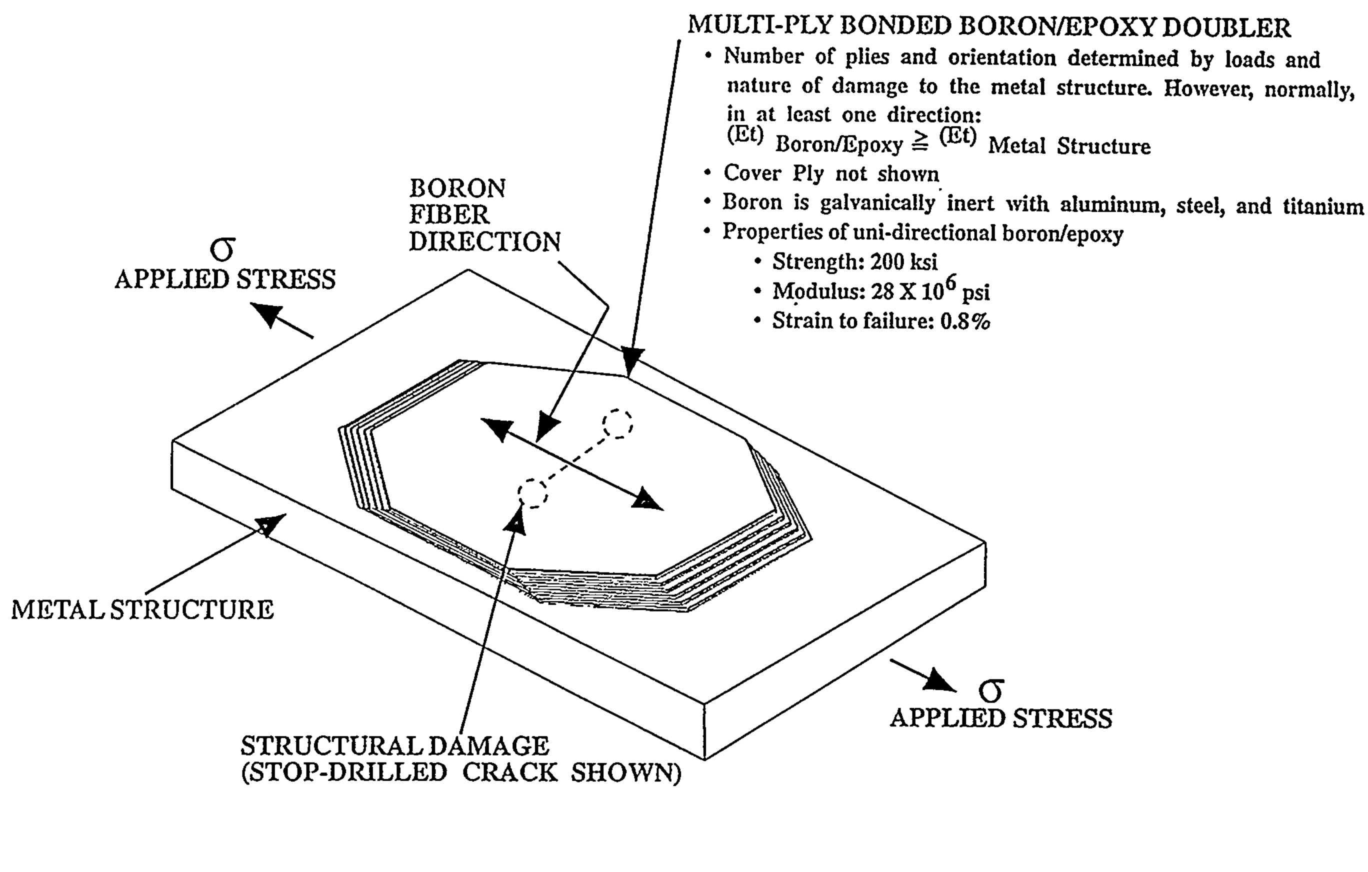


FIGURE 1 : Example of Bonded Composite Doubler Installation on Cracked Aluminum

damage and subsequent fatigue damage in fuselage lap joints, pressure bulkheads, and control surfaces are several common problems. This trend, coupled with the rapid increase in the use of composites in airplane structures [7], indicates a definite need to address any remaining uncertainties and to formally certify (FAA certification for commercial use) composite doubler installation and inspection processes.

Related Work

Over the past 20 years, military applications have demonstrated the success of composite doubler installations [8-11]. There are currently over 6,000 boron/epoxy doublers flying on U.S. and Australian military aircraft; some have been in operation since the mid-1970's [1, 10]. Composite repair concepts have been demonstrated on 767 (keel beam), MD-80 (trailing edge flap), C-130, C-141 (wing plank), Mirage III (wing skin) and F-111 (wing pivot fitting) aircraft structure. In the U.S., commercial applications have been limited to demonstration installations ("decals") over undamaged structure [3].

Wright-Patterson Air Force Base, in conjunction with the University of Dayton, is involved in an ongoing study of composite repairs to aluminum structures [12-13]. The work addresses various aspects of composite repairs including surface preparation methods, primer evaluation, on-aircraft adhesive curing, structural analysis, and inspection. The effects of process parameters and process deviations on bonded joints are a major thrust of the current activities.

Similarly, there is a wealth of information available regarding the fatigue, strength, and mechanical properties of boron doublers bonded to parent aluminum structure. References [14-16] address studies directed toward military applications while reference [17] provides much of the foundation for the fatigue tests described here. Reference [16] describes a program which quantified the benefits of composite doubler repairs on cracked metallic aircraft structures. This investigation used experimental fatigue tests and analytical modeling to determine stress intensity factors for repaired cracks and to generally assess the fatigue life of typical repair geometries. Warner Robins is currently using boron/epoxy doublers to repair fatigue cracks in C-141 wings. Researchers at Warner Robins have accumulated extensive strength and installation data in addition to the significant successful flight hours on these composite repairs.

References [3], [17] and [18] present the results from a study undertaken by Textron Specialty Materials, the Boeing Company, and Federal Express. They describe the 25 "decal" doublers which were installed in 1993 on two 747 aircraft. The aircraft are part of the Federal Express fleet and are used for normal air cargo operations. Although the doublers were installed over undamaged structure, they are load carrying structural members. The composite doublers were applied to leading edge, fuselage skin, wheel well bulkhead, and wing flap locations. This project continues to acquire flight service data in order to demonstrate the installation process and to evaluate the ability of the doublers to withstand actual flight loads. To date, all 25 doublers are successfully operating after over two years of service.

As a result of this experience, there is a technology base with respect to repair materials, repair processes, tooling concepts and analytical approaches. Certain repair patch design guidelines addressing thickness, fiber orientation, size, and edge taper have been established. The particular tests conducted in this program were designed to supplement the current database referenced above by focusing on commercial aircraft structural configurations. The AANC tests also employed severe worst-case scenarios in order to measure the damage tolerance of this technology. Nondestructive inspection (NDI) was a constant partner in all of the structural testing so that issues such as inspectability and quality assurance could be addressed. NDI of bonded composite doublers will not be discussed here but will be the subject of a future publication.

PROJECT DESCRIPTION - Application of Composite Doubler on L-1011 Aircraft

The goal of this project is to evaluate the viability of composite doublers as a reliable and cost-effective structural reinforcement method for the civilian fleet and to assist the FAA in developing guidance which assures the continued airworthiness of such doublers.

While there have been numerous studies and military aircraft installations of composite doublers, certain information gaps and a general resistance to new technology on the part of the aviation industry has suppressed the extension of composite doublers into commercial applications. The proof-of-concept project presented here is intended to remove any remaining obstacles to accepted use. By focusing on a specific commercial aircraft application - reinforcement of the L-1011 door frame - and encompassing all "cradle-to-grave" tasks (i.e. design, analysis, installation, and inspection), this program is designed to prove the capabilities of composite doublers. Figure 2 shows the structure around the passenger door as well as an L-1011 fuselage which was cut to produce door frame test beds for this study. The reinforcement doubler is placed around both upper and lower door corners. Due to fuselage bending considerations, the exact doubler location depends on whether the door is forward or aft of the wing. The successful execution of this study may be an initial step leading to other similar applications and, with direct FAA involvement, certification and expanded use of this technology. Follow-on plans include the expansion of this effort to include repair scenarios (i.e. damaged parent material).

To demonstrate the capabilities of composite doubler reinforcement technology in an area of known fatigue cracking, this project contains the following technical activities: 1) structural design of the doubler, 2) development of doubler installation procedures, 3) structural evaluation of the design, 4) inspection procedures, and 5) laboratory and flight tests of a composite doubler installed on an operating aircraft. The general issues which will be addressed are:

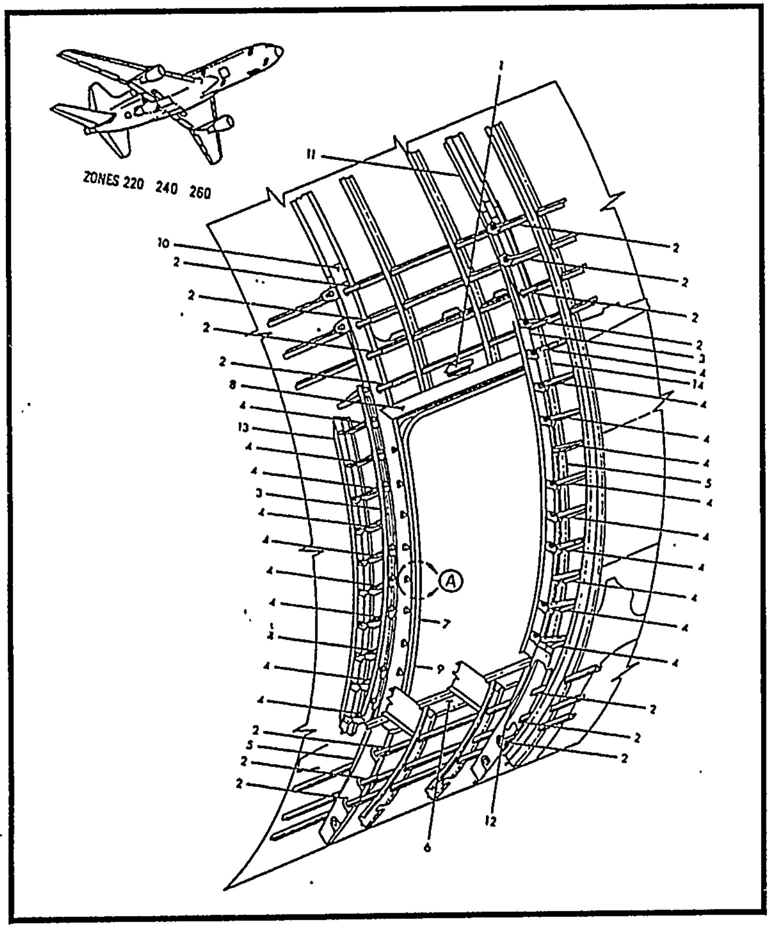
A. Patch design - strength, durability and reliability issues, flaw containment, optimum adhesive properties, and critical patch parameters.
B. Patch installation - (e.g. surface preparation, tooling, heat sinks, effect of underlying rivets, field work)
C. NDI techniques used to qualify and accept an initial installation and to perform periodic inspections. The NDI equipment will be required to inspect for flaws at three different structural levels: 1) in the parent material (crack or corrosion growth), 2) in the adhesive bond (debonds), and 3) in the composite doubler (delaminations).

The overall distribution of tasks among the team participants is listed below. These tasks, along with the flow of activities are summarized in Figure 3.

1. <u>Composite Patch Design and Analysis (Lockheed)</u> - The patch will be designed in accordance with the L-1011 door frame thickness and physical configuration. The composite ply drop-offs, which determine the doubler edge tapers, will be determined as will be the boron/epoxy laminate ply orientations, the doubler geometry, and the

- trial doubler installations
- strain survey

- validation of FEM's
- NDI development



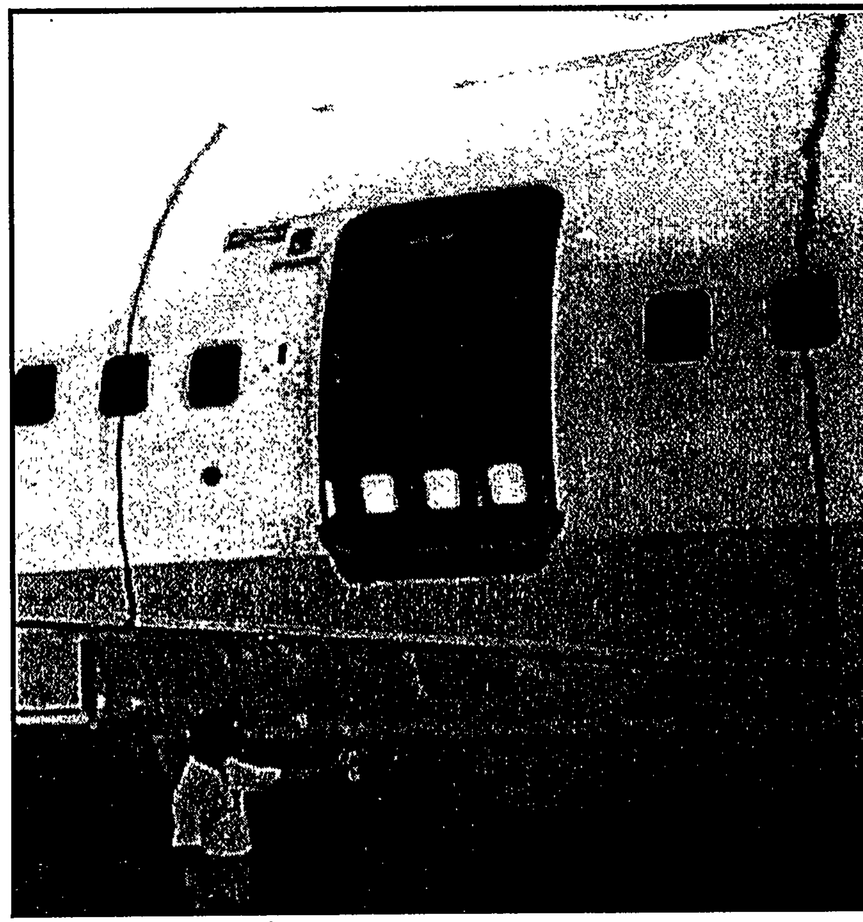


Figure 2. L-1011 Door Frame Test Structure

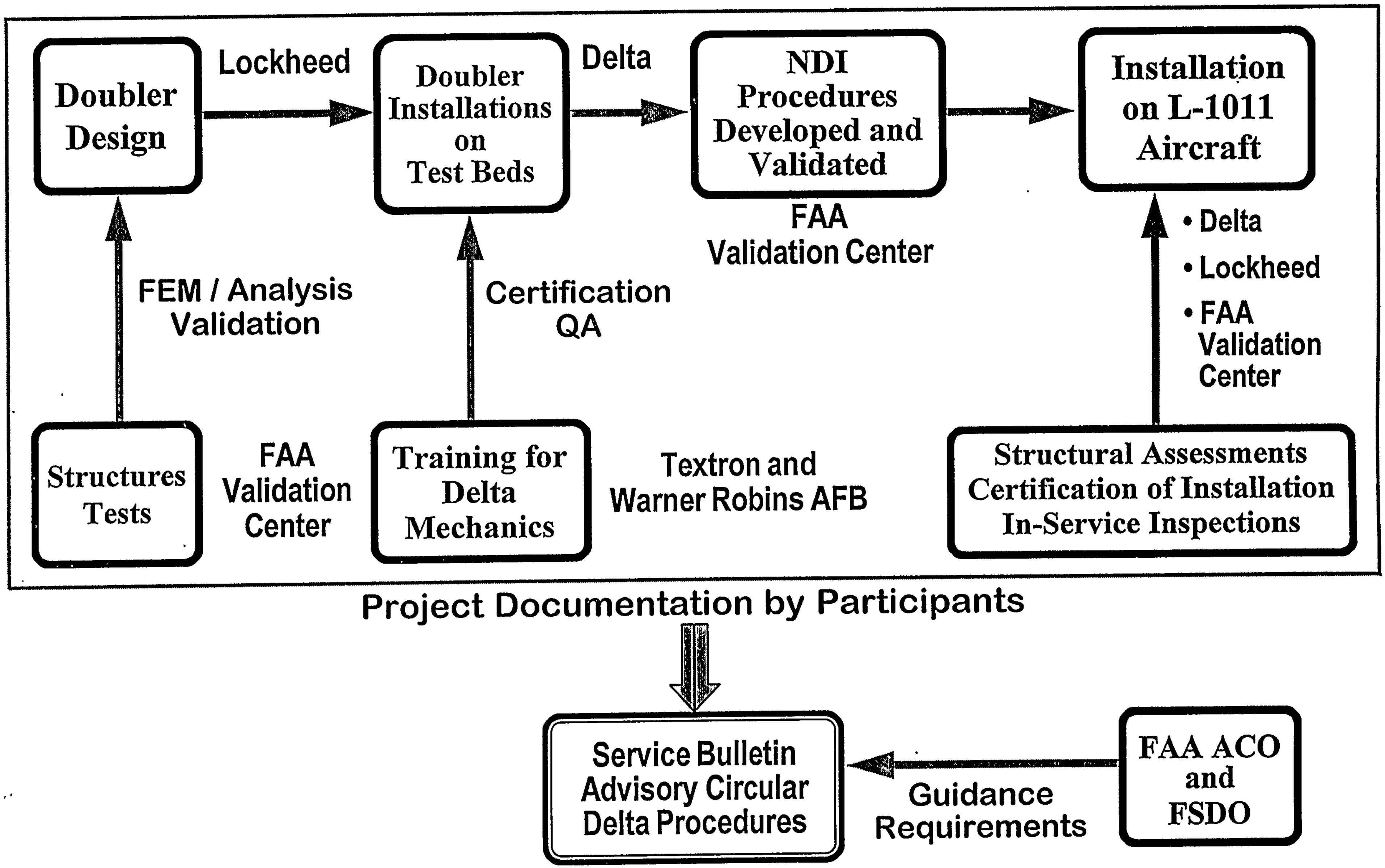


FIGURE 3 : Activity Summary for L-1011 Composite Doubler Project

final doubler location on the door frame corner. Finite Element Models (FEM) will be used to predict the internal stresses in the doubler and the surrounding structure. A damage tolerance analysis will also be performed beneath and adjacent to the doubler to establish inspection intervals. Typical flight load spectra will be used by Lockheed for the analysis.

2. *Installation Process and Material Properties (Textron and Warner Robins Air Force Base)* - Textron and Warner Robins AFB will conduct training sessions - addressing composite doubler installations in general and unique L-1011 door frame requirements - at Delta Airlines' maintenance facility. The training will also include a certification process for Delta mechanics so that they can perform independent installations of composite doublers in the future. In addition to the phosphoric acid surface preparation utilized by Textron, participation by the Air Force allows the silane surface preparation technique to also be transferred to commercial industry (see doubler installation discussion below). Another aspect of this project involves the addition of boron/epoxy material properties into Mil-Handbook 17. Testing conducted by Textron and Wright Patterson and Warner Robins Air Force Bases will provide the necessary material data.

3. *Structural Verification Testing, NDI Development, and Overall Project Management (Sandia Labs AANC)* - The structural integrity and NDI issues will be supported by subjecting laboratory test coupons and full-scale aircraft components to controlled fatigue/load tests. The fatigue and static tests on small coupon test specimens will study flaw initiation and growth, strain fields and load transfer. They will be followed by tests on an L-1011 aircraft structure which contains the selected passenger door application. The test sample, cut from an L-1011 aircraft, is approximately 14' H X 12' W and includes all of the substructure elements (see Fig. 2). The door frame will be subjected to a combined load environment of internal pressure (hoop stress) and axial or longitudinal stress. Figure 4 shows the multi-axis load test bed for the L-1011 door frame. The applied biaxial tension loads will approximate the stresses induced by normal flight pressure loads. The tests on this full-scale test bed will address both patch strength and application concerns. They will also validate the stresses and deflections predicted by the Finite Element Models (FEMs). The testing phase will conclude with the installation of a composite doubler on an operating L-1011 aircraft in Delta's fleet. This installation will be followed by an instrumented flight test. Inspection procedures for the application of conventional NDI - ultrasonic bond tester and eddy current - will be developed and validated. Both NDI and strain field monitoring will be integrated into the flight test program.

4. *Composite Doubler Installations, Aircraft Testing, and Process Specifications (Delta Airlines)* - After receiving proper training, Delta mechanics will install the composite doublers on the L-1011 door frame test bed and the L-1011 in Delta's fleet. Other activities to be carried out by Delta personnel include: 1) quality control of the entire process, 2) application of NDI procedures developed by AANC (see item 3 above), 3) support for ground fuselage pressure tests and follow-on flight test of L-1011 aircraft, and 4) production of engineering process specifications, using information provided by all participants. These process specifications will be presented to the FAA in a request for an alternate means of compliance (i.e. certification of composite doubler for door frame application).

5. *Oversight and Certification (FAA Aircraft Certification Office)* - The FAA Aircraft Certification Office (Atlanta office supporting Delta Airlines) is participating in all of the

project activities in order to provide sufficient oversight and guidance to team members. In this manner, each task is reviewed and approved in real time. The resulting process specification submitted by Delta, in preparation for the on-aircraft doubler installation, can then be processed in a minimum amount of time.

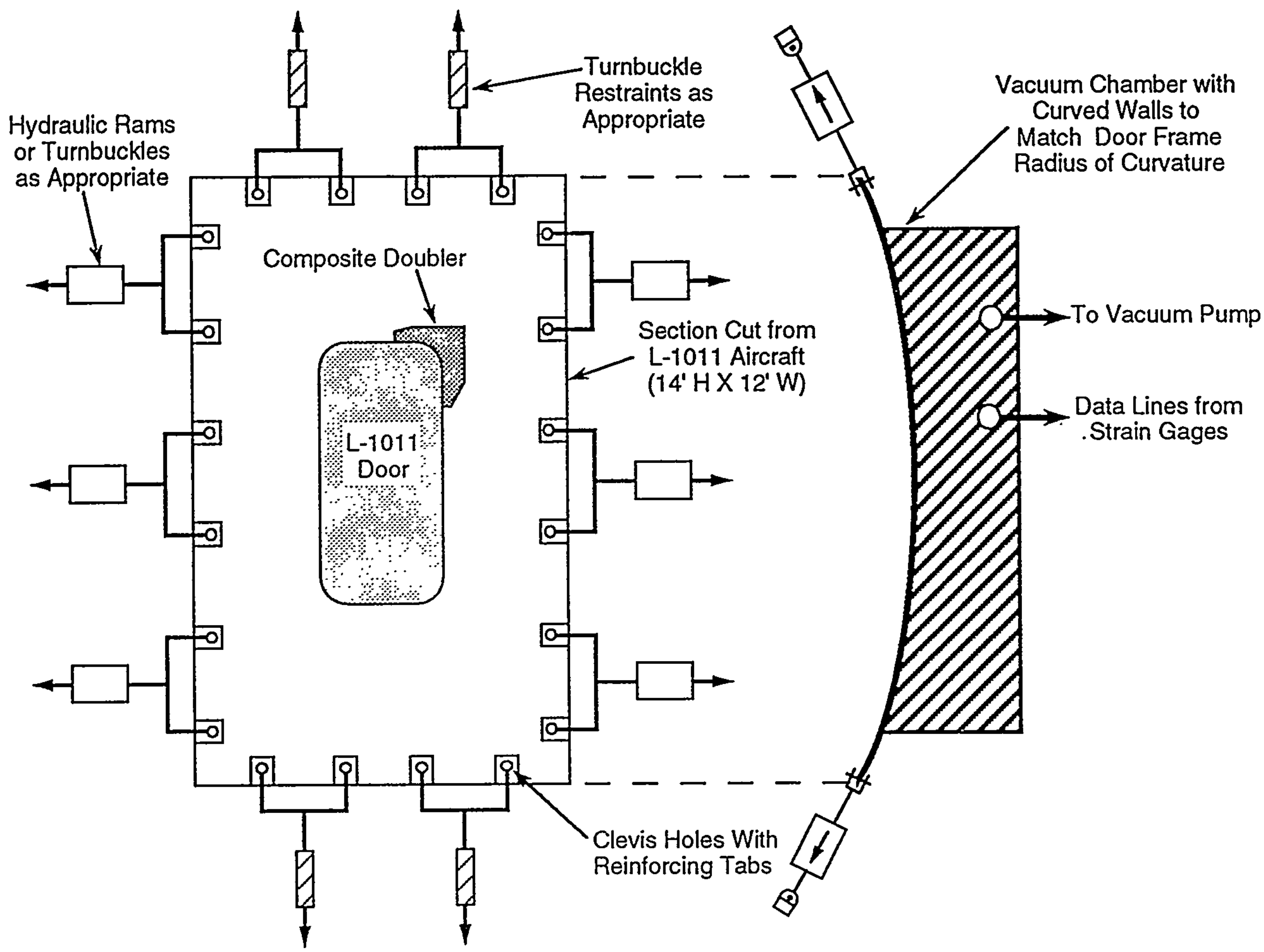


FIGURE 4 : Biaxial Test Facility to Provide Combined Cabin Pressure and Axial Load to the L-1011 Door Frame Test Bed

EXECUTION OF AANC FATIGUE AND STATIC ULTIMATE TESTS

Reference [17] describes fatigue and static ultimate tests which were conducted by Boeing on boron/epoxy doublers bonded to 7075-T6 aluminum plate. In these tests, an array of design parameters, including various flaw scenarios, the effects of surface impact, and other "off-design" conditions, were studied. It should be apparent to the reader that the fatigue and strength tests being conducted by the Sandia Labs AANC are an adjunct to the data obtained in the Ref. [17] study. A synopsis of the Boeing/Textron test results is provided in Appendix A so that the data from the AANC can be presented in the proper context.

Test Objectives - This test series utilized small-scale fatigue specimens to assess the strength and stability of composite doublers bonded to aluminum skin. 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. Nondestructive Inspections (NDI) were interjected throughout the test series in order to evaluate the reliability and limitations of various techniques.

Basic Specimen Design - The test series includes seven different specimen configurations. Each specimen consists of an aluminum "parent" plate, representing the original aircraft skin, with a bonded composite doubler. The doubler is 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, and disbond combinations. Figure 5 shows the most severe flaw scenario (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 in accordance with the L-1011 fuselage skin around the door frame. The specimens were designed for an 4" W X 14" L area of interest. To accommodate the end grips, the final specimen lengths were 18".

The boron-epoxy composite doublers were a symmetric, multi-axial lay-up of 13 plies: $[0, +45, -45, 90]_3$ with a 0^o 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 (see Fig. 1). 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). A 30:1 ply taper ratio was utilized; this results in a reduction in length of 30 times the ply thickness. Both the lay-up and taper ratio matched the preliminary doubler design (pre-analysis) produced by Lockheed. Each composite doubler had an overall thickness of 0.080" (approximately 0.0057" per ply plus an adhesive layer of 0.007"). The overall doubler dimension is shown in Figure 5. The number of plies and fiber orientations were chosen such that the crossectional stiffness ratio of boron/epoxy to aluminum was 1.4:1 $\{(Et)_{Al} = 1.4\ (Et)_{BE}\}$. For uniformity of test results, the 1.4 stiffness ratio was chosen to match the value employed in the Boeing test series. Further, the Boeing tests [17] showed that a stiffness ratio of 1.4 produced better results than similar test on specimens with ratios of less than 1:1. Reference [3] lists the properties of boron/epoxy composites. The modulus values used here were $E_1 = 26 \times 10^3$ ksi and $E_2 = E_3 = 2.5 \times 10^3$ ksi.

Doubler Installation - A Boeing specification (D658-101183-1) for composite doubler installations was generated by the Boeing/Textron/Federal Express study [17]; a summary is provided in Appendix B. The installation procedure includes a phosphoric acid anodize as a means of preparing the parent surface for the bonding process. The Air Force procedure referenced above (References [11-13] from Wright Patterson Air Force Base) is very similar to the process described in Appendix A except that the Air Force surface preparation step uses a silane chemical. Both installation procedures will be validated and certified during the course of the FAA/AANC program.

Table I: Composite Coupon Specimen Matrix And Fatigue Test Summary

Configuration	Description *	Fatigue Test Results Δ
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 (L = 3.495" when starter notch is subtracted) 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 - crack did not initiate from hole in aluminum plate
BE-6	Aluminum Control Specimen: unabated 0.5" crack in aluminum plate; plate has no reinforceing doubler	• crack propagated until specimen failed (L = 3.735") 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	• specimen not tested - currently being fabricated

* All doublers were installed using the PACS surface preparation technique

Δ Crack growth rates in all composite doubler specimens were 10 to 20 times slower than the Control Specimens (BE-6) which had no reinforcing doublers.

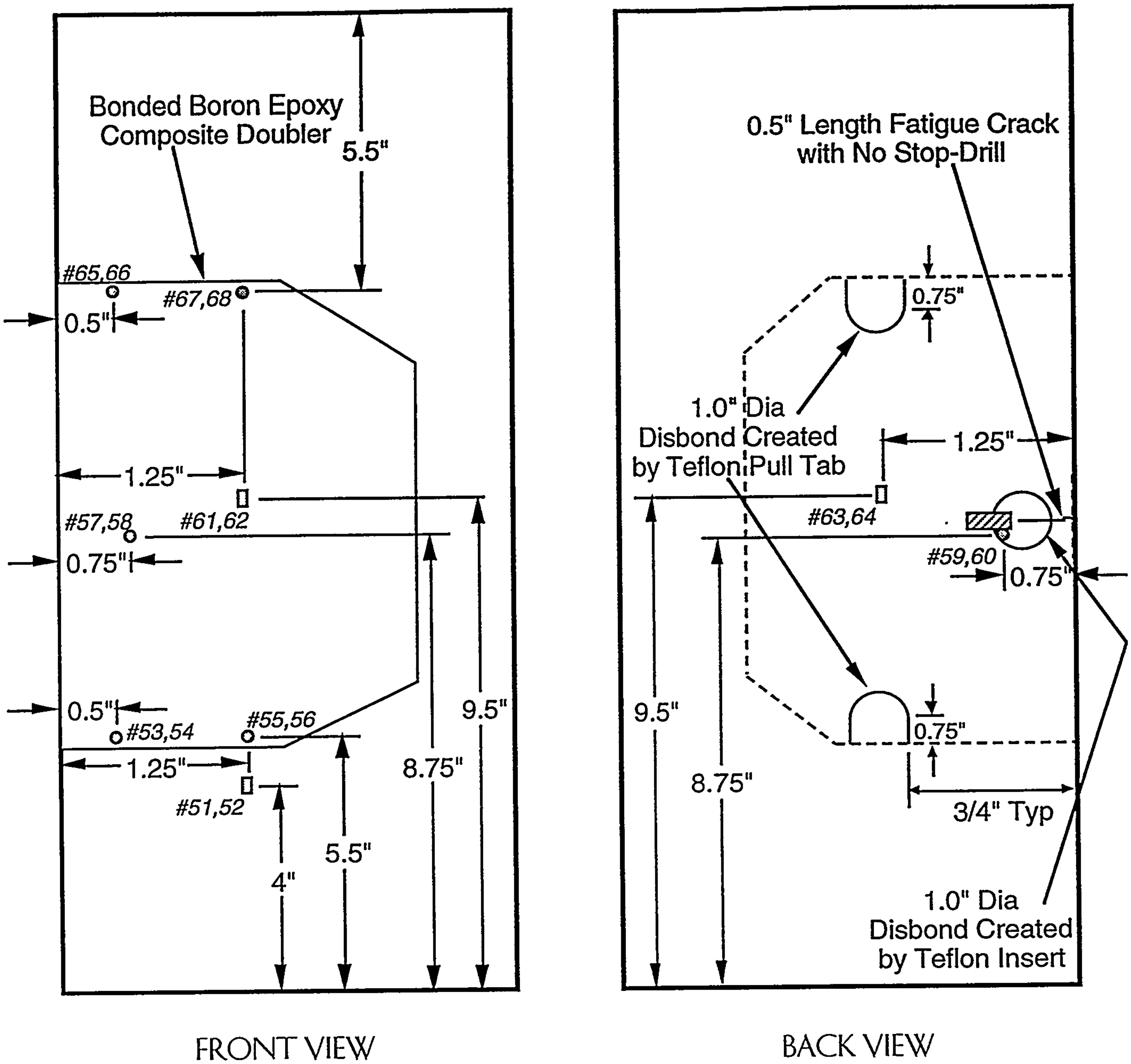
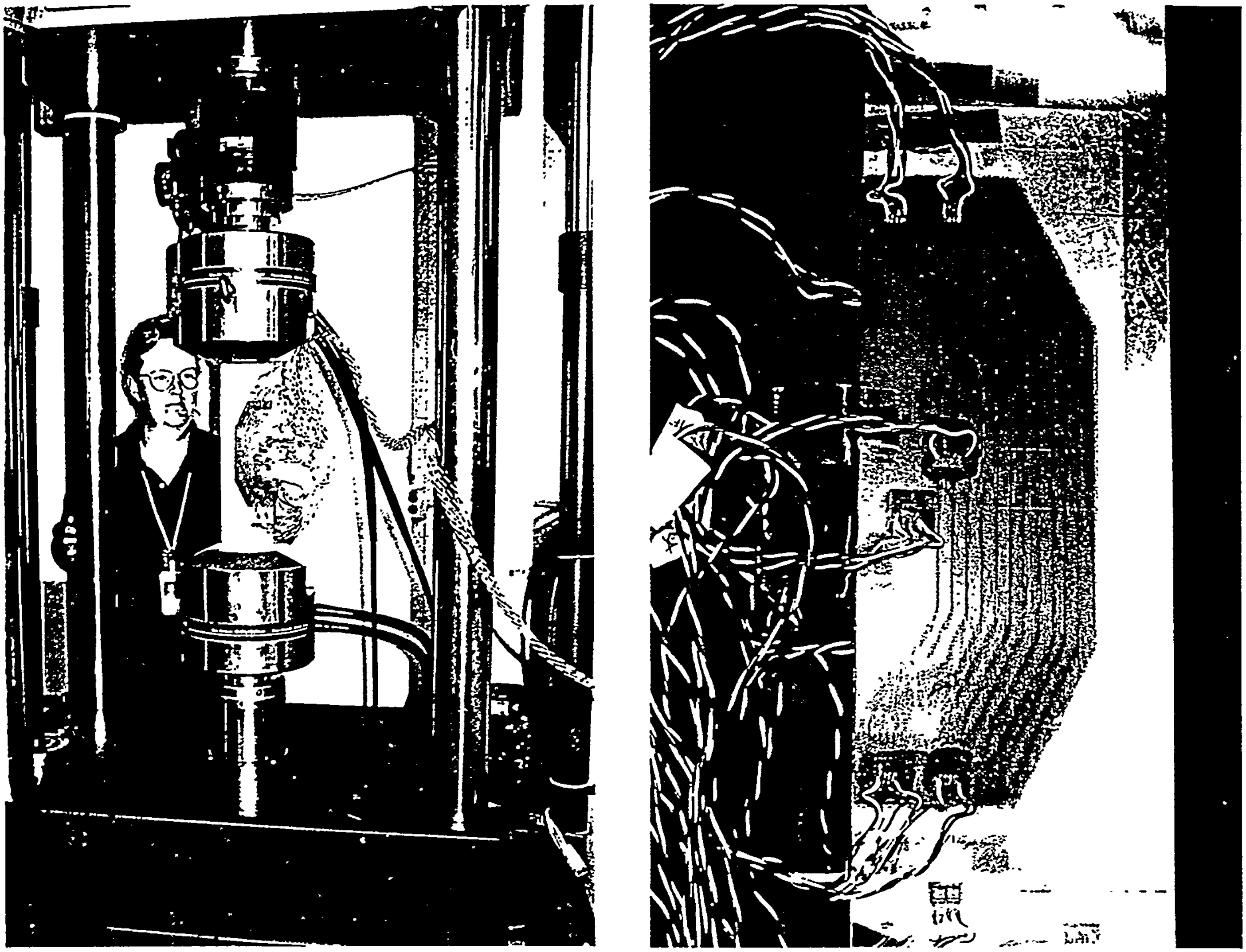


FIGURE 5 : Composite Doubler Fatigue Specimen with an Unabated Crack and Co-Located Disbond (Specimen BE-4); Numbers in Italics Represent Biaxial Strain Gage Channels

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, is intended 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 is based on the maximum hoop stresses observed in the L-1011 skin (cabin pressure plus flight loads). A computer-controlled, hydraulic, mechanical test machine was used to apply the fatigue loads. Figure 6 shows a fatigue specimen mounted in the tension test machine. Following is a summary of the fatigue and static ultimate tests.



A. Specimen Mounted in Mechanical Test Machine

B. Close-Up View Showing Doubler Installation Taper and Strain Gage Locations

FIGURE 6 : Composite Doubler Specimen Subjected to Tension-Tension Fatigue Tests

A. Fatigue Tests with Static Strain Measurements
   1) The 850 - 5600 lb. (3 KSI to 20 KSI) cyclic fatigue loads were applied at 5 Hz.
   2) Each fatigue test was stopped at 36,000, 54,000, and 72,000 fatigue cycles. 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.
   3) The specimens were inspected with multiple NDI techniques at 36,000, 54,000, 72,000 and 144,000 cycles in order to monitor crack, delamination, and disbond growth. The NDI techniques included visual (microscope aided), ultrasonics (Staveley Bondmaster device), and eddy current (Nortec 19e device).
   4) 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 5. Biaxial gages were used to measure both the axial and transverse strains in the anisotropic composite material.
   5) Static strain measurements were acquired at the following four fatigue test stopping points: 1) Fatigue Cycles = 0, 2) Fatigue Cycles = 36,000, 3) Fatigue Cycles = 72,000, and 4) Fatigue Cycles = 144,000. Strain values were acquired at a series of load levels up to the maximum L-1011 hoop stress of 20 KSI.

B. Static Tension Ultimate Tests
   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. The specimens tested were the BE-1 and BE-2 configurations. They represent the worst case conditions of unabated fatigue cracks and severe disbonds in the critical load transfer region.
   1) A 850 lb. pre-load was applied to eliminate the residual curvature in the test specimens; the strain gage bridges were balanced to produce a zero strain output signal.
   2) The load was increased, using displacement mode control, at a continuous rate of 0.05 inch/minute. Failure was defined as the point where the specimen was unable to sustain an increasing load.
   3) An extensometer, mounted across the crack, was used to obtain load vs. total displacement information. This provided a global modulus of elasticity for the composite/aluminum structure. The biaxial strain gages were continuously monitored to measure the strain fields during flaw propagation.

COMPOSITE COUPON TEST RESULTS

The fatigue and static ultimate tests are still underway at the AANC. Thus, static and fatigue test data is not presented for all of the test specimens described above. Further, it is anticipated that several additional specimens will be designed and tested in order to expand on this first round of coupon tests.

Fatigue Tests

Fatigue tests have been completed on specimens BE-1 through BE-6 with one repeat of a duplicate BE-6 specimen. The results are summarized in Table I and shown graphically in Figure 7. 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. 7 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.

2. Fatigue Cracks With No Abatement - Specimens BE-1 and BE-4 survived 144 K fatigue cycles with crack growths of 2" or less. Without any type of crack abatement (e.g. stop-drill at crack tip), crack propagation began shortly after fatigue testing was initiated. Specimens BE-1 and BE-4 produced very similar crack growth curves (see Figure 7). 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 5). 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 7 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.

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. The end result was that the crack did not propagate after 144 K fatigue cycles. The bonded composite doubler picks 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 critical 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. Although the AANC NDI tests demonstrated the detection of disbonds as small as 0.25" in diameter, 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. Crack propagation in the specimens, and the accompanying displacements as the crack opened each cycle, produced cohesive failure (cracking) in the adhesive. However, this failure was localized about the length of the crack and did not result in any disbonds (adhesive failure). Finally, comparisons between the BE-1 (no disbonds) and the BE-2, 3, and 4 (engineered disbonds) fatigue curves in Figure 7 show that the large disbonds did not decrease the composite doubler's performance.

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. [Again, note that an optimum installation or a specimen without a fatigue crack as in BE-5 would be able to sustain much higher fatigue cycles. Therefore, the life extension factor of 20, calculated using non-optimum installations, is considered conservative.]

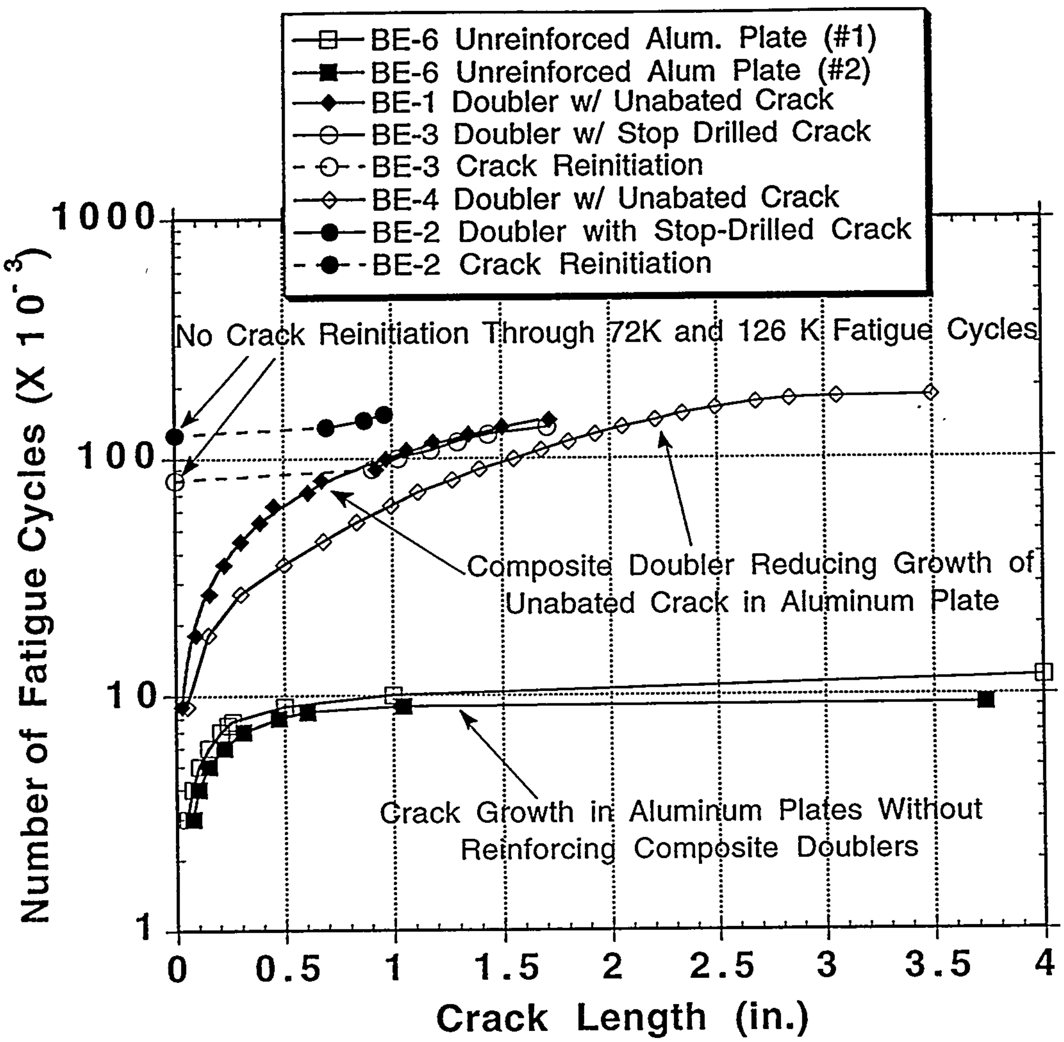


FIGURE 7 : Fatigue Crack Growth in 2024-T3 Plates With and Without Reinforcing Composite Doublers

In Figure 7, the number of fatigue cycles are plotted using a log scale because it clearly shows the crack arresting affect of the composite doublers. Overall, the crack propagation is reduced by a factor of 10 (with an extrapolation to 20 for longer cracks extending beyond 2" in length) through the use of boron/epoxy doublers. 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. Figure 7 also shows that the crack growth rates for all of the specimens can be approximated by a bilinear fit to the data plotted on a semi-log scale. This simply demonstrates the well known power law relationship between fatigue cycles (N) and crack length (a). The first linear portion extends to a = 0.30" in length. The slopes, or crack growth rates, vary depending on the localized configuration of the flaw (e.g.

stop-drilled, co-located disbond, presence of doubler). The second linear portion extends to the point of specimen failure. A comparison of these linear approximations shows that the crack growth rate is reduced 20 to 40 times through the addition of a composite doubler. Also, as noted above, all of the doubler specimen crack growth rates are the same after the crack propagates past the initial flaw configuration at the specimen's edge (e.g. engineered disbonds). This occurs at approximately a = 1.5".

6. Comments on Fatigue Loading Spectrum - The fatigue tests were conducted using a 3 KSI to 20 KSI sinusoidal load spectrum. The 3 KSI pre-load was intended to eliminate the residual curvature in the test specimens caused by the different coefficients of thermal expansion between the aluminum and boron-epoxy material. However, the pre-load was not able to completely eliminate all of the specimen curvature. As a result, there were bending loads introduced into the tension fatigue tests. The accompanying stress reversals produced a slight amount of "oilcanning" which is not commonly found in aircraft structures. Thus, the fatigue load spectrum exceeded the normal fuselage pressure stresses and the performance values sited here should be conservative.

Strain Field Measurements

Figure 5 shows the typical strain gage layout which was used to monitor: 1) the load transfer into the composite doublers and, 2) the strain field throughout the composite plate. A summary of the strain field in the test coupons can be seen in the series of curves shown in Figure 8. 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 8). 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 in Figure 8 registered very little strain since they were mounted over disbonds (see Fig. 5) which produced strain relief in the doubler.

Although the strains remained constant in the critical load transfer region, Figure 9 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 9 show how the doubler picks up more load as the crack propagates and the plate relieves its load. The same information can be obtained by looking at how the plate strains are reduced as the crack propagates.

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.

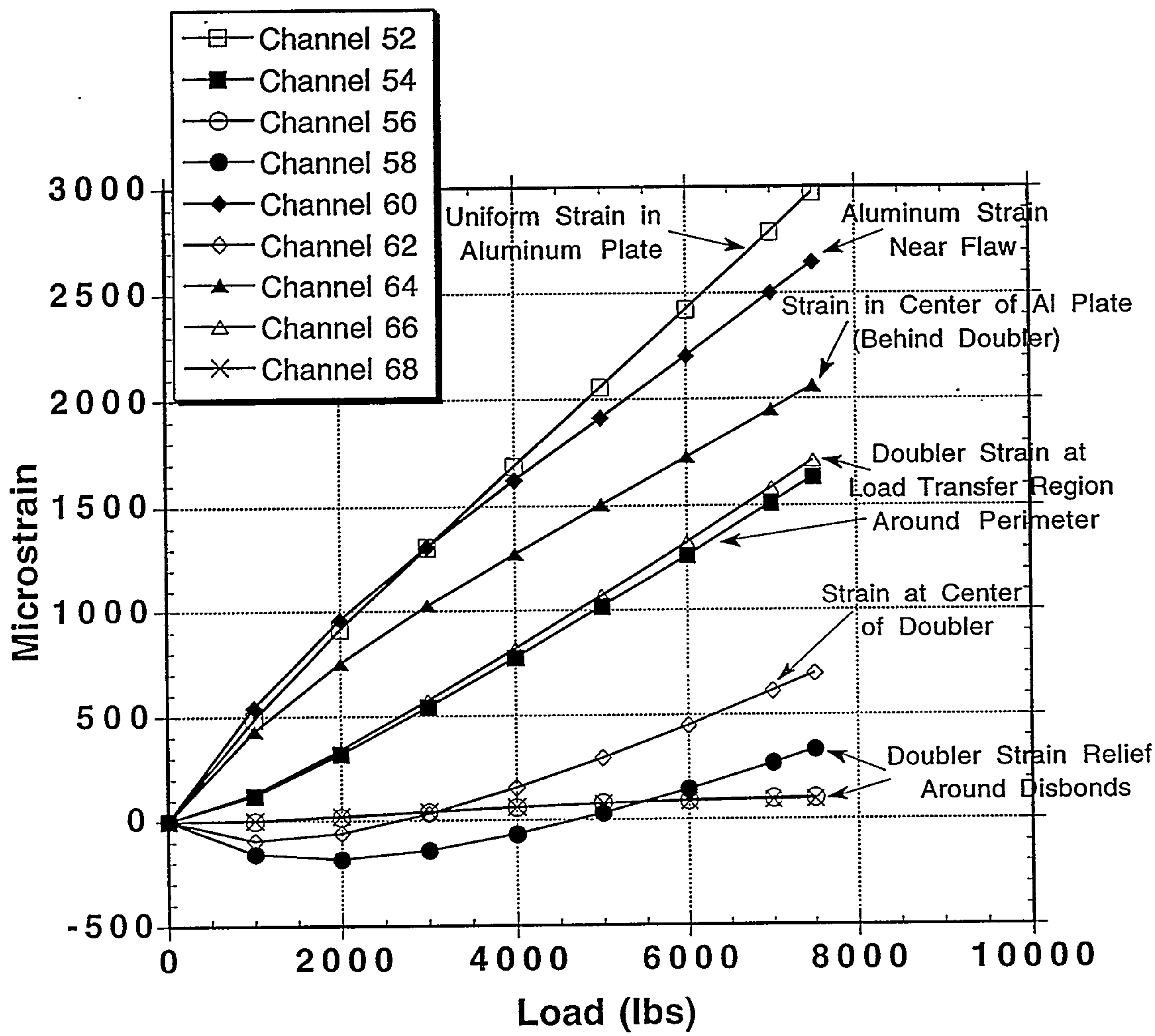


FIGURE 8 : Aluminum and Composite Strain Fields - Load Transfer into Composite Doubler (Specimen BE-4); Strain Gage Locations are Shown in Figure 5.

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. The adhesive fracture propagated up to the point where the composite laminate tapered to only a 3 ply thickness. At this point, the composite laminate fractured vertically producing enough deformation in the specimen to release the load. Figure 10 shows the strain field in specimen BE-2 up through failure. The aluminum plate begins to yield at approximately 12,000 lbs while the doubler continues to increase its load in a linear fashion until failure occurs at 16, 600 lbs.

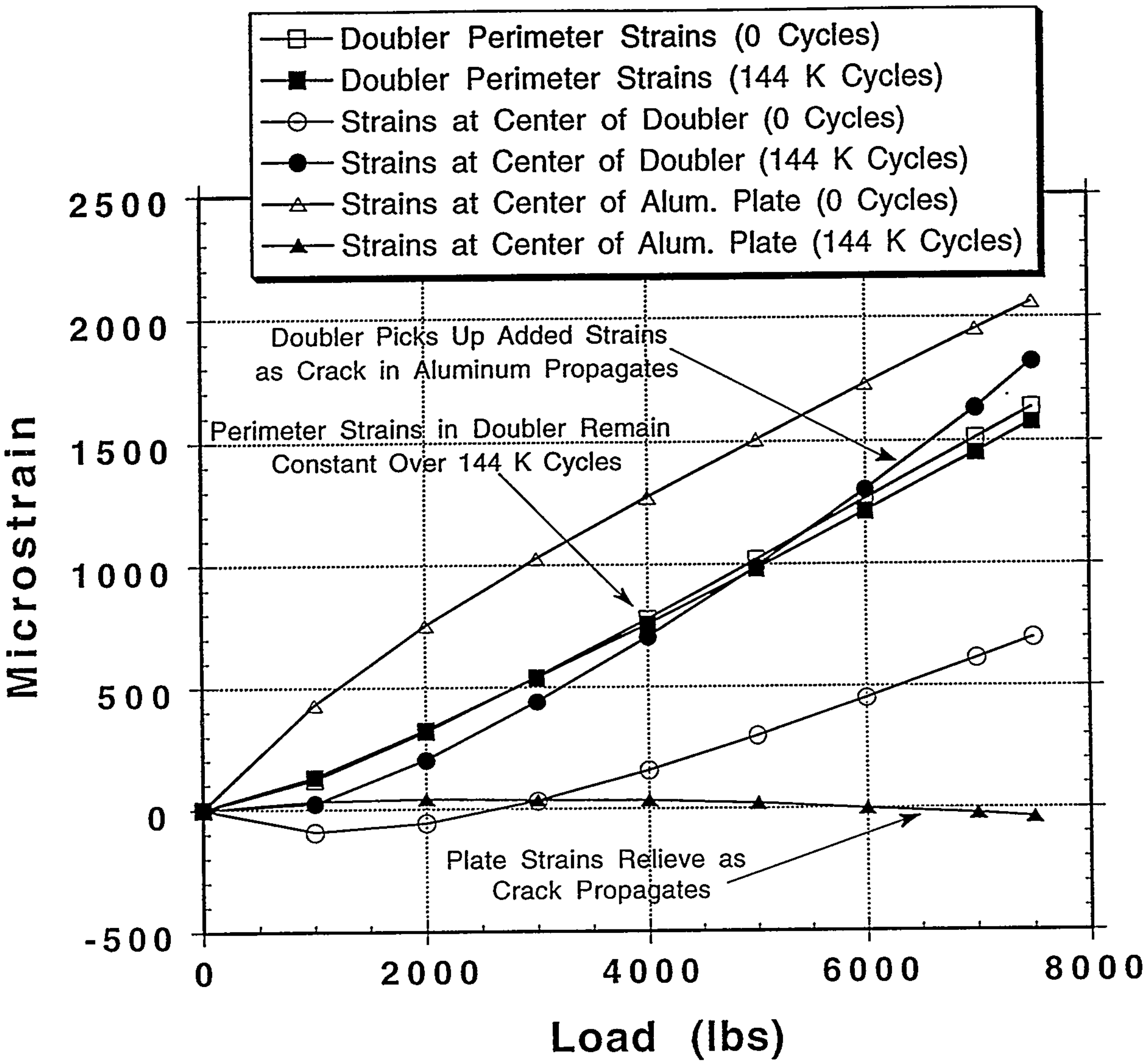


FIGURE 9 : Strain Field Changes as a Result of Crack Propagation (Specimen BE-4)

In calculating the ultimate tensile stress, the cross-sectional dimensions of the aluminum and the bonded doubler were used. If ultimate stresses were determined using the aluminum plate dimensions alone as in Ref. [17], then the values listed here would be even greater.

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.

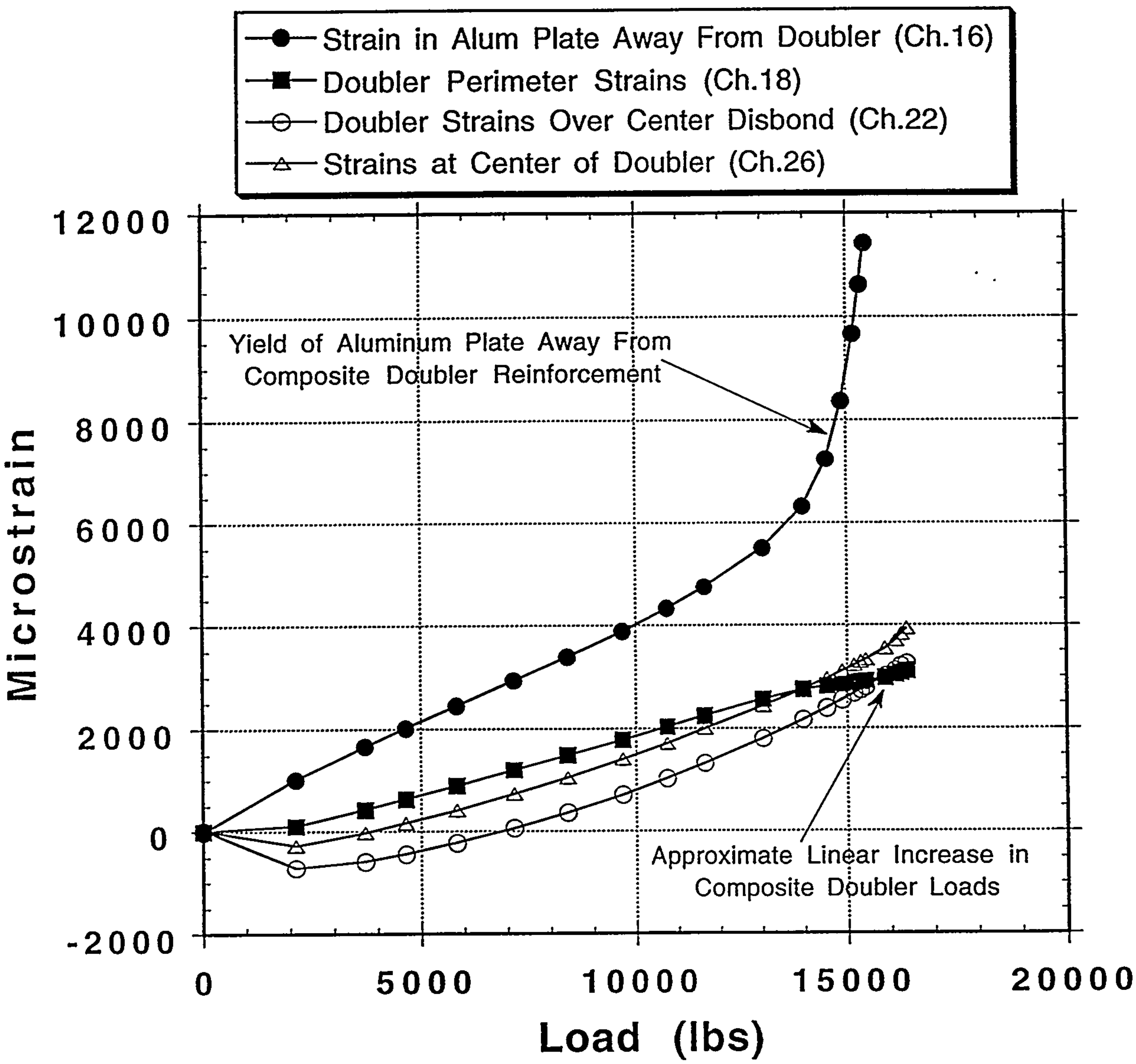


FIGURE 10 : Strain Fields in Composite Doubler and Aluminum Plate During Residual Strength Test (Specimen BE-2)

CONCLUSIONS

In spite of significant successes in military applications, 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. By focusing on a specific commercial aircraft application - reinforcement of the L-1011 door frame - and encompassing all "cradle-to-grave" tasks (i.e. design, analysis, installation, structural integrity and inspection), this program is designed to firmly establish the capabilities of composite doublers. The parties participating in this project are the Aging Aircraft NDI Validation Center at Sandia National Labs, Lockheed Aeronautical Systems, Delta Air Lines, Warner Robins Air Force Base, Wright Patterson Air Force Base, Textron Specialty Materials, Inc., and the FAA.

The fatigue and strength tests recently completed by the AANC were designed to supplement the current database on composite doubler performance by focusing on commercial aircraft structural configurations. The AANC tests quantitatively demonstrated the damage tolerance of boron/epoxy composite doublers in the presence of compounding flaw scenarios.

Fatigue Tests - The composite doublers produced significant crack growth mitigation when subjected to simulated pressure tension stress cycles. Even specimens with unabated fatigue cracks and co-located disbonds were able to survive four fatigue lifetimes (144 K cycles) without specimen failure (less than 2" crack growth). During the course of fatigue cycling, all crack growth occurred in the aluminum plates. No fractures were found in any of the composite laminates. Comparisons with control specimens which did not have composite doubler reinforcement showed that the fatigue lifetime was extended by a factor of 20.

It should be noted that the relative importance of the fatigue data will be determined by the ongoing damage tolerance analysis being carried out by Lockheed. This analysis will establish the allowable flaw growth and inspection intervals for the L-1011 door frame application. It should also be noted that a more desirable basis of comparison for the performance characteristics discussed above will be provided by specimen BE-7 (normal installation with no flaws) which has not yet been tested.

Adhesive Disbonds - The fatigue specimens contained engineered disbonds 3 to 4 times the size detectable by current inspection techniques. 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. 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. This value remained constant over four fatigue lifetimes indicating that there was no deterioration in the bond strength. The stresses in the doubler increased to pick up the loads released by the plate during crack propagation.

Residual Strength - The static load-to-failure tests were loosely termed static ultimate tests. However, since the specimens were tested after cracks were grown, these tests were actually residual strength tests. Comparisons of the test results with tabulated values for 2024-T3 ultimate tensile strength, which do not use flawed specimens, should be conservative. Even the existence of disbonds and fatigue crack 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.

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. 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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APPENDIX A: Overview of Results from Boeing/Textron Fatigue and Static Ultimate Tests (see Ref. [17] for detailed discussion of results)

1. Material: 7075-T6 aluminum, 0.063" thick
2. Major Parameters Studied -
   a. Geometric Design Variables - ply orientation (unidirectional vs. 0° and + 45°); doubler geometry and ply taper; number of plies (doubler thickness); abated (stop-drilled) vs. unabated cracks.
   b. Environmental Exposure - effects of cure pressure; effects of impact on boron/epoxy surface; effects of humidity, immersion in Skydrol fluid, and low temperature performance.
   c. Flaw Growth - saw cut cracks placed in parent aluminum; disbonds engineered between doubler and aluminum plate.

3. Fatigue Test Results - (3 to 20 KSI stress cycles; 330,000 fatigue cycles were considered runout since this corresponds to 4 times the design objective of the 737 aircraft)
   a. Performance of Baseline Doubler - 10 specimens went to 300K cycles with no crack growth; 1 specimen survived 300K cycles, however, crack growth reinitiated at 113K cycles (defect found in stop-drilled hole).
   b. Test Performed Without a Boron Doubler - stop-drilled cracks propagated to failure in all 3 tests with an average failure Of 3.1K cycles.

c. Ply Taper and Lay-Up - no effect of changing ply drop-off ratio from 25:1 to 12:1; no effect of unidirectional (0°) vs. 0° and ± 45°.
d. Number of Plies - stiffness ratio was reduced from 1.4:1 Baseline configuration to 0.9:1 (i.e. 4 vs. 6 plies); 1 of the 4 ply specimens went to 300K cycles with no crack growth; 2 of the 4 ply specimens failed at 254K and 265K, respectively.
e. No Stop Drill - crack propagation occurred in all 3 specimens tested, however, the boron doublers held the specimens intact up to 300K cycles.
f. Disbonds - disbonds were placed over the stop-drilled crack and at the doubler edge; at room temperature, 2 specimens went to 300K cycles with no crack growth and 1 specimen went to 300K cycles with crack re-initiation at 149K cycles; during simultaneous temperature cycling (-54°C to 72°C), 1 specimen went to 300K cycles with no crack growth and 2 specimens went to 300K cycles with crack re-initiation at 147K and 211K cycles, respectively.
g. Environmental Exposure - no effect from exposure to 85% relative humidity or Skydrol hydraulic fluid; at low temperature (-54°C), 6 specimens went to 300K cycles with no crack growth and 4 specimens went to 300K cycles with crack re-initiation at 44K, 47K, 84K and 188K cycles, respectively.
h. Variation in Doubler Cure Pressure - (0.2 atm vs. 0.9 atm); all 6 specimens went to 300K cycles with no crack growth.
i. Impact - 20 specimens went to 300K cycles with no crack growth following impact which produced visually evident surface damage.

4. Static Tensile Ultimate Tests - (19 tests on non-fatigued specimens; 96 tests on fatigue specimens which experienced 300K cycles)
   a. 2 of the 117 specimens failed below the 78 KSI A-basis statistical minimum for 7075-T6 - 1 at 77 KSI and 1 at 69 KSI .
   b. The average strength for the tests was 83 KSI.
   c. The boron doubler was able to restore the load carrying capability of the aluminum plate even when an unabated crack existed under the doubler.

APPENDIX B: Composite Doubler Installation Process Summary

(see Ref. [3, 13, 17, 18] for additional details)

1. Aluminum Surface Preparation - solvent cleaned per Boeing Aircraft Specification (BAC) 5750; alkaline cleaned per BAC 5749; oxide removal per BAC 5514; phosphoric acid anodize as per BAC 5555; anodize is implemented in the field using a Phosphoric Acid Containment System (PACS).
2. Adhesive Process - aluminum surface is primed per Boeing Materials Specification (BMS) 5-89; co-cure the BMS 5-101 Type II structural adhesive simultaneously with the boron/epoxy doubler.
3. Doubler Installation - boron/epoxy doubler is laid up in accordance with the application design requirements; cure for 90 to 120 minutes at 225°F to 250°F at 0.5 atm vacuum bag pressure; computer-controlled heater blankets are used to provide the proper temperature cure profile in the field.

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