MAR 16 1999

SANDIA REPORT

SAND98-1015

Unlimited Release

Printed February 1999

RECEIVED
MAR 19 1999
OST

Full-Scale Structural and NDI Validation Tests of Bonded Composite Doublers for Commercial Aircraft Applications

Dennis Roach and Phil Walkington

Prepared by
Sandia National Laboratories
Albuquerque, New Mexico 87185 and Livermore, California 94550

Sandia is a multiprogram laboratory operated by Sandia Corporation,
a Lockheed Martin Company, for the United States Department of
Energy under Contract DE-AC04-94AL85000.

Approved for public release; further dissemination unlimited.

Sandia National Laboratories

Issued by Sandia National Laboratories, operated for the United States Department of Energy by Sandia Corporation.

NOTICE: This report was prepared as an account of work sponsored by an agency of the United States Government. Neither the United States Government, nor any agency thereof, nor any of their employees, nor any of their contractors, subcontractors, or their employees, make any warranty, express or implied, or assume any legal liability or responsibility for the accuracy, completeness, or usefulness of any information, apparatus, product, or process disclosed, or represent that its use would not infringe privately owned rights. Reference herein to any specific commercial product, process, or service by trade name, trademark, manufacturer, or otherwise, does not necessarily constitute or imply its endorsement, recommendation, or favoring by the United States Government, any agency thereof, or any of their contractors or subcontractors. The views and opinions expressed herein do not necessarily state or reflect those of the United States Government, any agency thereof, or any of their contractors.

Printed in the United States of America. This report has been reproduced directly from the best available copy.

Available to DOE and DOE contractors from
Office of Scientific and Technical Information
P.O. Box 62
Oak Ridge, TN 37831

Prices available from (703) 605-6000
Web site: <http://www.ntis.gov/ordering.htm>

Available to the public from
National Technical Information Service
U.S. Department of Commerce
5285 Port Royal Rd
Springfield, VA 22161

NTIS price codes
Printed copy: A03
Microfiche copy: A01

DISCLAIMER

Portions of this document may be illegible in electronic image products. Images are produced from the best available original document.

SAND98-1015
Unlimited Release
Printed February 1999

Full-Scale Structural and NDI Validation Tests of Bonded Composite Doublers for Commercial Aircraft Applications *

Dennis Roach
Phil Walkington
Airworthiness Assurance Department

Sandia National Laboratories
P. O. Box 5800
Albuquerque, NM 87185-0615

Abstract

Composite doublers, or repair patches, provide an innovative repair technique which can enhance the way aircraft are maintained. Instead of riveting multiple steel or aluminum plates to facilitate an aircraft repair, it is possible to bond a single Boron-Epoxy composite doubler to the damaged structure. 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. This report focuses on a series of full-scale structural and nondestructive inspection (NDI) tests that were conducted to investigate the performance of Boron-Epoxy composite doublers. Full-scale tests were conducted on fuselage panels cut from retired aircraft. These full-scale tests studied stress reductions, crack mitigation, and load transfer capabilities of composite doublers using simulated flight conditions of cabin pressure and axial stress. Also, structures which modeled key aspects of aircraft structure repairs were subjected to extreme tension, shear and bending loads to examine the composite laminate's resistance to disbond and delamination flaws. Several of the structures were loaded to failure in order to determine doubler design margins. Nondestructive inspections were conducted throughout the test series in order to validate appropriate techniques on actual aircraft structure. The test results showed that a properly designed and installed composite doubler is able to enhance fatigue life, transfer load away from damaged structure, and avoid the introduction of new stress risers (i.e. eliminate global reduction in the fatigue life of the structure). Comparisons with test data obtained prior to the doubler installation revealed that stresses in the parent material can be reduced 30% - 60% through the use of the composite doubler. Tests to failure demonstrated that the bondline is able to transfer plastic strains into the doubler and that the parent aluminum skin must experience significant yield strains before any damage to the doubler will occur.

* This work was performed for the Federal Aviation Administration (FAA) Technical Center under US Department of Transportation Contract DTFA 03-95-X-90002. This document is currently under review by the FAA for parallel publication by the Department of Transportation.

Acknowledgments

Numerous people have contributed to the success of validating bonded composite doublers for commercial aircraft applications. In addition to the FAA Airworthiness Assurance Center (Sandia Labs), team members for the L-1011 door corner repair included Lockheed-Martin Aeronautical Systems (doubler design and analysis - Bob Bell, Surendra Shah, and Kevin Jones), Delta Air Lines (Engineering Repair Authorization document, doubler installation and NDI support - John Marshall, Malcolm Westberry, Chuck Minafo, and Raymond Worley), Textron Systems Div. (specimen fabrication and installation - Tom Shahood and Jeff Brown), the FAA Atlanta Aircraft Certification Office (review and approval - Charles Perry and Paul Sconyers), and the FAA William J. Hughes Technical Center (project oversight - Pete Versage, Chris Smith, and Dave Galella; FAA Chief Scientist NDI - Al Broz). Specific contributions to the full-scale testing information presented here came from the following individuals. Tom Rice (Sandia Labs) helped design the fuselage biaxial test facility, Darin Graf (Sandia Labs) assisted in instrumentation, specimen inspections, and structural testing, and Marlene Uribe (Sandia Labs) helped with the instrumentation and test set-up and provided overall consultation regarding structural testing. These contributions are gratefully acknowledged. This work was funded by the FAA William J. Hughes Technical Center under a U.S. Department of Transportation contract.

FOREWORD

As part of the Federal Aviation Administration's (FAA) National Aging Aircraft Research Program (NAARP), the FAA William J. Hughes Technical Center, established a major center at Sandia National Laboratories (SNL): the Airworthiness Assurance NDI Validation Center (AANC). The AANC conducts numerous projects related to the validation of improved aircraft maintenance practices. The Center also supports technology development initiatives. To facilitate these activities, the AANC has set up a hangar facility at the Albuquerque International Airport.

One of the primary goals of NAARP is to foster new technology associated with the repair of civil aircraft. 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 flaws develop throughout the aircraft's skin and substructure elements. These flaws can take the form of cracks, corrosion, disbonds, dents, and gouges. Composite doublers, or repair patches, provide an innovative repair technique which can enhance the way aircraft are maintained. Instead of riveting multiple steel or aluminum plates to facilitate an aircraft repair, it is possible to bond a single Boron-Epoxy composite doubler to the damaged structure.

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 use of bonded composite doublers offers the airframe manufacturers and airline maintenance facilities a cost effective technique to safely extend the lives of their aircraft. However, before this advanced aircraft repair technique could be accepted for commercial aircraft use, uncertainties surrounding the application, subsequent inspection and long-term endurance of composite doublers had to be addressed.

This document is one in a series of reports covering the AANC's comprehensive evaluation of composite doublers for commercial aircraft use. The development and validation effort addressed the full array of engineering issues including design, material allowables, installation, damage tolerance, quality assurance, in-service surveillance (nondestructive inspection), and FAA/industry requirements. For completeness, beneficial overlaps of information will be presented in each report. The full suite of reports, each containing a similar foreword section, are:

- 1. Development and Validation of Nondestructive Inspection Techniques for Composite Doubler Repairs on Commercial Aircraft (SAND98-1014)*
- 2. Damage Tolerance Assessment of Bonded Composite Doubler Repairs for Commercial Aircraft (SAND98-1016)*
- 3. Full-Scale Structural and NDI Validation Tests on Bonded Composite Doublers for Commercial Aircraft Applications (SAND98-1015)*

Report #1: "Development and Validation of Nondestructive Inspection Techniques for Composite Doubler Repairs on Commercial Aircraft" - The purpose of this report is to document the NDI techniques and procedures which have been assessed to inspect bonded composite doubler installations on aircraft structures. The intent of the inspections are to detect: 1) disbonds, delaminations, and porosity in the composite laminate, and 2) cracks in

the parent aluminum material. An array of conventional and advanced NDI techniques were evaluated. Flaw detection sensitivity studies were conducted on applicable eddy current, ultrasonic, X-ray and thermography based devices. The pulse-echo ultrasonic technique deployed in this program uses the traditional A-scan approach as its basis, however, significant improvements are realized through the adoption of C-scan imaging. An X-ray inspection was modified from its original specification in the L-1011 NDT Manual. A series of tests were performed in order to: 1) verify that composite doublers do not impede X-ray inspections, and 2) study X-ray optimization when inspecting through composite doublers. This study concluded that a team of NDI techniques can identify flaws in composite doubler installations well before they reach critical size. The development of appropriate inspection reference standards, critical to performing proper inspections, is also discussed.

Report #2: "Damage Tolerance Assessment of Bonded Composite Doubler Repairs for Commercial Aircraft" - This report focuses on a series of fatigue and strength tests which were conducted to study the damage tolerance and fatigue life enhancement associated with Boron-Epoxy composite doublers. Tension-tension fatigue and ultimate strength tests attempted to grow engineered flaws in coupons with 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. The structural tests were used to: 1) assess the potential for interply delaminations and disbonds between the aluminum and the laminate, and 2) determine the load transfer and crack mitigation capabilities of composite doublers in the presence of severe defects. A series of specimens were subjected to ultimate tension tests in order to determine strength values and failure modes. Coupon test configurations, the loads applied, the test procedures, and all associated results are documented in this report.

Report #3: "Full-Scale Structural and NDI Validation Tests on Bonded Composite Doublers for Commercial Aircraft Applications" - This report describes a series of structural and nondestructive inspection (NDI) tests which were conducted to investigate the performance of Boron-Epoxy composite doublers. Full-scale tests were conducted on fuselage panels cut from retired aircraft. These full-scale tests studied stress reductions, crack mitigation, and load transfer capabilities of composite doublers using simulated flight conditions of cabin pressure and axial stress. Also, structures which modeled key aspects of aircraft structure repairs were subjected to extreme tension, shear and bending loads to examine the composite laminate's resistance to disbonds and delaminations especially in high peel stress regions. Nondestructive inspections were conducted throughout the test series in order to validate pertinent techniques on actual aircraft structure.

The test results were also used to verify design and analyses methodologies for composite doubler technology. The primary test article was a large fuselage section cut from a retired All Nippon Airways (ANA) L-1011 aircraft. The fuselage test article included a passenger door cut-out and contained all substructure frame, longeron, and stringer elements. Several other test configurations - consisting of composite doublers mounted on simulated aircraft panels - were examined in order to assess the response of composite doublers in worst-case shear and bending load scenarios. These two test configurations were loaded to failure in order to determine safety factors associated with current doubler design methodologies.

Background and Deliverables - The Federal Aviation Administration sponsored this project at the AANC to determine the viability of bonded composite doublers and to gain FAA approval for their use 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. In addition to the AANC, other project team members included Lockheed-Martin, Delta Air Lines, and Textron Specialty Materials. Appropriate FAA oversight was provided through the Atlanta Aircraft Certification Office (ACO) and the FAA's William J. Hughes Technical Center. The project deliverables will assist the FAA in developing guidance which assures the continued airworthiness of composite doublers.

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, quality assurance, 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 ACO maintains the documents under the FAA project number SP1798AT-Q. The documentation package includes:

<u>Report</u>	<u>Report Number</u>
1. Boron-Epoxy Material Allowables	LG95ER0193
2. Damage Tolerance Assessment	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 L-1011 Composite Doubler	LG95ER0157
6. L-1011 Composite Doubler Drawing (Upper Fwd. Corner, P-3 Passenger Door)	LCC-7622-378
7. Nondestructive Inspection Procedures	AANC-PEUT-Comp-5521/4-004

The first use of the above documentation package was to support the installation of an FAA-approved Boron-Epoxy composite repair on a Lockheed L-1011 aircraft. The repair has been installed on the upper forward corner of a P3 passenger door frame. The aircraft is currently operating in the Delta Air Lines fleet. Four post-installation inspections, spanning one year of aircraft operation, have shown the doubler to be free of flaws. A second important product of the results cited above is the Lockheed-Martin Service Bulletin 093-53-278 which allows the door corner composite doubler to be installed on all L-1011 aircraft. With the successful completion of the L-1011 door corner application, the FAA and AANC are now conducting a program with Boeing and Federal Express to develop, certify, and install a more generic set of composite doubler applications for a variety of common aircraft repairs.

This Page Left Intentionally Blank

Full-Scale Structural and NDI Validation Tests of Bonded Composite Doublers for Commercial Aircraft Applications

Table of Contents

<u>Section</u>	<u>Title</u>	<u>Page</u>
1.0	INTRODUCTION AND TEST SERIES OVERVIEW	1
1.1	Doubler Design Guidelines	6
1.2	Damage Tolerance and Fracture Control Plan.....	9
1.2.1	Damage Tolerance and Analysis Methodologies.....	9
1.2.2	Damage Tolerance Establishes Fracture Control Plan.....	13
1.2.3	Damage Tolerance Testing.....	16
1.3	Full-Scale Testing Philosophy.....	16
1.4	Conformity Inspection and FAA Oversight.....	17
PART I:	CABIN PRESSURIZATION TEST OF	
	FUSELAGE DOOR SURROUND STRUCTURE.....	19
2.0	FUSELAGE DOOR SURROUND STRUCTURE TESTS.....	21
2.1	Participants in Test Article Modifications and Testing.....	21
2.1.1	Conformity Inspection.....	22
2.2	Description of Test Article.....	22
2.2.1	L-1011 Fuselage Structure.....	22
2.2.2	Door Surround Structure Before Composite Doubler Installation.....	22
2.2.3	Reinforcement of Test Article for Load Application Points.....	26
2.2.4	Repair of Fuselage Test Article with a Composite Doubler.....	30
2.3	Structural Test Loads and Procedures.....	32
2.3.1	Test Facility Description.....	32
2.3.2	Instrumentation.....	36
	Door Surround Structure Test Before Composite Doubler Installation.....	37
	Door Surround Structure Test After Composite Doubler Installation.....	37
2.4	Pressure-Tension Test Procedures.....	37
2.5	Nondestructive Inspections.....	38
3.0	FUSELAGE DOOR SURROUND STRUCTURE TEST RESULTS.....	41
3.1	Structural Tests Before Composite Doubler Installation.....	41
3.1.1	Strain Field Analysis.....	44
3.1.2	Stress Field Analysis.....	50
3.1.3	Bending Effects.....	54
3.2	Structural Tests After Composite Doubler Installation.....	55
3.2.1	Strain Field Analysis.....	61

<u>Section</u>	<u>Title</u>	<u>Page</u>
3.2.2	Stress Field Analysis.....	73
3.2.3	Bending Effects.....	79
3.2.4	Load Transfer.....	79
3.3	Comparison of Results from “Before Doubler” and “After Doubler” Tests.....	83
3.4	Comparisons Between Experimental and Analytical Results.....	90
3.4.1	Finite Element Model of Doubler Installation.....	90
3.4.2	Before Doubler Installation.....	91
3.4.3	After Doubler Installation.....	94
3.4.4	Overview of FEM in Light of Full-Scale Test Results.....	98
3.5	Nondestructive Inspection Results.....	99
3.5.1	Inspection Intervals for L-1011 Aircraft.....	99
3.5.2	Ultrasonic Inspections.....	100
3.5.3	X-Ray Inspections.....	101
PART II:	SUBSIZE DOOR CORNER TEST (“C” SECTION TEST)	105
4.0	SUBSIZE DOOR CORNER TEST.....	107
4.1	Objective of Test.....	107
4.2	Subsize Door Corner Test Specimen.....	107
5.0	STRUCTURAL TEST LOADS AND PROCEDURES.....	111
5.1	Instrumentation.....	113
5.2	Nondestructive Inspection.....	114
6.0	SUBSIZE DOOR CORNER TEST RESULTS.....	117
6.1	Limit Load Test.....	117
6.1.1	Strains in the Aluminum Substrate.....	117
6.1.2	Strains in the Composite Doubler.....	118
6.1.3	Rosette Strain Elements in the Aluminum and Composite.....	118
6.2	150% Limit Load Test.....	119
6.2.1	Strains in the Aluminum Substrate and Composite Doubler.....	120
6.3	Ultimate Failure Test.....	121
6.3.1	Specimen Ultimate Failure.....	121
6.3.2	Strains in the Aluminum Substrate.....	122
6.3.3	Strains in the Composite Doubler.....	126
6.3.4	Rosette Strain Elements in the Aluminum and Composite.....	126
6.4.5	Maximum Principal Stresses and Directions in Aluminum and Composite.....	129
6.4	Comparisons with Finite Element Model.....	131
6.5	Post-Failure Nondestructive Inspection.....	136

<u>Section</u>	<u>Title</u>	<u>Page</u>
PART III: FOUR POINT BENDING TESTS.....		139
7.0	FOUR-POINT BENDING TEST.....	141
7.1	Objective of Test.....	141
7.2	Four Point Bending Test Specimen.....	141
8.0	STRUCTURAL TEST LOADS AND PROCEDURES.....	142
8.1	Instrumentation.....	143
9.0	FOUR-POINT BENDING TEST RESULTS.....	144
9.1	Strains and Failure in the 4B-1 Test Specimen.....	144
9.2	Strains and Failure in the 4B-2 Test Specimen.....	146
9.3	Strains and Failure in the 4B-3 Test Specimen.....	149
PART IV: COMPOSITE DOUBLER INSTALLATION ON A COMMERCIAL AIRCRAFT.....		151
10.0	L-1011 COMPOSITE DOUBLER INSTALLATION SUMMARY.....	153
10.1	Composite Doubler Repair of Delta L-1011 Aircraft.....	153
10.2	Nondestructive Inspection of Door Surround Structure and Composite Doubler.....	161
10.3	Quality Assurance Measures.....	163
10.4	FAA and Industry Approvals.....	165
11.0	BENEFITS OF COMPOSITE DOUBLERS ON COMMERCIAL AIRCRAFT....	167
11.1	Engineering Advantages.....	167
11.2	Economic Advantages.....	168
PART V: CONCLUSIONS.....		173
12.0	CONCLUSIONS FROM FULL-SCALE TEST VALIDATION.....	175
	References.....	180
Appendix A	Engineering Drawings of Biaxial Test Facility for Aircraft Fuselage Section and Instrumentation Scheme.....	185
Appendix B	Tabulated Data From Full Scale Structural Tests.....	197

Full-Scale Structural and NDI Validation Tests of Bonded Composite Doublers for Commercial Aircraft Applications

List of Figures and Drawings

<u>Figure</u>		<u>Page</u>
1	Schematic of Bonded Composite Doubler Installation on an Aluminum Skin.....	5
2	Sample Bonded Composite Doubler Installations Showing Two Families of Potential Aircraft Repair Applications.....	6
3	Residual Strength Curve.....	13
4	Crack Growth Curve Showing Time Available for Fracture Control.....	15
5	Probability of Flaw Detection vs. Flaw Size.....	15
6	L-1011 Basic Structure.....	23
7	Geometry of Door Surround Installation.....	24
8	L-1011 Door Surround Structure Test Bed.....	25
9	Structural Detail of Area to Be Repaired by Composite Doubler.....	26
10	Load Application Points on the Fuselage Test Panel.....	27
11	Reinforcement of Tie Point Locations in the Axial Direction.....	28
12	Reinforcement of Tie Point Locations in the Hoop Direction.....	28
13	Stringer/Skin Reinforcement Assembly in the Longitudinal Direction.....	29
14	Reinforcement Plates Installed in the Hoop Direction with the Turnbuckle Restraints Attached.....	30
15	L-1011 Door Corner Boron-Epoxy Repair Doubler.....	31
16	Biaxial Test Facility Design Drawing Showing the Door Surround Structure Mounted in the Vacuum Chamber and the Hoop and Axial Load Application Hardware.	33
17	Door Surround Structure Mounted in the Test Frame.....	34
18	Test Set-Up Showing Assembly of Strongbacks for Load Restraints and the Data Acquisition System.....	35
19	Vacuum Line, Pressure Transducer and Relief Valve Mounted in the Fuselage Window Port.....	35
20	Strain Gage Layout for Fuselage Door Structure Tests Without Doubler.....	42
21	Total Hoop and Axial Loads Applied to the Test Article Before the Doubler was Installed.....	43
22	Axial Load Scatter in Individual Load Cells (North and South); <i>Before Doubler Test</i>	43
23	Hoop Load Scatter in Individual Load Cells in Critical, Center Load Area (East and West); <i>Before Doubler Test</i>	44
24	Hoop Strains Along Upper (Horizontal) Portion of Door Corner.....	45
25	Hoop Strains Along Vertical Portion of Door Corner.....	46

<u>Figure</u>	<u>Page</u>
26 Axial Strains Along Upper (Horizontal) Portion of Door Corner.....	47
27 Axial Strains Along Vertical Portion of Door Corner.....	48
28 Shear Strains Along Upper (Horizontal) Portion of Door Corner.....	49
29 Shear Strains Along Vertical Portion of Door Corner.....	50
30 Strain Components Measured by Rectangular Strain Gage Rosette.....	51
31 Principal Stresses in Door Surround Structure Before Doubler Installation.....	52
32 Principal Stress Directions in Door Surround Structure Before Doubler Installation.....	53
33 Composite Doubler Installation and NDI on L-1011 Door Corner.....	56
34 Door Surround Structure Test Article With Strain Gages Installed.....	57
35 Strain Gage Layout for Fuselage Door Structure Tests After Doubler.....	58
36 Total Hoop and Axial Loads Applied to the Test Article After the Doubler was Installed.....	59
37 Axial Load Scatter in Individual Load Cells (North and South); <i>After Doubler Test</i>	60
38 Hoop Load Scatter in Individual Load Cells in Critical, Center Load Area (East and West); <i>After Doubler Test</i>	60
39 Zone 1 (Far Field) Hoop Strains After Doubler Installation.....	62
40 Zone 2 (Upper Door Cut-Out) Hoop Strains After Doubler Installation.....	63
41 Zone 3 (Upper Edge of Doubler) Hoop Strains After Doubler Installation.....	64
42 Zone 4 (Door Cut-Out Corner) Hoop Strains After Doubler Installation.....	65
43 Zone 5 (Vertical Door Cut-Out) Hoop Strains After Doubler Installation.....	66
44 Zone 6 (Forward Portion of Doubler) Hoop Strains After Doubler Installation.....	67
45 Zone 1 (Far Field) Axial Strains After Doubler Installation.....	68
46 Zone 2 (Upper Door Cut-Out) Axial Strains After Doubler Installation.....	69
47 Zone 3 (Upper Edge of Doubler) Axial Strains After Doubler Installation.....	70
48 Zone 4 (Door Cut-Out Corner) Axial Strains After Doubler Installation.....	71
49 Zone 5 (Vertical Door Cut-Out) Axial Strains After Doubler Installation.....	72
50 Zone 6 (Forward Portion of Doubler) Axial Strains After Doubler Installation.....	73
51 Principal Stresses in Region of Doubler Horizontal Leg.....	74
52 Principal Stresses in Region of Doubler Vertical Leg.....	75
53 Principal Stress Directions in Region of Doubler Horizontal Leg.....	76
54 Principal Stress Directions in Region of Doubler Vertical Leg.....	76
55 Comparison of Composite Strains and Nearby Aluminum Strains.....	80
56 Axial and Hoop Load Transfer in Horizontal Leg of Doubler; Zones 2 and 3.....	81
57 4Axial and Hoop Load Transfer Along Door Cut-Out Corner; Zones 4 and 5.....	82
58 Axial and Hoop Load Transfer in Vertical Leg of Doubler; Zone 6.....	83
59 “Before Doubler” Versus “After Doubler” Strains in Far Field; Similarities Show Consistency of Test Set-up Before and After Doubler Installation.....	84
60 Comparison of Strains Before Versus After Doubler Installation Around Perimeter and at Door Handle Cut-Out Region (Hoop Strain Reduction)	85
61 Comparison of Strains Before Versus After Doubler Installation Around Perimeter and at Door Handle Cut-Out Region (Axial Strain Reduction)	86

<u>Figure</u>	<u>Page</u>
62 Comparison of Strains Before Versus After Doubler Installation Around Door Corner Radius (Hoop Strain Reduction)	87
63 Comparison of Strains Before Versus After Doubler Installation Around Door Corner Radius (Axial Strain Reduction)	88
64 Comparison of Principal Stresses Before and After Doubler Installation; Area Around Perimeter and at Door Handle Cut-Out.....	89
65 Comparison of Principal Stresses Before and After Doubler Installation; Area Around Door Corner Radius.....	90
66 Finite Element Model of Fuselage Section and Composite Doubler.....	92
67 Comparison of Experimental and Analytical (FEM) Stresses for the Fuselage Section Without the Composite Doubler.....	93
68 Comparison of Experimental and Analytical (FEM) Stresses for the Fuselage Section Without the Composite Doubler.....	94
69 Comparison of Experimental and Analytical Membrane Stresses in the Aluminum Skin; With the Doubler Installed.....	95
70 Comparison of Experimental and Analytical Strains in the Composite Doubler....	96
71 Comparison of Experimental and Analytical (FEM) Principal Stresses in the Aluminum Skin; With the Doubler Installed.....	97
72 Comparison of Experimental and Analytical (FEM) Principal Strains in the Composite Doubler.....	98
73 Automated Ultrasonic Scanner Inspecting the Composite Doubler on the L-1011 Fuselage Test Article.....	101
74 Sample C-Scan Image from Ultrasonic Inspection of a 72-Ply, Composite Doubler Reference Standard.....	102
75 Engineering Drawing of Subsize Door Corner ("C" Section) Test Specimen.....	108
76 Boron-Epoxy Doubler Dimensions for "C" Section Test.....	109
77 Comparison of Stress Fields for Different L-1011 and Test Specimen Limit Load Conditions.....	110
78 Photograph of Subsize Door Corner Test Article Mounted in a Uniaxial Test Machine.....	111
79 Close-Up Views of Subsize Door Corner Test Article.....	112
80 Strain Gage Locations for Subsize Door Corner Test ("C" Section Test)	113
81 Thru-Transmission Ultrasonic C-Scan of Composite Doubler Before Structural Testing.....	114
82 Pulse-Echo Ultrasonic A-Scan Results Showing Good (Bonded) and Flawed (Disbonded) Doubler Areas.....	115
83 Strains in the Aluminum Substrate (Limit Load Test)	117
84 Strains in the Composite Doubler (Limit Load Test)	118
85 Rosette 1 and 2 Strain Elements (Limit Load Test)	119
86 Strains in the Aluminum Substrate (150% Limit Load Test)	120
87 Cracks Along Base of Doubler, At Interface With Aluminum Plate, Reveal Disbond Growth During Ultimate Failure Test.....	123
88 Cracks Along Base of Doubler, At Interface With Aluminum Plate, Reveal Disbond Growth Around Corner Radius.....	123

Figure	Page
89 Strains in the Aluminum Substrate Up To Specimen Failure.....	124
90 Compressive Strain Along Top of Reinforcing Aluminum Angle (Ultimate Failure Test)	125
91 Strains in Composite Doubler Up To Specimen Failure.....	127
92 Comparison of Critical Strains in Aluminum and Composite Material Showing Load Transfer into Doubler.....	128
93 Rosette 1 and 2 Strain Elements on Aluminum and Composite (Ultimate Failure Test)	129
94 Maximum Principal Stresses at Rosette Gages 1 and 2.....	130
95 Direction of Principal Stress at Rosette Gages 1 and 2.....	131
96 Stress Contour Plot Generated by Finite Element Model for 150% Limit Load Condition (P = 15,420 lbs.)	132
97 Comparison Between Composite Doubler Stresses and Values Predicted by Finite Element Model.....	133
98 Comparison Between Aluminum/Composite Strains and Values Predicted by Finite Element Model.....	134
99 Comparison Between Aluminum Strains and Values Predicted by Finite Element Model.....	135
100 Comparison Between Strains Measured in Angle Bracket and Those Predicted by Finite Element Model.....	136
101 Thru-Transmission Ultrasonic C-Scan of Composite Doubler After Ultimate Failure Test.....	137
102 Comparison of UT Inspection Results Before and After Structural Testing.....	138
103 Composite Doubler Four-Point Bending Test Specimen.....	141
104 Photograph of Four Point Bending Specimen Mounted in Test Fixture.....	142
105 Pure Bending Test Configurations.....	143
106 Bending Strains for Sample 4B-1 Test with Composite on Bottom.....	145
107 Delamination Failure Mode of Four Point Bending Specimen 4B-1 with Composite Laminate on Bottom.....	146
108 Bending Strains for Sample 4B-2 Test with Composite on Top.....	147
109 Delamination Failure Mode of Four Point Bending Specimen 4B-2 with Composite Laminate on Top.....	148
110 Bending Strains for Sample 4B-3 Test with Composite on Bottom.....	149
111 Failed Four Point Bending Specimens 4B-1, 2, and 3.....	150
112 Delta L-1011 Aircraft During Installation of Composite Doubler.....	154
113 Anodized Door Corner Prepared for Doubler Bonding Process.....	155
114 Boron-Epoxy Laminate Following Autoclave Debulk Process.....	155
115 Composite Doubler Positioned on Door for Cure Process.....	156
116 Cure Process - Doubler Beneath Heat Blanket, Insulation, and Vacuum Bag Assembly; Heat Controller is Shown in Foreground.....	157
117 Heat Lamps Positioned Inside Aircraft to Help Maintain Cure Temperature on Back Skin.....	158
118 Composite Doubler Following Cure Process.....	158

<u>Figure</u>	<u>Page</u>
119 Conventional Door Corner Repair with Four, Riveted Metal Plates.....	159
120 Overall View of Composite Doubler on Delta Aircraft Fuselage.....	160
121 Ultrasonic Scanner System Inspecting a Tapered Region of Doubler.....	162
122 Sample Ultrasonic C-Scan from L-1011 Doubler Inspection.....	162
123 Pulse-Echo C-Scan of Composite Doubler Reference Standard Showing Ultrasonic Flaw Imaging.....	163
124 Quality Assurance Wedge Test for Surface Preparation.....	165

<u>Drawing</u>	<u>Page</u>
AANC-2 Biaxial Test Facility with Door Surround Structure Installed.....	187
AANC-3 Substructure Elements Contained in Door Surround Structure Test Article.....	189
AANC-4 Tie-Point Locations and Reinforcement of Test Article for Load Application.....	190
AANC-5 Strain Gage Locations for Structural Test Before Composite Doubler Installation.....	193
AANC-6 Strain Gage Locations for Structural Test After Composite Doubler Installation.....	194
WBAT1006 Vacuum Chamber, L-1011 Widebody Aircraft Test Facility.....	196

1.0 Introduction and Test Series Overview

In 1991, the FAA's William J. Hughes Technical Center established The Airworthiness Assurance NDI Validation Center (AANC) at Sandia National Laboratories. Its primary mission is to support technology development, validation, and transfer to industry in order to enhance the airworthiness and improve the aircraft maintenance practices of the U.S. commercial aviation industry. The Center conducts projects in a myriad of engineering disciplines. The results are placed in the public domain so that the industry at-large can reap the benefits of FAA-funded R & D efforts. To support the Center's goals, the FAA/AANC has set up a hangar facility at the Albuquerque International Airport which contains a series of transport and commuter aircraft. The facility replicates a working maintenance environment by incorporating both the physical inspection difficulties as well as the environmental factors which influence maintenance reliability. Sandia's charter with the FAA includes a wide array of airworthiness assurance disciplines such as nondestructive inspection, structural mechanics, computer science, fire safety, and corrosion.

The development and application of new aircraft repair techniques needs to keep pace with the growing understanding of aircraft structural aging phenomena. One of the primary goals of the Federal Aviation Administration's (FAA) National Aging Aircraft Research Program (NAARP) is to foster new technology associated with the repair of civil aircraft. 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 flaws develop throughout the aircraft's skin and substructure elements. These flaws can take the form of cracks, corrosion, disbonds, dents, and gouges. Composite doublers, or repair patches, provide an innovative repair technique which can enhance the way aircraft are maintained. The high modulus of Boron-Epoxy composite material enables a doubler to pick up load efficiently and effectively when bonded to a metal structure. The load transfer occurs by shear through the adhesive.

The use of bonded composite doublers offers the airframe manufacturers and airline maintenance facilities a cost effective technique to safely extend the lives of their aircraft. Instead of riveting multiple steel or aluminum plates to facilitate an aircraft repair, it is now possible to bond a single Boron-Epoxy composite doubler to the damaged structure. However, before this advanced aircraft repair technique could be accepted for commercial aircraft use, uncertainties surrounding the application, subsequent inspection and long-term endurance of composite doublers had to be addressed. Towards that end, the Federal Aviation Administration sponsored a project at the AANC to validate bonded composite doubler technology and to gain FAA approval for composite doubler use on commercial aircraft.

An industry team consisting of an Original Equipment Manufacturer (Lockheed-Martin) and an airline (Delta Air Lines) was formed. FAA oversight was provided through the Atlanta Aircraft Certification Office (ACO) and the FAA's William J. Hughes Technical Center. The overall goal of this project was to establish the capabilities of composite

doublers and to assist the FAA in developing guidance which will assure the continued airworthiness of these aircraft repairs. The AANC team completed a comprehensive technology development, validation, and application program which established the performance of composite doublers as an improvement over metallic doublers.

Full-Scale Structural and NDI Test Program - 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. Through the use of analysis, laboratory tests, and flight demonstrations on an in-service L-1011 airplane, this project investigated composite doubler design, fabrication, installation, structural integrity, and NDI issues. This report focuses on a series of structural tests and NDI inspections that studied the structural response and damage detection in Boron-Epoxy doublers installed on representative aircraft structure. The tests were conducted to evaluate the general use of bonded composite doublers to repair metallic aircraft structures. Specific emphasis was placed on select areas of the doubler design which experience the highest strains and must survive large peel stresses. Specimen geometry and load conditions were engineered to exceed design strains corresponding to the worst-case flight conditions.

The primary test article was a large fuselage section cut from a retired All Nippon Airways (ANA) L-1011 aircraft. The fuselage test article included a passenger door (P3 passenger door) cut-out and contained all substructure frame, longeron, and stringer elements. These full-scale tests studied load transfer into composite doublers using simulated flight conditions of cabin pressure and axial stress. In addition, the fuselage structure tests evaluated the effectiveness of composite doublers in reducing the stresses around a damaged or weakened structure. This test bed was instrumented with strain gages and subjected to simulated fuselage pressure differential loads using a customized fuselage test fixture (see Sections 2.0 and 3.0).

Two other test configurations - consisting of composite doublers mounted on simulated aircraft panels - were examined in order to assess the response of composite doublers in worst-case shear and bending load scenarios. The primary issues to be addressed were doubler delamination and disbond especially in high peel stress regions. Tension, shear and bending loads were applied to induce interply delaminations or disbonds between the aluminum and the laminate. The test results described in this document verify Lockheed's analysis which also indicated acceptable interply and laminate peel stresses. These two test configurations were loaded to failure in order to: 1) experimentally determine a design margin for the onset of delaminations and disbonds, and 2) verify the safety factors associated with current doubler design methodologies.

The reliability and efficiency of composite doubler inspection operations are key to ensuring the continued airworthiness of these doublers. The AANC used comprehensive validation exercises to quantify the reliability and implementation costs of a complete inspection methodology. This report describes the utilization of conventional and advanced NDI techniques to detect flaws in bonded composite doublers and their parent

aluminum material. In this project, close consultation with the FAA and the air transport industry was pursued in order to meet the necessary requirements. Active industry involvement was essential to the efficient execution of the AANC activities and ensured the relevance of any resulting recommendations.

Composite Doubler Use in Commercial Aircraft Repair - In addition to developing general information to support composite doubler use, this program introduced composite doubler technology to the U.S. commercial aircraft fleet. 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. By focusing on a particular commercial aircraft application - reinforcement of the L-1011 door frame - and encompassing all "cradle-to-grave" issues such as design, analysis, installation, long term durability, and nondestructive inspection, this program was designed to assess the capabilities of composite doublers.

Although a specific application provided the foundation for this project, 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, quality assurance, nondestructive inspection, and FAA/industry oversight of these doublers. The first use of the doubler performance assessment presented herein was to support the structural analysis of the L-1011 composite doubler (Reference [1]).

AANC and Validation Issues for Bonded Composite Doublers - The use of composite doublers in commercial aviation must address issues such as installation, 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 validated composite repair technology will increase.

Efforts to bring newly developed technology to the field can encounter some obstacles. Field personnel may be reluctant to accept new repair practices and associated NDI procedures for several reasons. The technology may not be fully field tested; there may not be enough experience under field conditions. It may require the purchase of new equipment and aircraft maintenance facilities want proof that the capital outlay is justified. Further, it may require retraining personnel. The AANC was set up to comprehensively address these obstacles and reduce the risks involved in introducing new maintenance practices to the field. The Center does this by evaluating the performance of new hardware, software, and NDI procedures; by demonstrating and documenting the performance of systems; and by supporting the economic analyses of new maintenance practices.

Bonded Composite Doublers on Aircraft Structure - 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 safely extend the lives of their aircraft. Flight demonstrations and operational testing have confirmed that under proper conditions, composite doublers can provide a long lasting and effective repair or structural reinforcement [2-5]. Reference [6] describes a series of analytical models which were developed to study the stress field in and around composite doublers and the crack growth life extension resulting from composite doubler use.

The comprehensive goal of these AANC validation efforts are to address any remaining uncertainties about composite doublers and thus, assure: 1) proper design and installation processes, and 2) the continued safe operation of the doublers over time. Through the use of laboratory test structures and flight demonstrations on an in-service L-1011 airplane, this study investigated general composite doubler design, fabrication, installation, structural integrity, and nondestructive evaluation.

Repairs and reinforcing doublers using bonded composites have numerous advantages over mechanically fastened repairs. Adhesive bonding eliminates stress concentrations, and new potential crack initiation sites, caused by additional fastener holes. Composites are readily formed into complex shapes permitting the repair of irregular components. Also, composite doublers can be tailored to meet specific anisotropy needs thus eliminating the undesirable stiffening of a structure in directions other than those required. Other advantages include corrosion resistance, a high strength-to-weight ratio, and potential time savings in installation. The economic advantages stem primarily from time savings in installation and the secondary effect of reduced aircraft downtime. Exact dollar values depend on the complexity of the repair installation and the number of repairs installed.

Typical Composite Doubler Installation and NDI - Figure 1 shows a typical bonded composite doubler repair over a cracked parent aluminum structure. Sample composite doubler installations, showing two families of potential aircraft repair applications, are shown in Figure 2. The number of plies and fiber orientation are 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.

The taper at the edge of the doubler is used to produce a gradually increasing stress gradient in the area of primary load transfer. In some applications, such as the L-1011 door corner doubler design, lightning protection is provided by a copper wire mesh which is imbedded in an adhesive film and applied as a top ply over the doubler. The lightning protection ply has a larger footprint than the composite laminate in order to provide a conductive link between the copper mesh and the surrounding aluminum skin. Finally, a top ply of fiberglass is installed to supply mechanical and environmental protection for the installation.

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

**(a) Sample Fuselage Skin Repair
(composite doubler approx. 12" X 10")**

**(b) Sample Door Corner Repair
(composite doubler approx. 5 ft.² footprint)**

Figure 2: Sample Bonded Composite Doubler Installations Showing Two Families of Potential Repair Applications

1.1 Doubler Design Guidelines

Moving Technology into Routine Maintenance Programs - Reference [7] describes airframe maintenance programs from an airlines' perspective. It describes sources of aircraft damage and how the damage is addressed through inspection and repair tasks. A detailed maintenance program is required to ensure that an aircraft can be operated safely for an extended period of time. The emphasis of the maintenance program must be

constantly adjusted to cater to the age of the fleet [7]. These revisions require the co-operation of the aircraft manufacturers, the airlines, and the airworthiness authority to ensure that changes made are technically correct, stringent enough to assure the aircraft's continued safety, with due consideration to the cost to the airline. Aircraft structures suffer continuous degradation throughout their service lives. Corrosion, fatigue, impact and accidental damage from assorted ground activities all contribute to this structural degradation. The airlines and aircraft maintenance depots accomplish permanent and interim repairs suited to the situation and in line with current industry practices. These acceptable repair practices must be continuously revisited and expanded to take advantage of new materials, new processes, and new techniques that offer both engineering and economic advantages. Through the steady and comprehensive introduction of test data, analyses, and in-service composite doubler installations on commercial aircraft a critical database is being assembled to accurately guide enhancements to formal maintenance programs. This is an important step in the evolution of composite doubler applications since it will eventually eliminate the need for each bonded composite repair to be preceded by a lengthy research and testing program.

Doubler Design Guidelines – References [8-11] provide an excellent set of guidelines for designing composite doubler repairs. The primary issues to be addressed include the optimum location, size, shape, and laminate taper for the patches. The major factors that determine the patch design parameters are the stress levels at the repaired flaw, the stress levels in the composite doubler (maximum allowable fiber stresses), and the stress levels in the adhesive layer between the doubler and the aluminum skin. The important, fundamental results produced by Jones and Callinan in ref. [8] are worth reviewing in some detail in order to prepare for the damage tolerance discussions later in this report.

The ref. [8] crack repair study used a unidirectional Boron-Epoxy laminate (fiber perpendicular to crack) as a baseline design. The study found that for patches that cover the entire length of the crack, a one ply (0.0057" thick) patch reduces the stress intensity factor to 33.5% of its value for the unpatched crack. Furthermore, the rate of decrease in the stress intensity factor, K_{IP} , as the patch thickness increases was found to be quite low. A six layer patch, for example, produced a K_{IP} value of 19.5% of the unpatched value. Thus, an increase in patch thickness by a factor of 6 only produces an additional 14% reduction in the stress intensity factor. Another important finding of the ref. [8] study was that in thin patches, the stresses over the flaw (in both the patch and the adhesive) are critical. However, as the patch thickness is increased, the shear stresses in the adhesive at the edges of the patch footprint become critical and may exceed allowable limits.

The findings summarized above produced a series of design requirements that are necessary to maintaining the structural integrity of a composite doubler repair [8].

- 1) Fiber Strain – For a maximum Boron fiber strain of 5,000 $\mu\epsilon$, the stress in the fibers must satisfy

$$\sigma_f \leq 0.005 E_{11} \quad (1)$$

where E_{11} is Young's modulus in the direction of the fibers (28.0×10^6 psi). When the fiber stress concentration, K_f , is considered the governing equation for fiber strain becomes

$$\sigma_f = K_f \sigma \leq .14 \text{ lb/in}^2 \quad (2)$$

Thus, if the applied stress, σ , is known the maximum permissible value of K_f and the minimum permissible patch thickness can be determined.

- 2) Shear Stress in Adhesive – For a maximum allowable adhesive shear stress of 7 KSI and considering the stress concentrations at the crack and at the edge of the patch, the following design equations must be satisfied

$$\tau_{a(c)} = K_{a(c)} \sigma \leq 7,000 \text{ lb/in}^2 \quad (3)$$

$$\tau_{a(e)} = K_{a(e)} \sigma \leq 7,000 \text{ lb/in}^2 \quad (4)$$

Equations (2) and (3) can be solved to produce a doubler thickness to satisfy the stress requirements at the crack (for both laminate and adhesive). However, this often produces adhesive stresses at the edge of the doubler that exceed those allowed by eq. (4). Ref. [8] showed that the adhesive stresses at the edge of the doubler can be reduced to admissible levels by stepping the thickness of the doubler from one or two plies at the outer perimeter to full thickness over a taper region. The length of the taper region is normally chosen to produce an edge taper ratio (ratio of thickness increase to width of taper) of 10 to 30. Figure 1 shows a typical doubler edge taper used to gradually transfer load from the aluminum skin and reduce the shear stress in the adhesive. An ongoing study at the AANC regarding the application of composite doublers to DC-10 skin damage encountered this need to balance doubler thickness with an appropriate edge taper [12].

Ref. [8] provides quantitative stress reductions corresponding to the use of stepped doublers. It also lists the effects of varying adhesive thickness and the difference between single- and double-sided repairs.

Doubler Installation Examples - Numerous composite doubler repairs have been installed on military aircraft (both U.S. and foreign). Significant advances have been made on commercial aircraft applications [13-20], however, most of these have occurred on non U.S. certificated aircraft. Hence, the basis of this FAA-sponsored program is to safely and comprehensively introduce the use of composite repairs to the U.S. commercial aircraft fleet. Some examples of successful composite doubler repairs follows. Reference [14] describes a well-documented and extensive composite repair program carried out by the U.S. Air Force. In this program, composite doublers were used to repair cracks emanating from weep holes in the C-141 wing plank risers.

Reference [15] gives an overview of the applications pursued by the Aeronautical and Maritime Research Laboratory in Australia which has pioneered the use of this technology. These composite doubler applications include crack repairs on an F111 wing pivot fitting and stiffening of a corroded keel beam in an Ansett Airlines 767 aircraft. References [17-20] describe the program being led by the FAA's Airworthiness Assurance Center at Sandia National Labs. This program is working to validate the use of composite doublers and streamline their introduction to the U.S. commercial aircraft fleet. An L-1011 door corner repair is now operating in the Delta Air Lines fleet while a family of DC-10 fuselage skin repairs are being readied for a pilot program with Federal Express. A series of demonstration programs [13, 15] have produced hundreds of simulated repair installations on commercial and military aircraft. Since the parent structures were not flawed in these demonstration programs, there were no detrimental effects stemming from stress risers beneath the doublers. However, these composite doubler programs are accumulating thousands of successful flight hours/cycles that can be used to further establish the viability of composite doubler repair technology.

1.2 Damage Tolerance and Fracture Control Plan

1.2.1 Damage Tolerance and Analysis Methodologies

Damage Tolerance of Composite Doublers – Inspection requirements (sensitivity and inspection intervals) are driven by Damage Tolerance Analyses (DTA). However, the stack of metal parent materials (isotropic), composite lamina (anisotropic), and adhesive layers makes the analysis quite complex and hinders the calculation of an exact DTA. It is difficult to determine the effects of flaw size and the point at which a flaw size/location becomes critical. This is especially true of disbond, delamination, and porosity flaws. Thus, an increased emphasis is placed on quantifying the probability that a flaw of a particular size and location will be detected by a piece of NDT equipment. *In any surveillance of aircraft structure there are three main aspects to the inspection requirements: 1) the damage tolerance analysis (DTA) which determines the flaw onset and growth data (especially critical flaw size information), 2) the sensitivity, accuracy, and repeatability of NDI techniques which, in concert with the DTA, establishes the minimum inspection intervals, and 3) the impediments that the NDI techniques must contend with while achieving the required level of sensitivity.* In addition to the Ref. [17] report, detailed discussions on damage tolerance assessments for composite doubler installations are presented in references [21-25].

The reference [24] observations mirror one of the primary results obtained in the damage tolerance assessment produced in this project [17]: adhesively bonded doublers are extremely damage tolerant to large disbonds and other detrimental conditions such as impact and hot-wet conditioning. If, in fact, disbond and delamination flaws do not grow even under extreme environmental conditions, then an acceptable design should be predicated on the fact that the stresses in the adhesive are kept below a limiting or threshold value. As a result, reference [24] introduces an essential design methodology that considers damage tolerance. It uses a fatigue threshold load, P_f , and a fatigue

threshold strain, ϵ_f , below which irreversible damage in the adhesive will not occur. For thin skin repairs, the equations used to determine the threshold load and strain values are as follows:

$$P_f = 2 (t W_f E T)^{1/2} \quad (5)$$

$$\epsilon_f = 2 (t W_f E / T)^{1/2} \quad (6)$$

where,

t = thickness of the adhesive

T = thickness of the adherend (skin)

E = Young's modulus of the skin

W_f = threshold value of the strain energy density of the adhesive

W_f can be determined experimentally [25]. Ref. [24] also describes the maximum load, P_{max} , that can be carried by a bond in a symmetrical bonded joint as,

$$P_{max} = 2 (t W_c E T)^{1/2} \quad (7)$$

where W_c is the maximum strain energy density of the adhesive. Thus, composite doubler repair design guidelines are that P_{max} is greater than the ultimate load for the repaired structure and that P_f is greater than the limit load. Ref. [24] also points out that these critical design variables are affected by the loading rate. A conservative estimate for P_{max} can be obtained by using the value of the maximum von Mises equivalent stress in the adhesive, σ_e , as measured in high strain rate tests. For FM73, the adhesive used in this study, $\sigma_e = P_{max} = 5,800$ psi and the threshold stress $\sigma_{th} = 3,600$ psi. This analysis approach clearly shows the importance of the adhesive in determining the overall performance of the bonded repair. Ref. [24] goes on to point out the effects of the inelastic strain build-up in the adhesive layer that can accumulate with each load cycle. This hysteresis must be considered when determining the loads and fatigue cycles necessary to reach the maximum strain. The approach outlined above can be used to certify that a composite doubler design will satisfy the damage tolerance provisions of the U.S. Federal Aviation Regulations (FAR) Part 25.

The abilities of nondestructive inspection techniques to meet the DTA flaw detection requirements are presented in references [17-18, 26-28]. The fundamental result from the ref. [18] NDI study is that a team of NDI techniques can identify flaws well before they reach critical size. Crack detection in the parent aluminum material can be accomplished using conventional eddy current and X-ray techniques. Also, ultrasonic and thermography methods have been successfully applied to the problem of disbond and delamination detection.

Analysis of Composite Repairs – Numerous efforts have developed, refined, and advanced the use of methodologies needed to analyze composite doubler installations. Obviously, this is a critical element in the repair process since a badly implemented repair is detrimental to fatigue life and may lead to the near-term loss of structural integrity. The difficulties associated with analyzing the stress fields and flaw tolerance of various composite doubler designs and installations are highlighted in references [5], [21], and [23]. Doubler design and analysis studies [5-6, 24, 29-36] have led to computer codes and turn-key software [37-38] for streamlining the analyses. These developments have taken great strides to eliminate the approximations and limitations in composite doubler DTA. In references [21] and [31], Baker presents an extensive study of crack growth in repaired panels under constant amplitude and spectrum loading. The installation variables evaluated were: 1) doubler disbond size, 2) applied stress, 3) doubler thickness, 4) min-to-max stress ratios (R ratio), and 5) temperature.

In refs. [21] and [31], a predictive capability for the growth of cracks repaired with composite doublers was developed using Rose's analytical model [32] and experimental fatigue studies. The important stress variables include the stress range, $\Delta\sigma_\infty$, and stress ratio, R , where,

$$\Delta\sigma_\infty = \sigma_{\max} - \sigma_{\min} \quad (8)$$

$$R = \sigma_{\min} / \sigma_{\max} \quad (9)$$

A Paris-type crack growth relationship is assumed between da/dN and ΔK for the repaired crack such that,

$$da/dN = f(\Delta K, R) = A_R \Delta K^{n(R)} \quad (10)$$

where a is the crack length, N is the number of fatigue cycles, and A_R and $n(R)$ are constants for a given R value. Tests results in [21] and [31] produced crack growth constants and were used to validate the model for crack mitigation effects of composite doublers. It was determined that Rose's model for predicting the stress-intensity range, ΔK , provides a good correlation with measured crack growth data (da/dN), however, anomalies were observed in the cases of temperature and R-ratio effects. Estimates of crack growth in composite doublers containing various disbond sizes were also determined.

In lieu of using computationally expensive, three-dimensional finite elements, reference [33] presents the use of a simple analysis using Mindlin plate theory. The aluminum parent plate and composite doubler are modeled separately by the Mindlin plate finite element (using ANSYS) and the adhesive layer is modeled with effective springs connecting the doubler to the aluminum plate. The model showed excellent agreement with existing boundary element solutions and three-dimensional finite element solutions when calculating the stress intensity factors for double-sided patches. However, the Mindlin plate theory produced appreciably different K values than a three-dimensional

FEM for single-sided doubler repairs. These results highlight some of the difficulties in modeling composite doubler repairs and the need for innovative schemes to address single-sided repairs.

Complete three dimensional FEM analyses of composite doubler repairs are provided in reference [34]. Ref. [34] addressed one-sided repairs and showed that the stress intensity factor reaches an asymptotic value, rather than increasing indefinitely as would be the case for an unrepaired crack. Furthermore, the stress intensity factor can be approximated by an analytical expression that provides a close, yet conservative, estimate for repairs over all crack lengths. While the stress intensity factor for a one-sided repair is much less than the unrepaired configuration, it exceeds the value for the corresponding two-sided repair. This analytical finding supports test results that show the secondary bending induced by the shift in neutral axis in a one-sided patch has a detrimental effect on the efficiency of bonded composite repairs.

No discussion of design and analysis methodologies is complete without a mention of a closely-coupled validation program. Reference [35] presents a detailed design and analysis validation effort to substantiate a safety-critical repair to an F-111 lower wing skin. The repair substantiation involved both detailed FEM stress analysis and structural testing ranging from coupons to quasi full-scale specimens representing a spar-stiffened wing box structure. The intercomparison of results provides a high level of confidence that the static residual strength has been restored to the original ultimate strength levels. It also provides a good foundation for the subsequent management of the repaired structure by establishing inspection intervals with sufficient safety factors.

The test results presented in this document and in reference [17] supplement the composite doubler analyses efforts described above and provide a basis of comparison with computational models. Analysis improvements, however, must be validated by successful flight performance of operational doublers. This can only be accumulated over a long period of time. Continued surveillance of installed doublers will provide quantitative flight performance history and produce a conservative safety factor. Thus, regardless of the excellent damage tolerance results accumulated to date, NDI will continue to play a critical role in the use of composite doublers.

This full-scale test report, along with references [17-19], describes the validation program that accompanied the L-1011 door corner repair. In these four documents, the attempts to generalize the performance test results are discussed. Every effort was made to design the test specimens and extrapolate the results to as wide a range of composite doubler repairs as possible. The overall goal in this approach is to minimize and optimize the testing that must compliment each new composite doubler installation. In order for composite doubler technology to be useful to the commercial aircraft industry, the design-to-installation cycle must be streamlined. An ongoing study at the AANC is addressing composite doubler repairs on DC-10 fuselage skin [20] with the goal of streamlining the design, validation, and certification process. The end result will be the revision of the DC-10 Structural Repair Manual (alternate repairs for existing riveted metallic doublers) thus allowing more rapid and widespread use of specific doubler

repairs. It should be noted that a closely monitored pilot program will be completed prior to any revision of the DC-10 Structural Repair Manual.

Need for Damage Tolerance Assessments - One of the primary concerns surrounding composite doubler technology pertains to long-term survivability, especially in the presence of non-optimum installations. This test program demonstrated the damage tolerance capabilities of bonded composite doublers. The fatigue and strength tests quantified the structural response and crack abatement capabilities of Boron-Epoxy doublers in the presence of worst case flaw scenarios. The engineered flaws included cracks in the parent material, disbonds in the adhesive layer, and impact damage to the composite laminate. Environmental conditions representing temperature and humidity exposure were also included in the coupon tests.

1.2.2 Damage Tolerance Establishes Fracture Control Plan

Establishing Damage Tolerance - Damage tolerance is the ability of an aircraft structure to sustain damage, without catastrophic failure, until such time that the component can be repaired or replaced. The U.S. Federal Aviation Requirements (FAR 25) specify that the residual strength shall not fall below limit load, P_L , which is the load anticipated to occur once in the life of an aircraft. This establishes the minimum permissible residual strength $\sigma_P = \sigma_L$. To varying degrees, the strength of composite doubler repairs are affected by crack, disbond, and delamination flaws. The residual strength as a function of flaw size can be calculated using fracture mechanics concepts. Figure 3 shows a sample residual strength diagram. The residual strength curve is used to relate this minimum permissible residual strength, σ_P , to a maximum permissible flaw size a_P .

Figure 3: Residual Strength Curve

A fracture control plan is needed to safely address any possible flaws which may develop in a structure. Nondestructive inspection is the tool used to implement the fraction control plan. Once the maximum permissible flaw size is determined, the additional information needed to properly apply NDI is the flaw growth versus time or number of cycles. Figure 4 contains a flaw growth curve. The first item of note is the total time, or cycles, required to reach a_p . A second parameter of note is a_d which is the minimum detectable flaw size. A flaw smaller than a_d would likely be undetected and thus, inspections performed in the time frame prior to n_d would be of little value. The time, or number of cycles, associated with the bounding parameters a_d and a_p is set forth by the flaw growth curve and establishes $H(\text{inspection})$. Safety is maintained by providing at least two inspections during $H(\text{inspection})$ to ensure flaw detection between a_d and a_p .

Inspection Intervals - An important NDI feature highlighted by Fig. 4 is the large effect that NDI sensitivity has on the required inspection interval. Two sample flaw detection levels $a_{d(1)}$ and $a_{d(2)}$ are shown along with their corresponding intervals $n_{d(1)}$ and $n_{d(2)}$. Because of the gradual slope of the flaw growth curve in this region, it can be seen that the inspection interval $H_1(\text{inspection})$ can be much larger than $H_2(\text{inspection})$ if NDI can produce just a slightly better flaw detection capability. Since the detectable flaw size provides the basis for the inspection interval, it is essential that quantitative measures of flaw detection are performed for each NDI technique applied to the structure of interest. This quantitative measure is represented by a Probability of Detection (PoD) curve such as the one shown in Figure 5. Regardless of the flaw size, the PoD never quite reaches 1 (100% possibility of detection). Inspection sensitivity requirements normally require a 90-95% PoD at a_p . For any given inspection task, the PoD is affected by many factors such as: 1) the skill and experience of the inspector, 2) accessibility to the structure, 3) exposure of the inspection surface, and 4) confounding attributes such as underlying structure or the presence of rivets. Thus, the effects of circumstances on PoD must be accounted for in any NDI application and associated fracture control plan.

As an example of the DTA discussed above, reference [1] describes the design and analysis process used in the L-1011 program. It presents the typical data - stress, strength, safety factors, and damage tolerance - needed to validate a composite doubler design. The design was analyzed using a finite element model of the fuselage structure in the door region along with a series of other composite laminate and fatigue/fracture computer codes. Model results predicted the doubler stresses and the reduction in stress in the aluminum skin at the door corner. Peak stresses in the door corner region were reduced by approximately 30% and out-of-plane bending moments were reduced by a factor of 6. The analysis showed that the doubler provided the proper fatigue enhancement over the entire range of environmental conditions. The damage tolerance analysis indicated that the safety-limit of the structure is increased from 8,400 flights to 23,280 flights after the doubler installation (280% increase in safety-limit). It established an inspection interval for the aluminum and composite doubler of 4,500 flights.

Figure 4: Crack Growth Curve Showing Time Available for Fracture Control

Figure 5: Probability of Flaw Detection vs. Flaw Size

1.2.3 Damage Tolerance Testing

A series of fatigue coupons were designed to evaluate the damage tolerance performance of bonded composite doublers. The general issues addressed were: 1) doubler design - strength, durability, 2) doubler installation, and 3) NDI techniques used to qualify and accept installation. 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. The flaws included fatigue cracks (unabated and stop-drilled), aluminum cut-out regions, and disbond combinations. The most severe flaw scenario was an unabated fatigue crack which had a co-located disbond (i.e. no adhesion between doubler and parent aluminum plate) as well as two, large, 1" diameter disbonds in the critical load transfer region of the doubler perimeter. Tension-tension fatigue and residual strength tests were conducted on the laboratory specimens. The entire damage tolerance assessment program and the test results are presented in the following chapters. Through-transmission ultrasonics, resonance UT, and eddy current inspection techniques were interjected throughout the fatigue test series in order to track the flaw growth.

General Use of Results - The objective of this test effort was to obtain a generic assessment of the ability of Boron-Epoxy doublers to reinforce and repair cracked aluminum structure. By designing the specimens using the nondimensional stiffness ratio, it is possible to extrapolate these results to various parent structure and composite laminate combinations. The number of plies and fiber orientations used in these tests resulted in an extensional stiffness ratio of 1.2:1 $\{(Et)_{BE} = 1.2 (Et)_{Al}\}$. Independent Air Force [39] and Boeing studies [40-41] have determined that stiffness ratios of 1.2 to 1.5 produce effective doubler designs. Lockheed-Martin has also used this range of stiffness ratios in military composite doubler designs.

1.3 Full-Scale Structural Testing Philosophy

The goal of structural testing is to determine how structures behave statically and dynamically under realistic loading conditions. It is often necessary to utilize full-scale or large-scale test articles in order to adequately account for all aspect of a structure's configuration - especially the unique aspects of connections and fastened joints - and to properly apply the operating boundary conditions. In this program, use of an actual aircraft fuselage section allowed for the most accurate assessment of stress concentrations, residual stresses, and variations in structural response across fuselage cut-outs and doublers. The understanding and insight gained from these tests include: 1) correlation with analytical models of structural behavior, 2) evidence that the composite-reinforced structure can meet its intended strength, deflection, durability, and global compliance requirements, and 3) proof that the doubler and surrounding structure can meet its performance requirements such as fatigue resistance and damage tolerance.

In the case of the aircraft structures addressed here, full-scale tests can be used to determine how much the stress has to be reduced to increase the life a required amount.

This is achieved through appropriate comparisons between the test strain data and S-N fatigue curves for the structure's material. The comparison can be based on similar loading as described by the stress ratio, R. Similarly, S-N characteristics and stress data from aircraft panel tests can be used to assess damage accumulation and make life predictions based on damage tolerance in the presence of different R values. This process is an approximation, however, it serves as a guide for considering the stress levels and the affect of stress reductions on the life of an aircraft structure.

Multiple engineering factors and the need for approximating assumptions hinder the theoretical analyses of structural response. As a result, the experimental approach provides the best means for assessing structures by using full-scale specimens under actual service conditions with all physical conditions present. However, it should be noted that the experimental process does not preclude the need for analysis. It provides a powerful tool to supplement theory and produces a "reality check" to validate analytical approaches.

1.4 Conformity Inspection and FAA Oversight

In this program, appropriate conformity checks and FAA oversight was obtained on all aspects of specimen fabrication, testing, and data acquisition. The following items were witnessed by the FAA or an FAA designated representative. The test plan was reviewed and approved by a Designated Engineering Representative.

1. Fabrication of the test specimens - composite doubler fabrication and installation.
2. Impact and hot-wet conditioning of test specimens.
3. Conformity inspection of coupon test articles to assure adherence to specified structural configuration.
4. Verification that the calibration and operation of test equipment was current.
5. Verification of strain gage locations.

This Page Left Intentionally Blank

PART I: CABIN PRESSURIZATION TEST OF FUSELAGE DOOR SURROUND STRUCTURE

This Page Left Intentionally Blank

2.0 Fuselage Door Surround Structure Tests

Full-scale tests were conducted on large fuselage panels cut from retired aircraft. These tests studied load transfer around composite doublers subjected to simulated flight conditions of fuselage pressure and axial stress. The test series utilized a complete L-1011 door surround structure to: 1) assess general composite doubler response in representative flight load scenarios, 2) evaluate the effectiveness of composite doublers in reducing the stresses around a damaged or weakened structure and the associated crack mitigation capabilities, and 3) verify design and analyses methodologies for composite doubler technology. The test bed was instrumented with strain gages and subjected to simulated fuselage pressure differential loads using a customized fuselage test fixture described in this chapter. 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 that can be used by airline technicians on commercial aircraft in the field.

The door surround structure modification and test configuration information is presented in section 2.2 and is contained in the attached engineering drawings numbered AANC-2 through AANC-6 (see Appendix A). The complete composite doubler design is contained in the Lockheed engineering drawing LCC-7622-378 [42] and is described further in Section 2.2.4.

2.1 Participants in Test Article Modifications and Testing

1. Original fabrication of the L-1011 aircraft is per Lockheed design drawings. The history of the aircraft which supplied the door surround structure test article is described in Section 2.2.
2. The door surround structure test article was reinforced around the perimeter - to accommodate the test load application points - by the Sandia Labs AANC as per Section 2.2.2
3. The Boron-Epoxy doubler was fabricated and installed on the aluminum substrate by Delta Air Lines as per the Reference [1] doubler design drawing and the Reference [43] process specifications.
4. Strain gages were installed on the test article by the Sandia Lab's AANC as per Section 2.3.2. Structural tests were performed by the Sandia Lab's AANC as per Section 2.4.
5. Test article review, and associated modifications and tests, were conformed and/or witnessed by the FAA or its authorized designee as per 2.1.1.

2.1.1 Conformity Inspection

The following items were witnessed by the FAA or an FAA designated representative.

1. Conformity inspection of the L-1011 door surround structure component (also referred to as the test article herein) before the composite doubler was installed to assure adherence to the specified structural configuration.
2. Composite doubler fabrication.
3. Installation of composite doubler on test article.
4. Conformity inspection of the L-1011 door surround structure component after the composite doubler was installed to assure adherence to the specified structural configuration.
5. Verification that the calibration and operation of test equipment was current.
6. Verification of test set-up (for both "before doubler installation" and "after doubler installation" tests). Witness all tests.
7. Verification of strain gage locations (for both "before doubler installation" and "after doubler installation" tests).

2.2 Description of Test Article

2.2.1 L-1011 Fuselage Structure

The pressure vessel of the L-1011 consists of the flight station, a constant diameter fuselage, and the afterbody. The fuselage is a conventional semi-monocoque structure, except in the window belt area, with a circular cross section having a radius of 117.5 in. The P-3 door is located just forward of the wing at Frame Station (FS) 878 on both sides of the aircraft. Figure 6 shows the basic L-1011 structure and the location of this door. The fuselage shell is assembled from large panels of aluminum alloy skins, frames, stringers and doublers. Stringers consist of metal "J" sections spaced approximately 8.5" apart. The metal frames run circumferentially at a nominal spacing of 20 inches.

The P-3 passenger door is a plug type door located on the left and right sides of the shell between FS 855 and FS901. The surrounding structure, shown in Figure 7, is designed to carry fuselage shell and door pressure loads around the cutout. The structural reinforcement consists of sill longerons at Water Line (WL) 280 and WL 200 and edge frames at FS855 and FS901. Bonded doublers extend fore and aft between FS 829 and FS 923 and two stringer bays above and below the cutout. Aluminum triplers consist of 7075-T6 plate riveted to the surrounding structure to provide skin edge reinforcement. Nominal thickness of the skin construction is as follows: 1) outer skin = 0.068" - 0.075", 2) doubler = 0.071", and 3) tripler = 0.285" thick.

2.2.2 Door Surround Structure Before Composite Doubler Installation

The test article was a door surround structure cut from a retired L-1011 aircraft. It had planform dimensions of approximately 141" H (151" arc length in the hoop direction) X 114.75" W. The fuselage test article included a passenger door (P3 passenger door) cut-

out and contained all substructure frame, longeron, and stringer elements. Figure 8 shows a photograph of the door surround structure section prior to being cut from the L-1011 fuselage. The source of the L-1011 door surround structure test bed was an Al Nippon Airways (ANA) aircraft, S/N 1154P, Manufacture Date: November 11, 1977. The structure matches the P3 door configuration chosen for the on-aircraft installation. The door surround structure was cut from the right side of the aircraft. It contained eight circumferential frames and six longitudinal stringers, as well as, the upper and lower longeron around the door cut-out.

Figure 6: L-1011 Basic Structure

Applicable Lockheed drawings for this P3 door surround structure configuration at fuselage station 878 are: 1) structure surround passenger door, fuselage station 878 installation drawing 1552177, 2) structural installation assembly - upper fuselage section 3 drawing 1503646, and 3) fuselage complete - structure (L-1011-1) drawing 1504973-103. During all structural tests, a door was mounted and sealed in the fuselage structure.

Figure 7: Geometry of Door Surround Installation

Drawings AANC-2 and AANC-3 show the boundaries of the door surround structure test article. In the longitudinal direction, the specimen contained circumferential frame stations 923, 912, 901, 866, 883, 870, 865, 855, 846.25, and 829. In the hoop direction, the specimen contained stringer numbers 12 through 17, as well as, the upper and lower longeron around the P3 door cut-out. Figure 9 shows structural details of the upper, forward corner of the door surround structure. It contains a close-up of the door corner area which was reinforced with the composite doubler. The drawing also highlights three potential crack origins (listed as "crack ref") at the corners of the passenger door and emergency door handle release cut-outs. This figure provides an important reference point for later discussions on strain gage placement and test loads (see Section 2.2).

(a) Structural Configuration of Passenger Door Area

(b) Section Cut from Retired L-1011 Aircraft for Full Scale Tests

Figure 8: L-1011 Door Surround Structure Test Article

Figure 9: Structural Detail of Area to Be Repaired by Composite Doubler

2.2.3 Reinforcement of Test Article for Load Application Points

The perimeter of the door surround structure was reinforced with bonded and bolted doubler plates to connect to the tension rams and clevises in the axial direction and the turnbuckle restraints in the hoop direction (see also Section 2.3.1 on Test Facility Description). The reinforcing plates were bonded to the parent fuselage structure using Hysol EA-9394, two part structural adhesive which had a shear strength of 4,500 psi.

Figure 10 shows the load application and restraint points on the door surround structure (see also drawing AANC-3 and AANC-4 in Appendix A). Figures 11-12 show the series of reinforcing plates that were attached to the test article (see also drawing AANC-4 sheets 2 and 3). The primary purpose of the reinforcing plates was to eliminate the possibility of local material failure at the load application points. In addition, the particular stack up of plates shown in the drawings was designed to apply the loads at approximately the location of the section's neutral axis. This, in turn, minimized the amount of bending imparted to the structure. Overall, the number of attachment points were selected to apply the desired far-field stresses to the structure such that a uniform stress field was produced in the area of interest around the composite doubler (i.e. eliminate test boundary condition effects).

Figure 10: Load Application Points on the Fuselage Test Panel

Figure 11: Reinforcement of Tie Point Locations in the Axial Direction

Figure 12: Reinforcement of Tie Point Locations in the Hoop Direction

In the longitudinal direction, 12 tie points were used on both sides of the structure. Eight (8) of the tie points were at each stringer/skin assembly (reference stringers 12 through 17). In addition to being attached to the fuselage skin, the load plates were also connected to the vertical web of the stringer in order to transfer the proper load into the stringer. Four (4) of the tie points were around the window cut-out. The window belt area does not have accompanying stringers so the reinforcing plates were placed symmetrically about the fuselage skin as shown in drawing AANC-4, sheet 2. Figure 13 is a photograph showing several of the stringer/skin reinforcement joints in the longitudinal direction. Also shown in the photo are several whiffle tree and load train assemblies.

Figure 13: Stringer/Skin Reinforcement Assembly in the Longitudinal Direction

In the hoop direction, 16 tie points were used on both sides to restrain the door surround structure during the application of the vacuum (pressure differential loads). The reinforcement design in the hoop direction included upper and lower plates which were bonded and bolted to the fuselage skin in a symmetric lay-up as shown in Figure 12 and drawing AANC-4, sheet 3. A one inch diameter hole was then drilled through this assembly of plates, including the original skin, to accommodate the clevice pins. Each clevice was attached to a turnbuckle so that initial boundary conditions could be adjusted to provide a uniform hoop load. Figure 14 is a photograph of the reinforcing plates in the hoop direction with the turnbuckle restraints attached.

Figure 14: Reinforcement Plates Installed in the Hoop Direction with the Turnbuckle Restraints Attached

2.2.4 Repair of Fuselage Test Article with a Composite Doubler

Overview of Boron-Epoxy Composite Doubler - Reinforcement of the door surround structure at the load application points was a local modification of the test article. These reinforcements did not affect the strains in the area of interest around the upper, forward corner of the P3 door cut-out. The important test article modification involved the installation of a Boron-Epoxy doubler to the upper, forward corner of the P3 door surround structure. The door surround structure was tested before and after the doubler was installed to evaluate strain field improvements produced by the composite doubler. The Boron-Epoxy doubler design discussed in this analysis is shown in Figure 15. The doubler consists of 72 plies oriented in a quasi-isotropic lay-up of sequence [45,0,-45,90]. This type of lay-up is the most effective at reducing inter-laminar peel stresses. In addition, peel stresses in the adhesive bond line are reduced by tapering the edges of the doubler away from the door corner. The rectangular cutout in the doubler is required to facilitate the existing emergency door release handle.

Doubler Stiffness Requirements – To provide adequate stress reduction and crack closure, the doubler is required to have a minimum stiffness ratio (for extension and shear) of 1.2. In order to avoid an overly stiff doubler that creates stress concentration

problems in the skin adjacent to it, this ratio should not be much greater than 1.5. The stiffness ratios are defined as:

Extension:

$$R_x = (E_x t_{\text{laminate}})_{\text{BE}} / (E_x t)_{\text{Al}} \quad (11)$$

Shear:

$$R_{xy} = (G_{xy} t_{\text{laminate}})_{\text{BE}} / (E_x t)_{\text{Al}} \quad (12)$$

Classical laminated plate theory, along with Boron-Epoxy lamina properties [44], were used to arrive at the cured laminate moduli E_x and G_{xy} . The resulting R_x and R_{xy} values are as follows:

$$R_{x(-65^\circ\text{F})} = 1.55$$

$$R_{xy(-65^\circ\text{F})} = 1.53$$

$$R_{x(\text{RTA})} = 1.51$$

$$R_{xy(\text{RTA})} = 1.49$$

$$R_{x(180^\circ\text{F})} = 1.29$$

$$R_{x(180^\circ\text{F})} = 1.27$$

Thus, the L-1011 doubler provides adequate stiffness without overly restraining the adjacent skin.

Figure 15: L-1011 Door Corner Boron-Epoxy Repair Doubler

Surface Preparation and Composite Doubler Installation – The composite doubler was installed by Delta Air Lines technicians at Delta's Atlanta maintenance facility. The doubler fabrication was in accordance with the Lockheed design drawing LCC-7622-378 [42]. The Boron-Epoxy material was type 5521/4. The specifications for production of the Boron-Epoxy material are as follows: 1) SAE Aerospace Material Specification (AMS) 3865C for the fiber, and 2) AMS 3867B and AMS 3867/4A for the Boron-Epoxy pre-preg tape material. The adhesive material was Cytec AF-163 tape (0.06 PSF), and the primer was Cytec BR-127. The Boron-Epoxy composite doubler was a multi-ply lay-up as specified in References [1, 42]. The plies were cut to different lengths in both in-plane directions in order to taper the thickness of the resulting doubler edges. The parent aluminum plate on the L-1011 fuselage is 2024-T3, 0.068" thick.

The skin surface of the door surround structure was prepared using the Phosphoric Acid Containment System (PACS) equipment. The complete installation procedure is provided in Textron Specialty Materials Specification No. 200008-001 [43]. The general installation steps include: 1) aluminum surface preparation, 2) surface priming, 3) adhesive process, 4) Boron-Epoxy doubler installation and cure, and 5) doubler edge sealant, topcoat or other protective measures. Following composite doubler installation, the door surround structure was visually inspected and ultrasonically scanned to determine if there were any disbond or delamination flaws (see also section 3.5). Additional discussions on the composite doubler installation process is provided in Chapter 10.

2.3 Structural Test Loads and Procedures

2.3.1 Test Facility Description

The door surround structure test bed was subjected to a combined load environment of external vacuum (note: external vacuum was used to simulate the internal pressure which generates the primary hoop stresses in the airplane) and axial, or longitudinal, stress. A specialized structural test facility was developed to simulate maximum cabin pressure loads. It uses a computerized feedback system to maintain the proper ratio between hoop and axial loads. The differential cabin pressure was generated using a custom vacuum chamber while the axial loads were applied by hydraulic rams. The vacuum chamber design drawing is included in Appendix A. The applied biaxial tension loads approximated the stresses induced by normal flight pressurization (maximum L-1011 cabin pressure differential is $\Delta P = 8.835$ PSI). Other flight loads such as localized aerodynamic effects and fuselage torsion and shear loads were absent in this test series. However, these tests were augmented by FEM analyses which included the complete L-1011 flight load spectrum [1].

Figure 16 is a schematic of the multi-axis load test bed for the curved fuselage panels. It shows the vacuum chamber, the restraints in the hoop direction, and the load trains in the axial direction. Turnbuckles in the hoop direction were used to adjust 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 (six on each side with common hydraulic lines to assure constant axial loads) were used to supplement and or relieve axial loads during the tests.

Figure 16: Biaxial Test Facility Design Drawing Showing the Door Surround Structure Mounted in the Vacuum Chamber and the Hoop and Axial Load Application Hardware.

The door surround structure tests were conducted on a large, multi-purpose mechanical test frame. The frame, which measures 10' W X 20' L, contained a continuous grid of threaded and through-holes in order to accommodate any system of load restraints (e.g. I-beams, plates) and mechanical load application linkages. Photographs of this multi-purpose test frame with the door surround structure mounted in place are shown in Figures 17 and 18. Figure 17 is a view inside the perimeter of the test frame. It shows one side of the hoop restraints and the longitudinal whiffle trees attached to the test article. Figure 18 provides a view from the outside of the frame. The test article is in the background while the load restraint beams, hydraulic rams and data acquisition system are in the foreground. An engineering drawing of the complete test set-up on the mechanical test frame is shown in drawing AANC-2 (Appendix A). The dotted lines in

drawing AANC-2 represent the frame construction used to produce strongbacks (load restraints) in both the longitudinal and hoop directions.

The vacuum, or pressure differential loads, were generated by vacuum pumps connected to the door surround structure. Figure 19 shows the flexible hose used to connect the vacuum pumps to the test article. One of the window cut-outs was fitted and sealed with an aluminum plate. The plate contained the vacuum port, the pressure transducer, and the overpressure relief valve (see Figure 12b). The door surround structure was sealed to the vacuum chamber at stringers 12 and 17 and frame stations 829 and 923. An air tight seal between the fuselage skin and the vacuum chamber was achieved using malleable strips of caulking putty. The leak rate in the test assembly, on the order of tenths of a psi per minute, was comparable to large, industrial vacuum chambers.

Figure 17: Door Surround Structure Mounted in the Biaxial Test Facility

Figure 18: Test Set-Up Showing Assembly of Strongbacks for Load Restraints and the Data Acquisition System

Figure 19: Vacuum Line, Pressure Transducer and Relief Valve Mounted in the Fuselage Window Port

2.3.2 Instrumentation

The loads applied by the hydraulic tension rams or reacted by the turnbuckle restraints were monitored by load cells. A pressure transducer measured the pressure differential across the fuselage skin. All load cells, pressure transducers and data logging electronics were calibrated and were traceable to the National Institute of Standards and Technology. The load cells and pressure transducer were accurate to within 2% of the actual load or pressure. Drawing AANC-2 shows the unique load cell numbers in the 32 hoop (load cell numbers "LC-E-XX" or "LC-W-XX") and 12 longitudinal (load cell numbers "LC-N-YY" or "LC-S-YY") load train locations.

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 doubler on the fuselage stresses adjacent to the doubler.

The strain gage locations used to monitor the test article's response are shown in drawings AANC-5 and AANC-6. There are two strain gage deployment scenarios associated with this test series. First, prior to the composite doubler installation, a series of strain gages monitored the strain fields in the "original," unmodified door surround structure. Second, this same set of gage locations were augmented by additional strain gage installations after the composite doubler was installed.

Drawing AANC-5 shows the strain gage deployment for the "before doubler installation" test. Rosette gages were placed in far-field locations to measure strains in areas which were unaffected by the composite doubler. These exact same gages were monitored again after the doubler was installed. A series of Rosette gages were also placed on the aluminum skin within the perimeter of the composite doubler footprint (shown as dotted line in AANC-5). The locations shown correspond to high strain areas, potential crack initiation sites as indicated in Figure 9, and important load transition regions along the outer perimeter of the doubler. These gages allowed for "before-and-after" data comparisons to assess the effects of the doubler on the overall strain field. Finally, several external gage locations had matching strain gages mounted on the inside of the door surround structure at the same X,Y location. This allowed for an assessment of the bending strains at those locations.

Drawing AANC-6 shows the strain gage deployment for the "after doubler installation" test. Note the subset of gages which have the same location and reference number as the "before doubler installation" test. In this test, the strain gages inside the footprint of the composite doubler were installed after, and on top of, the doubler. They monitored the strain field on the Boron-Epoxy material. Additional biaxial and Rosette gages were added to monitor strain levels throughout the doubler and in areas adjacent to the doubler. Sections 3.1 and 3.2 provide additional information on the "before doubler" and "after doubler" instrumentation.

All strain gage locations indicated in AANC-5 and AANC-6 had a location tolerance of 0.125". Strain readings had an accuracy of ± 5 microstrain. The strain gages were installed using Micro Measurements Inc. M-bond 200 adhesive which is tolerant of all test article materials and environmental test conditions. A Micro Measurements Co. System 4000 data acquisition system was used to monitor pressure, loads and strains in the test article. The fuselage section was tested at Room Temperature Ambient (RTA) conditions.

Door Surround Structure Test Before Composite Doubler Installation - A static, biaxial load test was performed on the door surround structure test article in its original condition as cut from the retired ANA airplane. This test is also referred to as the "before composite doubler installation" test. The test used the custom facility discussed in Section 2.3.1, the instrumentation described in Section 2.3.2, the test procedure listed in Section 2.4, and the nondestructive inspection procedures described in Section 2.5. After completing this test, the door surround structure was shipped to the Delta Air Lines' Atlanta maintenance facility where Delta mechanics installed the Boron-Epoxy doubler as per References [1] and [42-43]. The door surround structure test article was then shipped back to the Sandia National Labs' AANC for additional testing.

Door Surround Structure Test After Composite Doubler Installation - A static, biaxial load test was performed on the door surround structure after the Boron-Epoxy doubler was installed on the test article. The test used the custom facility discussed in Section 2.3.1, the instrumentation described in Section 2.3.2, the test procedure listed in Section 2.4, and the nondestructive inspection procedures described in Section 2.5. Final analysis of the strain gage data is presented in Section 3.0. Comparisons are made between: 1) test data before and after the doubler was installed, and 2) test data and calculations from the finite element models described in Ref. [1].

2.4 Pressure-Tension Test Procedures

1. A small pre-load was applied in both the hoop (generated by the vacuum) and the axial (generated by the hydraulic rams) directions. A 1 PSI pre-load was sufficient to eliminate any slack in the load fixtures and to produce proper contact between the door surround structure and the vacuum chamber. Dry-run tests showed that this pre-load was necessary to achieve a uniform boundary condition in the hoop turnbuckles. The longitudinal loads were then set to produce the 2:1 ratio between the hoop and axial stresses.
2. The entire stress field on and around the composite doublers was monitored using the series of biaxial and Rosette strain gages described in Section 2.3.2. Load, strain, and pressure data was acquired at 0.5 PSI intervals up to the maximum pressure differential test goal of 10.0 PSI. The L-1011's maximum operating pressure is 8.835 PSI.
3. Vacuum pumps were used to produce the pressure differential across the door surround structure. After pre-loading the door surround structure (vacuum and axial

load), the strain gage bridges were balanced to produce a zero strain output signal. This data was used as the biaxial test starting point (Longitudinal Load = Hoop Load = 0 lbs.). The pressure differential and longitudinal tension loads were increased as per Table 1. In accordance with thin-walled cylinder theory, a ratio of 2:1 was maintained between the hoop and axial stresses [Hoop Stress $\sigma_H = Pr/t$ and Longitudinal Stress $\sigma_L = Pr/2t$; note that the stresses had a 2:1 ratio, the loads did not]. The hoop- and longitudinal-direction load cells provided the real-time feedback to assure that the 2:1 stress ratio was achieved.

4. The door surround structure was inspected with visual, ultrasonic, and eddy current NDI techniques before and after the structural tests (see also Section 2.5). The NDI experience gained in this test series was used to produce final step-by-step procedures for the on-aircraft inspections.
5. The door surround structure was subjected to 6 test cycles where one cycle is comprised of steps 1-4 above.

The stresses listed in Table 1 represent approximate calculations for uniform strain fields using equations for ideal, pressurized right circular cylinders. They do not account for the presence of any substructure elements. They were used to provide guidance and check points for the structural tests but were not intended to validate any strain gage readings during the tests.

2.5 Nondestructive Inspections

The NDI techniques deployed in this effort were capable of detecting disbonds, delaminations, and porosity in the composite doubler (ultrasonics), as well as cracks in the parent aluminum material [18, 27, 45-46, 51]. The door surround structure was inspected with visual (optical magnification), ultrasonic (UT), X-ray and eddy current (EC) NDI techniques before and after the composite doubler installation. Before the doubler was installed, a drawing was made of the area covered by the doubler footprint. It included the location of all rivets or surface structures which may provide a source of false flaw indications. This drawing was then used as a "template" map to aid the inspections performed after the doubler was installed.

After the doubler was installed, disbond and delamination inspections were performed using the Sandia Labs AANC ultrasonics inspection procedure for bonded composite doublers, AANC-UT-Comp-5521/4-002 [45]. This procedure is required by reference [42] for the L-1011 application. Inspections for cracks in the parent material were carried out using an Eddy Current (EC) procedure applied to both the composite and non-composite doubler side of the test article. Eddy current inspections were performed as per the Ref. [46] procedure AANC-EC-Comp-5521/4-003. X-ray inspections, currently in use in the door corner region, were conducted using the procedure listed in the L-1011 Nondestructive Testing Manual [51]. These NDI tests included an assessment of the resolution of X-ray inspections through composite doublers. Also, crack detection was pursued through the use of an optical inspection device. The eyepiece produced an image

magnification of seven and had an embedded set of crosshairs (tic marks) which provided a resolution of 0.005". The optical inspection technique is not part of standard aviation specifications. It was simply used to augment the results from the X-ray and EC inspections.

The documentation and validation of inspection procedures, as well as the appropriate approvals by the FAA, is described in another Sandia Labs report under the auspices of this same ACO project number [18].

Pressure Differential	Hoop Stress [†]	Hoop Load	Longitudinal Stress	Longitudinal Load *
0 psi	0 psi	(0 lbs)	0 psi	(0 lbs.)
0.5	864	(6,739)	432	(4,436)
1.0	1728	(13,484)	864	(8,872)
1.5	2592	(20,225)	1296	(13,307)
1.77 Δ	864	(23,856)	1529	(15,700)
2.0	3456	(26,957)	1728	(17,743)
2.5	4320	(33,696)	2160	(22,179)
3.0	5184	(40,435)	2592	(26,615)
3.5	6048	(47,174)	3024	(31,050)
4.0	6912	(53,914)	3456	(35,486)
4.5	7776	(60,653)	3888	(39,922)
5.0	8640	(67,392)	4320	(44,358)
5.5	9504	(74,131)	4752	(48,794)
6.0	10,368	(80,870)	5184	(53,229)
6.5	11,232	(87,610)	5616	(57,665)
7.0	12,096	(94,349)	6048	(62,100)
7.5	12,960	(101,088)	6480	(66,537)
8.0	13,824	(107,827)	6912	(70,972)
8.5	14,688	(114,566)	7344	(75,401)

† Hoop Stress $\sigma_H = Pr/t = F_H/A_H$; L-1011 Fuselage Radius $r = 117.5''$; Skin Thickness $t = 0.068''$; P = test pressure; F_H = hoop load; $A_H = t \times W$; width of test article $W = 114.75''$

* Longitudinal Stress was controlled to achieve the 2:1 ratio between longitudinal and hoop stress; $\sigma_L = F_L/A_L$ (for test set-up); F_L = longitudinal load; $A_L = t \times L$; the final door surround structure arc length was $L = 151''$; calculated stresses were used for an initial check of test set-up

Δ Pressure level corresponds to 20% of maximum L-1011 operating pressure of 8.85 psi. Data was acquired during loading and unloading of the structure to determine if there was any permanent shift in the strain readings

Table 1: Correlation of L-1011 Fuselage Cabin Pressure With Resulting Hoop and Axial Stresses

This Page Left Intentionally Blank

3.0 Fuselage Door Surround Structure Test Results

The data from the biaxial tests on the fuselage door surround structure will be presented in the following distinct sections: 1) structural tests before the composite doubler installation, 2) structural tests after the composite doubler installation, 3) comparison of results from the "before" and "after" tests, and 4) comparison between experimental and analytical results. Nondestructive inspections, performed to assess any flaw growth induced by the structural tests, will also be discussed. The test results are presented in terms which will allow the data to support a wide array of composite doubler applications. This full scale fuselage test series validates composite doubler technology using realistic structures and loads. The tests show that a properly designed and installed composite doubler is able to enhance fatigue life, transfer load away from damaged structure, and avoid the introduction of new stress risers (i.e. eliminate global reduction in the fatigue life of the structure).

3.1 Structural Tests Before Composite Doubler Installation

Prior to installing the composite doubler, a biaxial test was conducted on the door surround structure test article using the custom test facility and procedures described in Section 2.0. Figure 20 and drawing AANC-5 show the gage layout that was used to monitor the strain field throughout the upper forward corner of the door surround structure. The ten Rosette strain gages installed in the "before doubler" test are labeled R-1, R-2, R-4, R-5, R-6, R-7, R-8, R-9, R-10, and R-11 in Figure 20. For reference purposes, the doubler footprint is shown, however, all gages are mounted on the parent aluminum structure. The stress and strain values presented in this section provide a basis of comparison for the results obtained after the doubler was installed. Common gage locations and numbers were used in both the before and after doubler tests in order to make the comparisons straightforward.

The test loads were intended to simulate the maximum cabin pressure loads. In practice, the loads monitored during the test were slightly less than those needed to produce the maximum operation stresses (hoop and axial). However, extrapolations from the test environment to actual flight loads, including fuselage bending and aerodynamic loads, were achieved through comparisons with a validated, finite element model (ref. Section 3.4).

Table B-1 in Appendix B relates the pressure differential applied during the test (i.e. vacuum outside the test article vs. ambient pressure on the inside) to the resulting hoop and axial loads generated. Next to the pressure column, the loads monitored by each of the individual hoop and axial load cells are listed. Drawing AANC-2 shows the unique load cell numbers in the 32 hoop (load cell numbers "LC-E-XX" or "LC-W-XX") and 12 longitudinal (load cell numbers "LC-N-YY" or LC-S-YY") load train locations.

Table B-1 also lists the total hoop and axial loads applied uniformly along the circumferential and longitudinal dimensions, respectively. To summarize this data, Figure 21 provides a graph of the total hoop and axial loads applied to the test article. Opposite sides of the test article are summed up and plotted separately in order to demonstrate the consistency of boundary conditions provided during the test. Figures 22 and 23 show the scatter of individual loads monitored in the axial and hoop directions. These plots demonstrate the facility's ability to apply uniform loads across the area of interest and to maintain the desired 2:1 ratio between the hoop and axial stresses.

Figure 20: Strain Gage Layout on Fuselage Test Article Before Doubler Installation

Figure 21: Total Hoop and Axial Loads Applied to the Test Article Before the Doubler was Installed

Figure 22: Axial Load Scatter in Individual Load Cells (North and South); Before Doubler Test

Figure 23: Hoop Load Scatter in Individual Load Cells in Critical Center Load Area (East and West); Before Doubler Test

3.1.1 Strain Field Analysis

The complete set of strain gage readings from the door surround structure without composite doubler reinforcing is presented in Table B-2 (see Appendix B). Figures 24 - 25 show plots of the hoop strains, Figures 26 - 27 show plots of the axial strains, and Figures 28 - 29 show plots of the shear strains measured in the test article. The maximum hoop strains were found at the following locations: 1) the crack reference location (as sited in the Figure 9 drawing) along the vertical portion of the door cut-out (Ch. 38 = 2015 $\mu\epsilon$), 2) the upper (horizontal) leg of the doubler (Ch. 14 = 892 $\mu\epsilon$), and 3) the door corner radius (Ch. 32 = 1080 $\mu\epsilon$). The maximum axial strains were found at the following locations: 1) the crack reference location along the horizontal portion of the door cut-out (Ch. 27 = 792 $\mu\epsilon$), 2) the door corner radius (Ch. 33 = 757 $\mu\epsilon$), and 3) the crack reference location above the door handle cut-out (Ch. 48 = 712 $\mu\epsilon$). Finally, large shear strains (45° strain gages) were monitored at: 1) the door corner radius (Ch. 31 = 1275 $\mu\epsilon$), and 2) along the upper (horizontal) leg of the doubler (Ch. 13 = 1068 $\mu\epsilon$). The strain responses were primarily linear. However, some slight non-linearities, most notably at the lower strain levels, were observed (e.g. Ch 35 in Fig. 24 and Ch. 40 in Fig. 29). The strain magnitudes in these regions indicated that the material was not yielding. Rather, the non-linearities were probably caused by the global response of the multi-element fuselage structure where load shedding occurred into the substructure elements as the test loads were increased.

Figure 24: Hoop Strains Along Upper (Horizontal) Portion of Door Corner

Figure 25: Hoop Strains Along Vertical Portion of Door Corner

Figure 26: Axial Strains Along Upper (Horizontal) Portion of Door Corner

Figure 27: Axial Strains Along Vertical Portion of Door Corner

Figure 28: Shear Strains Along Upper (Horizontal) Portion of Door Corner

Figure 29: Shear Strains Along Vertical Portion of Door Corner

3.1.2 Stress Field Analysis

Principal Stresses - The principal stresses were determined from strain data obtained from the Rosette strain gages. In these three-element rectangular rosettes, gages are located at 0° , 45° , and 90° orientations corresponding to strains ϵ_1 , ϵ_2 , and ϵ_3 , as shown in Figure 30. From these individual strains, the maximum and minimum principal stresses can be calculated using the following equations:

$$\sigma_{\max} = \frac{E(\varepsilon_1 + \varepsilon_3)}{2(1 - \nu)} + \frac{E}{\sqrt{2}(1 + \nu)} [(\varepsilon_1 - \varepsilon_2)^2 + (\varepsilon_2 - \varepsilon_3)^2]^{1/2} \quad (13)$$

$$\sigma_{\min} = \frac{E(\varepsilon_1 + \varepsilon_3)}{2(1 - \nu)} - \frac{E}{\sqrt{2}(1 + \nu)} [(\varepsilon_1 - \varepsilon_2)^2 + (\varepsilon_2 - \varepsilon_3)^2]^{1/2} \quad (14)$$

Figure 30: Strain Components Measured by Rectangular Strain Gage Rosette

The maximum principal stresses for Rosettes R-1, R-2, and R-4 through R-11 are listed in Table B-3 and plotted in Figure 31. The largest stress in the door surround structure occurred at R-8 (crack reference location along the vertical portion of the door cut-out) and was 22.4 KSI. Stresses at the door corner radius (R-6) reached 15.3 KSI. Far field stresses ranged from 9.7 KSI (R-11) to 11.3 KSI (R-1). Thus, the greatest stress magnification factor around the door cut-out was found to be $22.4/9.7 = 2.3$.

Principal Stress Directions - The orientation of the maximum principal stress $\sigma_{\max}$ is defined by the angle θ (see Figure 30) which is determined from:

$$\tan 2\theta = \frac{2\varepsilon_2 - \varepsilon_1 - \varepsilon_3}{\varepsilon_1 - \varepsilon_3} \quad (15)$$

Figure 32 shows the direction of the principal stress for the ten Rosette gages. A principal angle of 0° would correspond to pure axial stress along the X axis while a principal angle of 90° would correspond to pure hoop stress. The regions which were

dominated by axial strains, as exhibited by principal stress directions of 0° to 20° , were: 1) R-4 and R-5 (along upper, horizontal edge of door cut-out), and 2) R-11 (far field forward of door corner area). The gages that were dominated by hoop strains, with principal stress directions in the 70° to 90° range, were: 1) R-1 (far field above door corner area), and 2) R-8 (along vertical, forward edge of door cut-out). The remaining Rosette gages measured strains with significant magnitudes in multiple directions. Their principal stress directions were between 20° and 70° .

Figure 31: Principal Stresses in Door Surround Structure Before Doubler Installation

Figure 32: Principal Stress Directions in Door Surround Structure Before Doubler Installation

Membrane Stresses - The strain data collected from the hoop and axial gages were used to calculate the corresponding membrane stresses using the following equations:

$$\sigma_h = \frac{E}{1-\nu^2} (\epsilon_h + \nu \epsilon_a) \quad (16)$$

$$\sigma_a = \frac{E}{1-\nu^2} (\epsilon_a + \nu \epsilon_h) \quad (17)$$

where E is the modulus of elasticity, ν is Poisson's ratio, σ_h is the hoop stress in the skin, σ_a is the longitudinal stress in the skin, ϵ_h is the hoop strain, and ϵ_a is the axial or longitudinal strain. From Mil-Handbook 5, the modulus of elasticity and Poisson's ratio for 2024-T3 aluminum are: $E = 10.5$ msi and $\nu = 0.33$, respectively. Table 2 lists the membrane stresses induced by the maximum test loads. The largest membrane stress is 21.5 KSI (R-8 hoop stress). Most of the stresses are less than 10 KSI which is well below the 42 KSI nominal yield value for 2024-T3 material.

Gage Identity	Axial Stress (psi)	Hoop Stress (psi)	Principal Stress (psi)	Principal Stress Dir. (degrees)	Location on Test Specimen
R.1, Ch. 1,2 & 3	1547	11210	11304	81	Far Field Above Door Cut-out
R.2, Ch. 12,13 & 14	2455	10176	12701	63	Outside Skin at Door Corner Radius
R.4, Ch. 24,25 & 26	8974	977	9574	16	Above Door Cut-out (Outside Skin)
R.5, Ch. 27,28 & 29	7369	-2871	7448	4	Above Door Cut-out (Inside Skin)
R.6, Ch. 30,31 & 32	3222	12403	15299	63	Outside Skin at Door Corner Radius
R.7, Ch. 33,32 & 35	10537	7845	15066	39	Inside Skin at Door Corner Radius
R.8, Ch. 36,37 & 38	966	21476	22366	77	Beside Door Cut-out (Outside Skin)
R.9, Ch. 39,40 & 41	-478	-1544	1330	38	Beside Door Cut-out (Inside Skin)
R.10, Ch. 48,49 & 50	10007	7670	13246	38	Above Door Handle Cut-out
R.11, Ch. 53,54 & 55	9988	5480	9702	4	Far Field Forward of Door Cut-out

Table 2: Membrane Stresses at Peak Loads for L-1011 Door Surround Structure Before Doubler Installation

The stress values in Table 2 were calculated using the measured strain values at peak loads along with the following material properties.

$$\begin{aligned}
 E_{\text{alum}} &= 10.5 \times 10^6 \text{ psi,} & \nu &= 0.33 \\
 \text{Pressure Differential} &= 10.0 \text{ psi} \\
 \text{Hoop load} &= 60,600 \text{ lbs} \\
 \text{Axial Load} &= 50,000 \text{ lbs}
 \end{aligned}$$

3.1.3 Bending Effects

The primary reason for gaging common locations on the inside and outside of the test article was to obtain information regarding out-of-plane bending in critical regions. The FEM of the door surround structure and other subelement structural tests (reference Part II of this document) both highlighted the existence of stress risers at the corner of the door cut-out. Therefore, external Rosettes R-4, R-6, and R-8 were matched by internal Rosettes R-5, R-7, and R-9. Bending strains were calculated using the individual strain gage readings and the following equation:

$$\varepsilon_b(\psi) = \frac{1}{2}(\varepsilon_e + \nu\varepsilon_i)(\psi) \quad (18)$$

where ε_b is the bending strain, ε_e is the external strain, ε_i is the internal strain, and ψ denotes the bending direction (hoop, axial, or shear).

A comparison of strain gage elements from the mating internal and external locations revealed the following levels of bending strains:

1. L-1011 reference crack location along the horizontal, upper edge of the door cut-out (R-4 vs. R-5) bending strains: $\varepsilon_b(h) = 158 \mu\varepsilon$, $\varepsilon_b(a) = 16 \mu\varepsilon$, $\varepsilon_b(s) = 290 \mu\varepsilon$.
2. Door corner radius (R-6 vs. R-7) bending strains: $\varepsilon_b(h) = 332 \mu\varepsilon$, $\varepsilon_b(a) = -420 \mu\varepsilon$, $\varepsilon_b(s) = 724 \mu\varepsilon$.
3. L-1011 reference crack location along the vertical, forward edge of the door cut-out (R-8 vs. R-9) bending strains: $\varepsilon_b(h) = 1074 \mu\varepsilon$, $\varepsilon_b(a) = -291 \mu\varepsilon$, $\varepsilon_b(s) = 215 \mu\varepsilon$.

These bending strains account for as much as 50% of the total strain in the structures being monitored. Figure 31 also shows the significant amount of bending that occurs across the fuselage skin at R-8/R-9. While the principal stress on the inside skin (R-9) was only 1.3 KSI, the principal stress on the outside skin (R-8) was 22.4 KSI.

3.2 Structural Tests After Composite Doubler Installation

Upon completion of the structural tests on the fuselage section in its original condition, the test article was shipped to the Delta Air Lines Atlanta maintenance facility. Personnel from Delta's composite shop installed the Lockheed-designed doubler on the door surround structure as per references [42] and [43]. The test article was then shipped back to Sandia Labs and re-tested. Figure 33 shows the doubler being positioned for cure on the door surround structure and an ultrasonic inspection on the installed doubler. The photos in Figure 34 show an overall view of the doubler on the fuselage test article and the strain gages installed on the composite doubler and surrounding aluminum skin.

Figure 35 and drawing AANC-6 show the 74 channels of strain that were used to monitor: 1) the strain field throughout the composite laminate and aluminum parent material, and 2) the load transfer into the composite doubler. The stress, strain, and load transfer values presented in this section quantify the doubler's performance on the complete aircraft structure under realistic loads. Figure 35 also shows the gaged area divided into six different zones. These zones are used in the following sections to present the test data.

A. Composite Doubler Being Positioned for Cure

B. Ultrasonic Inspection of Bonded Doubler

Figure 33: Composite Doubler Installation and NDI on L-1011 Door Corner

A. Fuselage Test Article with Composite Doubler Installed; Note Reinforcing Plates Around Perimeter of Fuselage Section for Load Application

B. Close-Up View of Doubler with Strain Gages Installed

Figure 34: Door Surround Structure Test Article With Strain Gages Mounted

Zone	Description
1	Far Field Strains
2	Upper (horizontal) portion of door
3	Upper (horizontal) portion of doubler
4	Around Corner of Door Cut-Out
5	Vertical Portion of Door Cut-Out
6	Forward Portion of Doubler

Figure 35: Strain Gage Layout for Fuselage Door Structure Tests After Doubler Installation

Table B-4 lists the test load data: pressure differential vs. hoop loads vs. axial loads. Drawing AANC-2 shows the unique load cell numbers in the 32 hoop (load cell numbers "LC-E-XX" or "LC-W-XX") and 12 longitudinal (load cell numbers "LC-N-YY" or "LC-S-YY") load train locations. The total hoop load (sum of 32 individual hoop load cells) and total axial load (sum of 12 axial load cells) applied uniformly along the circumferential and longitudinal dimensions are also listed in Table B-4. Figure 36 provides a graph of the total hoop and axial loads applied to the test article. Figures 37 and 38 show the scatter of individual loads monitored in the axial and hoop directions. These plots demonstrate the facility's ability to apply uniform loads across the area of interest (provide consistent boundary conditions) and to maintain the desired 2:1 ratio between the hoop and axial stresses.

Figure 36: Total Hoop and Axial Loads Applied to the Test Article After the Doubler was Installed

**Figure 37: Axial Load Scatter in Individual Load Cells (North and South);
After Doubler Test**

**Figure 38: Hoop Load Scatter in Individual Load Cells in Critical, Center Load
Area (East and West); After Doubler Test**

3.2.1 Strain Field Analysis

Table B-5 lists the complete set of strain gage data recorded during the test of the door surround structure with the composite doubler installed. The hoop strains, for both the composite and aluminum skins in Zones 1-6, are plotted in Figures 39-44. The largest hoop strains in the vicinity of the door cut-out were monitored at the following locations: 1) Z-3 aluminum just above the doubler's horizontal leg (Ch. 9 = 1141 $\mu\epsilon$), 2) Z-4 boron at crack reference location forward of door cut-out (Ch. 38 = 1618 $\mu\epsilon$), 3) Z-4 boron around the door corner radius (Ch. 32 = 697 $\mu\epsilon$), 4) Z-4 aluminum at crack reference location forward of door cut-out (Ch. 72 = 1439 $\mu\epsilon$), 5) Z-5 aluminum just below the vertical leg of the doubler (Ch. 66 = 1285 $\mu\epsilon$), and 6) Z-6 aluminum at crack reference location around the emergency release door handle cut-out (Ch. 74 = 474 $\mu\epsilon$). All strains in Zone 2, along the horizontal portion of the door cut-out, were compressive in the hoop direction; however, their magnitudes were all less than -300 $\mu\epsilon$. Except for the strain riser area around the door handle cut-out, Zone 6 hoop strains are all relatively small (400 $\mu\epsilon$ or less).

The axial strains measured during the test are plotted in Figures 45-50. Maximum strains in the axial direction were found at the following locations: 1.) Z-2 aluminum just aft of the doubler's horizontal leg (Ch. 18 = 853 $\mu\epsilon$), 2) Z-4 aluminum around the door corner radius (Ch. 69 = 957 $\mu\epsilon$), and 3) Z-4 boron and aluminum at crack reference location above the door cut-out (Ch. 24 = Ch. 27 = 509 $\mu\epsilon$). The axial strains in the upper portion of the doubler (Zone 3) were the smallest in magnitude: 200 $\mu\epsilon$ or less. The axial strains in Zone 5 (portion of vertical leg next to door cut-out) were all compressive with magnitudes less than 500 $\mu\epsilon$.

The strain responses were primarily linear. Some non-linearities were observed but, as noted in the "before doubler" results, they can be attributed to settling and load shifting in the multi-layered structure. Finally, it should be noted that the peak strain areas listed above agree with those found in the "before doubler" test. Their magnitudes, however, are less. *Direct before and after comparisons show that the strain field around the door cut-out was not shifted by the doubler. Rather, the strain levels in all areas were uniformly reduced. This data substantiates the doubler design (footprint and stiffness) since it demonstrates that the doubler does not create new stress risers due to excessive doubler stiffness in any one area.*

Figure 39: Zone 1 (Far Field) Hoop Strains After Doubler Installation

Figure 40: Zone 2 (Upper Door Cut-Out) Hoop Strains After Doubler Installation

Figure 41: Zone 3 (Upper Edge of Doubler) Hoop Strains After Doubler Installation

Figure 42: Zone 4 (Door Cut-Out Corner) Hoop Strains After Doubler Installation

Figure 43: Zone 5 (Vertical Door Cut-Out) Hoop Strains After Doubler Installation

Figure 44: Zone 6 (Forward Portion of Doubler) Hoop Strains After Doubler Installation

Figure 45: Zone 1 (Far Field) Axial Strains After Doubler Installation

Figure 46: Zone 2 (Upper Door Cut-Out) Axial Strains After Doubler Installation

Figure 47: Zone 3 (Upper Edge of Doubler) Axial Strains After Doubler Installation

Figure 48: Zone 4 (Door Cut-Out Corner) Axial Strains After Doubler Installation

Figure 49: Zone 5 (Vertical Door Cut-Out) Axial Strains After Doubler Installation

Figure 50: Zone 6 (Forward Portion of Doubler) Axial Strains After Doubler Installation

3.2.2 Stress Field Analysis

Principal Stresses - Equations (13) and (14) were used to calculate the principal stresses at each of the 12 Rosette gage locations. Table B-6 lists the set of maximum and minimum stresses/strains and their direction while Figures 51-52 plot the maximum principal stresses for R-1 through R-12. 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 3. *It can be seen that the stresses were reduced by as much as 53% through the application of the reinforcing doubler.*

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 3: Principal Stress Reductions Observed Within the Doubler Footprint

Figure 51: Principal Stresses in Region of Doubler Horizontal Leg

Figure 52: Principal Stresses in Region of Doubler Vertical Leg

Principal Stress Directions - The orientation of each maximum principal stress was calculated using equation (15) and the individual strain readings from each of the twelve Rosette gages. Figures 53 and 54 show the principal stress directions for R-1 through R-12. As in the "before doubler" test, a principal angle of 0° would correspond to pure axial stress along the X axis while a principal angle of 90° would correspond to pure hoop stress. The regions which were dominated by axial strains, as exhibited by principal stress directions of 0° to 20° , were: 1) R-3, R-4 and R-5 (along upper, horizontal edge of door cut-out), and 2) R-11 (far field forward of subject area). The regions that were dominated by hoop strains, with principal stress directions in the 70° to 90° range, were: 1) R-1 (far field above subject area), 2) R-2 (upper edge of doubler), 3) R-8 and R-9 (along vertical, forward edge of door cut-out), and 4) R-12 (forward edge of doubler). The remaining Rosette gages, R-6, R-7, and R-10, were all located around the corners of cut-outs in the skin. As a result, they measured strains with significant magnitudes in all three directions. Their principal stress directions were between 20° and 70° . These findings reaffirm the expected results for the strain field around a reinforced plate with a center cut-out. The area above the door cut-out should resist axial loads while shedding hoop loads while the area along the vertical face of the cut-out should resist hoop loads while shedding the axial loads.

Figure 53: Principal Stress Directions in Region of Doubler Horizontal Leg

Figure 54: Principal Stress Directions in Region of Doubler Vertical Leg

Membrane Stresses - In areas with biaxial strain gages, the membrane stresses were calculated using the hoop and axial strain readings along with equations (16) and (17). Similarly, the axial and hoop components on the Rosette gages were used to compute the axial and hoop membrane stresses. All material property values remain the same as per section 3.1.2. with the addition of the modulus of elasticity for Boron-Epoxy. For gages mounted on the composite laminate, reference [44] information and the ply lay-up for this doubler generates a modulus $E = 11.87$ msi. Table 4 lists the maximum stresses induced by the test loads during the "after doubler" test. Only one Boron-Epoxy stress exceeded 10 KSI: R-8 hoop stress at the crack reference location along the vertical portion of the door cut-out ($\sigma_h = 19.3$ KSI). Membrane stresses in the aluminum exceeded 10 KSI at the following locations: 1) R-11 axial stress (far field forward of subject area; $\sigma_a = 10.7$ KSI), 2) B-3 hoop stress (far field aft of subject area; $\sigma_h = 12.6$ KSI), 3) B-15 hoop stress (near lower edge of doubler's vertical leg; $\sigma_h = 14.4$ KSI), 4) B-17 axial stress (along upper, horizontal portion of door cut-out; $\sigma_a = 10.2$ KSI), and 5) B-18 hoop stress (crack reference location along the vertical portion of the door cut-out; $\sigma_h = 15.3$ KSI).

To place these stress values in perspective, comparisons can be made to the subsize door corner test which loaded a similar doubler to failure [see Section II of this document: "Subsize Door Corner Test (C-Section Test)"]. In this ultimate failure test, the stress in the composite doubler reached 92.3 KSI before failure occurred. Thus, the stresses required to fail this doubler were found to be 4.8 times greater than the stresses measured during the simulated cabin pressure tests presented here.

Also in the C-section test, the parent aluminum material beneath the doubler experienced a maximum stress level of 86 KSI. This means that the composite-reinforced aluminum plate was able to withstand stresses in excess of the Mil Handbook ultimate tensile stress value listed for the parent aluminum material: 83 KSI for 7075-T6 used in test article. In addition, the aluminum stresses measured at doubler failure were 5.6 times greater than the stresses experienced during the simulated cabin pressure tests.

Gage Identity	Axial Stress (psi)	Hoop Stress (psi)	Principal Stress (psi)	Princ. Stress Dir. (deg)	Location on Test Specimen
R.1, Ch. 1,2 & 3	651	9896	10056	80	Far Field Away from Door Cut-out
R.2, Ch. 12,13 & 14	28	6761	6879	82	Outside Skin at Door Corner Radius
R.3, Ch. 15,16 & 17	5282	-1239	5510	-11	Far Field Above Door Cut-out
R.4, Ch. 24,25 & 26	5989	-27	6205	11	Above Door Cut-out (Outside Skin)
R.5, Ch. 27,28 & 29	4858	-1473	4911	-5	Above Door Cut-out (Inside Skin)
R.6, Ch. 30,31 & 32	3571	9392	12881	58	Outside Skin at Door Corner Radius
R.7, Ch. 33,34 & 35	6220	6756	10259	-47	Inside Skin at Door Corner Radius
R.8, Ch. 36,37 & 38	898	19346	19363	86	Beside Door Cut-out (Outside Skin)
R.9, Ch. 39,40 & 41	477	5867	5932	-88	Beside Door Cut-out (Inside Skin)
R.10, Ch. 48,49 & 50	3705	1800	6221	-37	Above Door Handle Cut-out
R.11, Ch. 53,54 & 55	10700	6335	10549	11	Far Field Forward of Door Cut-out
R.12, Ch. 58,59 & 60	-3044	1989	2086	-88	FWD Portion of Doubler Vert. Leg
B.1, Ch. 4 & 5	4236	4086	n.a.	n.a.	Far Field FWD/Above Doubler
B.2, Ch. 6 & 7	4304	8046	n.a.	n.a.	Alum. Above Horiz. Leg
B.3, Ch. 8 & 9	1880	12601	n.a.	n.a.	Alum. Above Horiz. Leg
B.4, Ch. 10 & 11	3597	6873	n.a.	n.a.	Upper Portion of Doubler Horiz. Leg
B.5, Ch. 18 & 19	9192	713	n.a.	n.a.	Alum. Just Aft of Horiz. Leg
B.6, Ch. 20 & 21	4821	-1721	n.a.	n.a.	Aft Portion of Horiz. Leg
B.7, Ch. 22 & 23	3965	110	n.a.	n.a.	Lower Portion of Doubler Horiz. Leg
B.8, Ch. 42 & 43	2033	1917	n.a.	n.a.	Center of Doubler
B.9, Ch. 44 & 45	446	-482	n.a.	n.a.	FWD Portion of Doubler Vert. Leg
B.10, Ch. 46 & 47	1557	3286	n.a.	n.a.	Alum. FWD of Doubler Vert. Leg
B.11, Ch. 51 & 52	-1767	2883	n.a.	n.a.	FWD Portion of Doubler Vert. Leg
B.12, Ch. 56 & 57	157	7922	n.a.	n.a.	Aft Portion of Doubler Near Door
B.13, Ch. 61 & 62	3953	6061	n.a.	n.a.	Alum. FWD of Doubler Vert. Leg
B.14, Ch. 63 & 64	-1558	4056	n.a.	n.a.	Lower Portion of Doubler Vert. Leg
B.15, Ch. 65 & 66	2628	14360	n.a.	n.a.	Alum. Below Doubler Vert. Leg
B.16, Ch. 67 & 68	5992	2565	n.a.	n.a.	Far Field FWD/Below Doubler
B.17, Ch. 69 & 70	10223	528	n.a.	n.a.	Alum. at Door Corner Radius
B.18, Ch. 71 & 72	517	15280	n.a.	n.a.	Alum. at Door Corner Radius
B.19, Ch. 73 & 74	6003	6958	n.a.	n.a.	Alum. at Door Handle Cut-Out

$$E_{ALUM} = 10.5 \times 10^6 \text{ psi}$$

$$\text{Pressure} = 10.0 \text{ psi}$$

$$E_{BE} = 11.87 \times 10^6 \text{ psi}$$

$$\text{Axial Load} = 40,300 \text{ lbs}$$

$$\nu = 0.33$$

$$\text{Hoop Load} = 68,400 \text{ lbs}$$

**Table 4: Membrane Stresses at Peak Loads for Fuselage Structure
After Doubler Installation**

3.2.3 Bending Effects

As in the "before doubler" test, bending strains were calculated using the individual strain gage readings and equation (18). In the "after doubler" test, external Rosette gages R-4, R-6, and R-8 were matched by internal Rosette gages R-5, R-7, and R-9. Comparisons between common strain gage directions revealed the following bending strains in the hoop (h), axial (a), and shear/45° (s) directions:

1. L-1011 reference crack location along the horizontal, upper edge of the door cut-out (R-4 vs. R-5) bending strains: $\epsilon_b(h) = 62 \mu\epsilon$, $\epsilon_b(a) = 0 \mu\epsilon$, $\epsilon_b(s) = 135 \mu\epsilon$.
2. Door corner radius (R-6 vs. R-7) bending strains: $\epsilon_b(h) = 125 \mu\epsilon$, $\epsilon_b(a) = -170 \mu\epsilon$, $\epsilon_b(s) = 552 \mu\epsilon$.
3. L-1011 reference crack location along the vertical, forward edge of the door cut-out (R-8 vs. R-9) bending strains: $\epsilon_b(h) = 534 \mu\epsilon$, $\epsilon_b(a) = -163 \mu\epsilon$, $\epsilon_b(s) = 262 \mu\epsilon$.

These bending strains account for 30% to 50% of the total strain in the structures being monitored. Figure 52 also shows the significant amount of bending that occurs across the fuselage skin at R-8/R-9. While the principal stress on the inside skin (R-9) was only 5.9 KSI, the principal stress on the outside skin (R-8) was 19.3 KSI. These results are similar to those obtained during the "before doubler" tests, however, the magnitudes of the bending strains are lower. This shows that the doubler reduced the stress risers in the door corner area.

3.2.4 Load Transfer

Figure 55 shows a series of strain gage plots which accent the differences in strain between the composite doubler and the aluminum skin nearby. Plots of percent load transfer were obtained by calculating the ratio between doubler strains and strains in adjacent portions of the parent aluminum skin. Figures 56 - 58 show the resulting load transfer plots for various doubler and aluminum reference channels $\{\epsilon_{\text{doubler}} / \epsilon_{\text{alum(ref)}}\}$ throughout the doubler data zones (Z-2 through Z-6). The comparisons focus on strain levels around the outer perimeter of the doubler. References [1], [17] and [48-49] determined that this is the critical region of the doubler and that most of the load transfer occurs along the outer, tapered edges of the doubler. As a result, the allowable flaw size - which drives the nondestructive inspection requirements - is smallest around the perimeter of the doubler.

The curves in Figs. 56 - 58 indicate that 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. Several outliers were found with minimums of 30% and a high of 80% along the upper portion of the doubler. These load transfer levels agree with the results obtained in references [1], [17] and [48-49].

Figure 55: Comparison of Composite Strains and Nearby Aluminum Strains

Figure 56: Axial and Hoop Load Transfer in Horizontal Leg of Doubler;
Zones 2 and 3

**Figure 57: Axial and Hoop Load Transfer Along Door Cut-Out Corner;
Zones 4 and 5**

Figure 58: Axial and Hoop Load Transfer in Vertical Leg of Doubler; Zone 6

3.3 Comparison of Results from "Before Doubler" and "After Doubler" Tests

Several of the preceding sections have already highlighted the strain reductions produced by the composite doubler. These strain reductions, 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 stiffness would attract undesirably large loads). Figures 59 - 63 are plots of data obtained before and after the doubler was installed. Figure 59 shows far field strains away from the doubler-affected zone; these strains should be the same for both tests. The equivalency of these "before" and "after" strains indicates that the two tests were repetitive (applied loads and boundary conditions). Figures 60 and 61 compare hoop and axial strains along the top of the doubler and near the crack reference area of the door handle cut out. Figures 62 and 63 show hoop and axial strain reductions around the door corner radius. Table 5 lists some of the strain reductions achieved through the installation of the composite doubler. The strain reductions were primarily in the 40% - 60% range.

Strain Gage Number (Direction)	Strain Before Doubler ($\mu\epsilon$)	Strain After Doubler ($\mu\epsilon$)	Percent Reduction in Strain	Location
14 (hoop)	892 $\mu\epsilon$	473 $\mu\epsilon$	47 % *	Upper, horizontal leg
48 (axial)	712	264	63 % *	Crack ref. location at door handle cut-out
50 (hoop)	416	49	88 % *	Crack ref. location at door handle cut-out
27 (axial)	792	509	36 %	Upper portion of door corner radius
32 (hoop)	1080	697	35 % *	Mid-point of door corner radius
33 (axial)	757	380	50 %	Mid-point of door corner radius
38 (hoop)	2015	1618	20 % *	Lower portion of door corner radius

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

Table 5: Sample Strain Reductions Produced by Boron-Epoxy Doubler

**Figure 59: "Before Doubler" Versus "After Doubler" Strains in Far Field;
Similarities Show Consistency of Test Set-Up
Before and After Doubler Installation**

Figure 60: Comparison of Strains Before Versus After Doubler Installation Around Perimeter and at Door Handle Cut-Out Region (Hoop Strain Reduction)

Figure 61: Comparison of Strains Before Versus After Doubler Installation Around Perimeter and at Door Handle Cut-Out Region (Axial Strain Reduction)

Figure 62: Comparison of Strains Before Versus After Doubler Installation Around Door Corner Radius (Hoop Strain Reduction)

Figure 63: Comparison of Strains Before Versus After Doubler Installation Around Door Corner Radius (Axial Strain Reduction)

Figures 64 and 65 show similar comparisons of principal stresses before and after the doubler was installed. Table 6 reveals that principal stress reductions of 20% - 50% were produced by the reinforcing doubler.

Gage Identity	Principal Stress Before Doubler(psi)	Principal Stress After Doubler (psi)	Percent Reduction in Stress	Location
R.2, Ch. 12,13 & 14	12701 psi	6879 psi	46%	Outside Skin at Door Corner Radius
R.4, Ch. 24,25 & 26	9574	6205	35	Above Door Cut-out (Outside Skin)
R.5, Ch. 27,28 & 29	7448	4911	34	Above Door Cut-out (Inside Skin)
R.6, Ch. 30,31 & 32	15299	12881	16	Outside Skin at Door Corner Radius
R.7, Ch. 33,34 & 35	15066	10259	32	Inside Skin at Door Corner Radius
R.8, Ch. 36,37 & 38	22366	19363	14	Beside Door Cut-out (Outside Skin)
R.10, Ch. 48,49 & 50	13246	6221	53	Above Door Handle Cut-out

Table 6: Principal Stress Reductions Produced by Boron-Epoxy Doubler

Figure 64: Comparison of Principal Stresses Before and After Doubler Installation; Area Around Perimeter and at Door Handle Cut-Out

Figure 65: Comparison of Principal Stresses Before and After Doubler Installation; Area Around Door Corner Radius

3.4 Comparisons Between Experimental and Analytical Results

3.4.1 Finite Element Model of Doubler Installation

The performance of the 72 ply composite doubler was analyzed using a finite element model (FEM) of the fuselage structure containing the passenger door [1]. The biaxial pressure and axial load tests conducted on the door surround structure were limited in that they only simulated cabin pressure and did not account for flight loads. The FEM analysis, however, included the complete spectrum of flight and pressurization loads. The tests described in this document were used to assess the general performance of a composite doubler and to validate the analytical model. Results from the validated FEM

were then used to predict the doubler stresses and the reduction in stress in the aluminum skin at the door corner during maximum flight load scenarios.

The FEM of the door region included half of the fuselage circumference between fuselage stations FS769 and FS963. A total of 10,150 elements and 167,000 degrees-of-freedom were used to model the aluminum skin, the composite doubler, and reinforcing elements such as longerons, frames, and aluminum doublers [1]. Figure 66 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, is highlighted in the Fig. 66 inset. A baseline model was constructed without the doubler so that before and after doubler comparisons could be made. Analysis results were obtained for the following load conditions: internal pressure, fuselage down bending, fuselage side bending, and cooling to -65°F. Stresses stemming from the various load conditions were isolated so that direct comparisons could be made to the pressure only tests described here.

Validation of the FEM consisted of comparisons between the analytical results and strain gage measurements obtained during full-scale tests. Data from initial airframe tests conducted by Lockheed [50] were used to make strain comparisons at major substructural elements: stringers and longeron at fuselage Water Line 280. In general, the FEM predicted stresses were within 20% of the Ref. [50] data. This portion of the model validation is described in Ref. [1]. A second model validation effort involved comparisons with the biaxial test results described in this document. This part of the FEM validation is presented in Sections 3.4.2 and 3.4.3. Both comparisons with experimental data indicated that the FEM had sufficient accuracy to support the Boron-Epoxy doubler design and analysis.

3.4.2 Before Doubler Installation

Stresses and strains in the door surround structure without the composite doubler were calculated using the baseline FEM [1]. The loads applied to the model matched those generated during the actual "before doubler" tests (reference load cell data listed in Table 1). Appropriate boundary conditions were applied to mimic the restraints produced by the biaxial test facility. Figures 67 and 68 compare experimental data (principal or hoop/axial stresses at various gage locations) and analytical stress curves calculated by the FEM. In general, the agreement is quite good with the FEM predicting higher stresses than those determined from the strain gage data. This indicates a desirable conservatism in the analytical results. Most of the analytical stresses were within 15% of the test stresses. At the peak loads, the ratio of FEM vs. experimental stresses ranged from a minimum of 75% at R-4 to a "best fit" of 98.3% in the critical door corner area (100% = perfect data match). Figures 67 and 68 show that the: 1) FEM correctly maps out the stress field in the door corner region, and 2) the FEM correctly determined the principal stress directions.

Figure 66: Finite Element Model of Fuselage Section and Composite Doubler

Figure 67: Comparison of Experimental and Analytical (FEM) Stresses for the Fuselage Section Without the Composite Doubler

Figure 68: Comparison of Experimental and Analytical (FEM) Stresses for the Fuselage Section Without the Composite Doubler

3.4.3 After Doubler Installation

The baseline FEM was modified to include the composite doubler. Again, the loads applied to the model matched those generated during the actual “after doubler” test (reference load cell data listed in Table B-4). Figure 69 shows comparisons between the measured and FEM-produced membrane stresses in the parent aluminum skin. Figure 70 compares experimental and analytical strain levels in the composite doubler. In both cases the FEM is able to accurately predict the stress/strain field in the door corner region.

For the biaxial gage comparisons in Figures 69 and 70, the FEM results are primarily within 10% of the experimental data. The analytical results were slightly better on the doubler than on the aluminum. The principal stress values determined from the test data are contrasted with the corresponding FEM stresses in Figures 71 and 72. Once again, excellent agreement is obtained with the FEM. 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.

Figure 69: Comparison of Experimental and Analytical Membrane Stresses in the Aluminum Skin; With the Doubler Installed

Figure 70: Comparison of Experimental and Analytical Strains in the Composite Doubler

Figure 71: Comparison of Experimental and Analytical (FEM) Principal Stresses in the Aluminum Skin; With the Doubler Installed

Figure 72: Comparison of Experimental and Analytical (FEM) Principal Strains in the Composite Doubler

3.4.4 Overview of FEM in Light of Full-Scale Test Results

Results of the FEM analysis for the baseline structure (no doubler) indicated a large stress concentration in the corner region that includes significant out-of-plane bending. The modified FEM with the composite doubler installed showed that the doubler reduced peak stresses in the corner region by approximately 30%. The experimental data agrees with this analytical result. Thus, the computer analysis and test data both verified the primary goal of the reinforcing doubler: reduce stress risers. Since model and test agreement was obtained for the stress/strain fields before and after the doubler, then the 40% - 60% load transfer levels have been validated. Furthermore, the FEM was able to

assess the damage tolerance and crack mitigation capabilities of the doubler using the full flight load spectrum. 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. Additional insights into the analysis and complete listings of the safety margins and limit stresses are available in reference [1].

3.5 Nondestructive Inspection Results

Eddy current (EC) 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 [18], [27-28], [45-46], and [51] describe the applicable NDI techniques in detail, however, for completeness, the critical NDI techniques for the door corner, ultrasonics and X-ray, are addressed in Sections 3.5.2 and 3.5.3.

3.5.1 Inspection Intervals for L-1011 Aircraft

One of the major outputs from the doubler design effort was the determination of the inspection intervals to accompany the composite doubler installation. The existing inspections, X-ray, bolt hole EC, and surface scan EC [51], were retained with the following inspection intervals:

<u>NDI Technique</u>	<u>Critical Crack Length (in.)</u>	<u>Inspection Interval</u>
1. X-ray	1.00	after 1st yr. and 4,500 flights
2. ECSS	0.35	after 1st yr. and 5,690 flights
3. BHEC	0.07	after 1st yr. and 10,657 flights

These inspections detect cracks in the aluminum structure. An ultrasonic (UT) inspection is used to locate interply delaminations in the composite laminate and laminate-to-aluminum disbonds in the doubler installation. The reference [1] analysis determined the following UT inspection intervals for the composite doubler:

<u>NDI Technique</u>	<u>Critical Disbond or Delamination (in.)*</u>	<u>Inspection Interval</u>
Ultrasonic	0.50 diameter	after 1st 30 days
Visual, Coin Tap	5% of total area	after 1st 6 months
Ultrasonic	0.50 diameter	after 1st year
Ultrasonic	0.50 diameter	every 4,500 flights

* The maximum allowable flaw size of 0.5" diameter pertains to the perimeter of the doubler footprint within one inch of an edge of the doubler. Away from the doubler edge taper region, the maximum allowable contiguous flaw is equal to 5% of the total, full-thickness area of the doubler.

The close scrutiny of the doubler indicated by the frequent inspections (short intervals) listed above is due to the newness of this composite doubler technology in commercial aircraft applications. It is believed that with the accumulation of successful flight history, the inspection intervals will be extended and, in some cases, removed altogether. Note that after the initial close surveillance of the doubler (through first year following installation), the inspection interval reverts back to every major overhaul of the aircraft (i.e. D-check or heavy maintenance visit every 4,500 flights).

3.5.2 Ultrasonic Inspections

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. 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, the 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, time-of-flight values and signal waveforms.

The automated (motorized) C-Scan system used by the AANC to inspect bonded composite doublers is shown in Figure 73. The scanner is inspecting the composite doubler on the L-1011 fuselage test article. The scanning unit containing the transducer is moved over the surface of the test piece using a search pattern of closely spaced parallel lines. A mechanical linkage connects the scanning unit to X-axis and Y-axis position indicators which feed position data to the computer. 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. Additional information regarding scanner deployment, use of calibration blocks, and the setting of critical data acquisition gates is available in reference [18].

A UT inspection procedure was developed by the AANC and approved for use on L-1011 aircraft by Lockheed-Martin [45]. This procedure was applied to the doubler on the door surround structure test article. Figure 74 shows a sample C-scan from the ultrasonic

inspection of a composite doubler calibration standard. The standard contains engineered flaws which are used to set up the NDI equipment. It can be seen that a complete map of the composite doubler, which highlights any irregularities, can be obtained. Nonuniformities in the composite laminate and singular features, such as the Boron fibers, create color variations in the C-Scan. This demonstrates the sensitivity of the technique. Actual disbond and delamination flaws are revealed by continuous and distinct signal loss areas [18, 27-28]. Scans utilizing four different gate settings were used to accommodate the various doubler thicknesses in the L-1011 doubler (i.e. taper in doubler from 72 plies to 4 plies at the edge) [45].

Figure 73: Automated Ultrasonic Scanner Inspecting the Composite Doubler on the L-1011 Fuselage Test Article

3.5.3 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 the 72 ply doubler [18, 27]. Radiography was demonstrated to be a very effective inspection method to interrogate the interior of the parent material covered by the composite doubler. X-ray inspections are as effective as before a doubler is installed.

Several fatigue crack specimens and the L-1011 fuselage section were inspected through the 72 ply composite doubler. To form a basis of comparison, X-rays were also taken without the doubler placed over the cracked specimens. Details of the doubler are as follows: 72 ply, multi-axial lay-up with a fiberglass top coat (as per the Lockheed L-1011 doubler design drawing). A single screen lightning protection ply was placed on top of the doubler to assess any degradation in the X-ray caused by this copper mesh. The

specimens included 1st layer, 2nd layer, EDM notch and fatigue cracks ranging from 0.05" to 1.0" in length.

For the L-1011 fuselage section, the X-ray inspection, currently used by Delta to detect cracks around the door, was used after the doubler was installed [51]. Results from these inspections were used to determine the equipment settings needed to obtain a suitable image resolution while maintaining a film density of between 2 and 3 (as required by L-1011 NDT Manual). For these inspections, portable field equipment was deployed in order to mimic inspections carried out in the hangar.

Figure 74: Sample C-Scan Image from the Ultrasonic Inspection of a Composite Doubler Calibration Standard with Engineered Flaws

The required damage detection threshold for cracks under the doubler is 1.0". X-ray images showed the ability to detect cracks less than 1.0" in length. Fatigue cracks on the order of 0.38" in length were found under 0.40" thick (72 ply) Boron-Epoxy doublers. Comparisons with X-rays taken without composite doublers demonstrated that while the doubler may darken the X-ray image slightly it does not significantly impede the X-ray inspection. The initial set-up (80 kV, 12 mA 152.4 cm source-to-film-distance and 30 second exposure time) on medium speed film produced a film density of 0.98. Power and exposure times were adjusted in order to maintain the specified film density of between 2 and 3. 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. The full set of the X-rays described above were reviewed with NDT personnel from Delta and Lockheed.

This Page Left Intentionally Blank

**PART II: SUBSIZE DOOR CORNER TEST
("C" SECTION TEST)**

This Page Left Intentionally Blank

4.0 Subsize Door Corner Test

4.1 Objective of Test

The purpose of this test was to: 1) verify composite doubler analysis techniques, and 2) experimentally assess the potential for delaminations and disbonds in extremely thick composite doubler installations when subjected to large strains on a representative geometry. Disbonds are defined as a loss of adhesion between the Boron-Epoxy doubler and the aluminum plate and delaminations are defined as loss in adhesion between adjacent Boron-Epoxy plies. The specimen and load configuration was designed to provide strain/stress levels at the critical areas which exceed the design limit loads.

4.2 Subsize Door Corner Test Specimen

The test specimen is a subscale mock-up of a skin cut-out with a corner reinforced by a composite doubler. The important geometric variables match the configuration on the L-1011 aircraft. Figure 75 shows the basic "C" section geometry with one corner, around an appropriate radius, containing a bonded Boron-Epoxy doubler. This figure is a detailed engineering drawing of the test specimen showing the exact location of the load application points. The composite doubler was fabricated and installed in accordance with the Lockheed-Martin design for the L-1011 aircraft (drawing LCC-7622-378). The aluminum substrate was 7075-T6 material ($\sigma_y = 73$ ksi, $\sigma_u = 83$ ksi). Figure 76 shows the dimensions of the doubler including the boundaries of the taper region where the doubler thickness is gradually reduced from 72 plies down to 4 plies at the edge. The 7" radius of the cut-out matches the actual door surround structure on the L-1011. Reinforcing angle brackets were added to the top, bottom, and side of the test article in order to stabilize the structure (reinforce against out-of-plane deformations) and to provide bearing surface for the load application points. It should be noted that the specimen plate and reinforcing brackets are larger at the base of the test article (see Fig. 75) in order to control the location of the maximum stresses (i.e. force the maximum stress to occur in the upper, composite doubler corner).

The aluminum substrate was 0.25" thick (maximum variation in aluminum thickness was 0.01") and the composite doubler was 0.41" thick (maximum variation in composite patch thickness was 0.013"). Residual curvature in the specimen, caused by the difference in coefficients of thermal expansion between Boron-Epoxy and aluminum, was measured. It was determined that there was no more than a 1/8" bow in the specimen.

**Figure 75: Engineering Drawing of Subsize Door Corner
("C" Section) Test Specimen**

Figure 76: Boron-Epoxy Doubler Dimensions for "C" Section Test

Figure 77 shows a comparison of the stress levels obtained from several finite element model (FEM) analyses. Executions of the FEM for the complete L-1011 door surround structure are shown in solid lines while the FEM results for the subscale door corner test article are represented by a dotted line. It can be seen that for the actual L-1011 structure, the complete flight load scenario with fuselage pressurization has a maximum stress level of 51 KSI at the 67° location around the door cut-out radius. It can also be seen that the subscale door corner test closely matches the stress field for this design load condition. In fact, the maximum stress levels exceed the expected design loads so this test is conservative. Thus, this test is capable of subjecting the Lockheed-Martin doubler design to the appropriate peel stress scenario. The fuselage pressurization only curve is shown to point out that the maximum stresses in the corner radius of the doubler are not generated by pressurization alone (i.e. hoop tension). Rather, the maximum stresses are the result of the pressure loads combined with the flight shear/bending loads.

Figure 77: Comparison of Stress Fields for Different L-1011 and Test Specimen Limit Load Conditions

5.0 Structural Test Loads and Procedures

Figure 78 is a photograph of the test specimen mounted in the uniaxial test machine. A brittle white paint, placed around the perimeter/edge of the doubler to aid visual disbond and delamination detection, is visible. Figure 79 contains two close-up photographs of the test specimen. These views show the strain gage installations and the doubler taper at each end of the horizontal and vertical legs. The test axial load, P , was applied in 500 lb. increments. At each load level, the test was halted to visually check for disbonds/delaminations and to check for indications of yielding/failure (i.e. load drop off). Strain gage measurements were also acquired. Three different structural tests were conducted. NDI was applied between each test to monitor any flaw growth. The three structural tests conducted were:

1. Limit Load Test to 10,300 lbs.
2. 150% Limit Load Test to 15,600 lbs.
3. Ultimate Failure Test

After reaching 10,300 lbs. in the limit load test, the load was reduced to the 20% level (2,056 lbs.) to determine if there was any strain shift. After reaching 15,600 lbs. in the 150% limit load test, the load was reduced to 0 lbs. in order to determine if there was any permanent deformation in the test article. In the ultimate failure test, the load was uniformly and continuously increased until structural failure occurred. Failure was defined as an inability to sustain an increasing load (i.e. a "knee" in load vs. displacement curve).

Figure 78: Photograph of Subsize Door Corner Test Article Mounted in a Uniaxial Test Machine

Figure 79: Close-Up Views of Subsize Door Corner Test Article

5.1 Instrumentation

Uniaxial and Rosette (0° , 45° , and 90°) strain gages were installed to monitor the strain field throughout the aluminum skin and the composite doubler. Gage locations were guided by the FEM and were concentrated in the high strain areas around the corner radius. Figure 80 shows the strain gage layout; the long axis of each gage indicates the direction of the strain component being measured. Principle stresses and principal stress directions were determined from the Rosette gages. Liquid paper "whiteout" was applied to the entire perimeter/free edge of the doubler in the thickness direction. This provided a visual tool to aid the detection of disbonds or delaminations at any point during the tests. Finally, the machine load vs. crosshead displacement was recorded to determine the total displacement of the specimen at the load application points.

Figure 80: Strain Gage Locations for Subsize Door Corner Test ("C" Section Test)

5.2 Nondestructive Inspection

Nondestructive inspection (NDI) of the test article prior to testing consisted of: 1) visual inspection for cracks, delaminations, and irregularities in the doubler and aluminum plate, 2) Thru-Transmission Ultrasonic (TTU) scan of doubler, and 3) Pulse Echo (PE) ultrasonic inspection of the doubler. TTU scanning detects the presence of disbonds, delaminations, and porosity/voids in the material and displays the results in a two-dimensional C-scan of the specimen. PE ultrasonics uses time-of-flight information (signal travel time through material) to detect the same flaws and also determine the depth at which the flaw occurs. This technique makes it possible to differentiate between an interply delamination and a disbond at the composite-to-aluminum interface. The PE ultrasonic equipment can be implemented in the field so this inspection technique was also applied during the course of the structural testing.

Figure 81 shows the TTU inspection scan of the test article before structural testing. It was found that the specimen contained a flaw in the upper, horizontal leg of the composite doubler (see indication in Figure 81). The flaw was shaped like an ellipse with an approximate major and minor axis of 2" and 1", respectively. Thus, this test had an element of damage tolerance since the specimen was not in an unflawed, most desirable state. PE ultrasonics was able to determine that the flaw was a disbond at the composite-to-aluminum interface. Figure 82 shows a sample PE output from a bonded and disbonded doubler. The change in waveform - lack of a ringdown return signal from the aluminum plate - indicates the presence of the flaw and the fact that it occurs at the bond to the aluminum skin.

Figure 81: Through Transmission Ultrasonic C-Scan of Composite Doubler Before Structural Testing

No Disbond Present

Disbond Present

Figure 82: Pulse-Echo Ultrasonic A-Scan Waveforms Showing Good (Bonded) and Flawed (Disbonded) Composite Doubler Areas

This Page Left Intentionally Blank

6.0 Subsize Door Corner Test Results

6.1 Limit Load Test

Strain data from the Limit Load Test are listed in Table B-7. The maximum applied load in this test was 10,300 lbs. This generated the maximum design strains around the door corner area.

6.1.1 Strains in the Aluminum Substrate

Figure 83 plots the strains in the aluminum plate. In the vertical and horizontal legs, away from the radius of curvature area, the strains are similar and are relatively low. The maximum strain levels in the legs, at 10,300 lbs. load, were on the order of 2,200 $\mu\epsilon$ to 2,600 $\mu\epsilon$. Around the 7.0" radius of curvature in the aluminum, the strains are larger with the maximum value occurring at the 67° location (see Figure 80 and ref. strain channels 7-12). Note that this location for maximum strain agrees with the analytical prediction as shown in Figure 77. The maximum strain levels around the corner area, corresponding to the maximum design limit load of 10,300 lbs., are: 1) 4029 $\mu\epsilon$ at 45°, 2) 4959 $\mu\epsilon$ at 67°, and 3) 3343 $\mu\epsilon$ at 90°. Finally, the strain levels on the front and back of the plate at the 67° location were found to be approximately the same (4959 $\mu\epsilon$ for Channel 8 vs. 4696 $\mu\epsilon$ for Channel 7).

Figure 83: Strains in the Aluminum Substrate (Limit Load Test)

6.1.2 Strains in the Composite Doubler

The strain field in the composite doubler was similar to that of the parent aluminum plate, however, the magnitudes were less than half of those found in the aluminum. Figure 84 shows the strains measured in the Boron-Epoxy doubler. The vertical and horizontal legs, away from the 5.3" radius of curvature, have the lowest strains (maximum values range from 135 $\mu\epsilon$ to 849 $\mu\epsilon$). The largest measured strain occurred at the 67° location around the doubler's radius and had a maximum value of 1991 $\mu\epsilon$. This is approximately 40% of the maximum strain in the adjacent aluminum plate.

Figure 84: Strains in the Composite Doubler (Limit Load Test)

6.1.3 Rosette Strain Elements in the Aluminum and Composite

Results recorded by the two Rosette strain gages are shown in Figure 85. Rosette 1 (Channels 10, 11, and 12) experienced tension loads. The lateral strain component (Channel 12) perpendicular to the load, P , was negligible while the axial component (Channel 10) in the direction of the load and the 45° shear component (Channel 11) were approximately equal throughout the test. These strain levels indicate that the Principal

Strain direction is equidistant between the axial and shear direction, or at 67° . Principal stresses/strains and their direction will be discussed further in Section 6.3. Rosette 2 (Channel 13, 14, and 15) experienced compression loads. This indicates that the neutral axis for this structure in bending is between Rosette 1 and Rosette 2 as indicated in Figure 80.

Figure 85: Rosette 1 and 2 Strain Elements (Limit Load Test)

6.2 150% Limit Load Test

Strain data from the 150% Limit Load Test are listed in Table B-8. The maximum applied load in this test was 15,600 lbs. This test produced strains which exceeded the maximum design strains around the door corner and was used to ascertain if: 1) the aluminum would yield, and 2) the doubler would develop disbond or delamination flaws.

In general, the results are linear extrapolations of the data from the Limit Load Test, however, a slight nonlinearity (onset of yielding) was observed in the aluminum plate.

6.2.1 Strains in the Aluminum Substrate and Composite Doubler

Since the aluminum plate experiences the largest strains, it is expected that any yielding will occur here first. Figure 86 shows the strains in the aluminum plate up to the 150% limit load test level of 15,600 lbs. At the location of maximum strain - 67° around the corner radius - channels 7 and 8 began to yield slightly. Yielding began at approximately 12,500 lbs. when the strain levels reached 5700 $\mu\epsilon$ to 6000 $\mu\epsilon$. The maximum strain values for the 150% limit load test are summarized in Table 7 below. Once again, the maximum strains in the composite are roughly 30-40% of the maximum strains in the adjacent aluminum plate.

Figure 86: Strains in the Aluminum Substrate (150% Limit Load Test)

Strain Gage Number	Gage Location	Maximum Strain at 15,600 lbs. ($\mu\epsilon$)	Percent of Strain From Adjacent Aluminum Plate
1	Vertical Leg - Alum.	3636 $\mu\epsilon$	N/A
5	45° on Corner - Alum.	5873 $\mu\epsilon$	N/A
7	67° on Corner - Alum.	7370 $\mu\epsilon$	N/A
16	90° on Corner - Alum.	4859 $\mu\epsilon$	N/A
17	Horizontal Leg - Alum.	3721 $\mu\epsilon$	N/A
2	Vertical Leg - Composite	1304 $\mu\epsilon$	35.8% (vs. Ch. 1)
6	45° on Corner - Composite	1650 $\mu\epsilon$	28.1% (vs. Ch. 5)
9	67° on Corner - Composite	3083 $\mu\epsilon$	41.8% (vs. Ch. 7)
18	Horizontal Leg - Composite	1272 $\mu\epsilon$	34.1% (vs. Ch. 17)

Table 7: Summary of Maximum Strains in the Aluminum Plate and Composite Doubler for the 150% Limit Load Test

6.3 Ultimate Failure Test

In the ultimate failure test, the load was uniformly and continuously increased until structural failure occurred. The main concern with bonded doubler installations is that the doubler will disbond due to shear stresses in the adhesive layer. This test studied the failure mode of an extremely thick doubler when subjected to shear and in-plane bending loads (i.e. large peel stresses). Strain data from the Ultimate Failure Test are listed in Table B-9.

6.3.1 Specimen Ultimate Failure

The Subsize Door Corner Test Article withstood a maximum load, or failure load, of 25,660 lbs. In this test, failure was defined as the point where the structure was unable to sustain an increasing load. The specimen failure load was 2.5 times the limit load for this composite application. Thus, the safety factor for this doubler design exceeds the minimum goal of 1.5.

The test specimen's failure mode can be described by tracing the structure's response up through failure. 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. Recall, that an inspection of the specimen prior to testing revealed a disbond flaw at this location (see Section 5.2). This probably acted as the origin of the propagating disbond. The flaw

in the adhesive layer propagated along the horizontal leg of the doubler and around the door corner (see Figure 80). Also, a second phenomenon occurred simultaneous with the disbond growth. Because the doubler created a shift in the neutral axis, secondary 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 ability of the structure 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.

Figures 87 and 88 show photographs of the test specimen after failure. A crack line in the brittle white coating along the perimeter/free edge of the doubler clearly traces out the growth of the adhesive disbond. Figure 87 shows the disbond growth along the top and bottom of the horizontal leg. Figure 88 traces out the disbond growth around the corner radius. Both figures show the line of fracture (Y direction) across the horizontal leg of the doubler. An additional description of the specimen failure is provided in Section 6.5 "Post-Failure Nondestructive Inspection."

6.3.2 Strains in the Aluminum Substrate

During the Ultimate Failure Test, several areas in the aluminum plate yielded and exhibited nonlinear strain behavior. Figure 89 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 $5700 \mu\epsilon$ while in this test, yielding began at approximately $7000 \mu\epsilon$. 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 89. The maximum strain levels measured in the aluminum just prior to the failure load of 25,660 lbs. (last measurement at 25,060 lbs.) are listed in Table 8 below.

A. Disbond Growth in Top of Horizontal Leg

B. Disbond Growth in Bottom of Horizontal Leg

Figure 87: Cracks Along Base of Doubler, At Interface With Aluminum Plate, Reveal Disbond Growth During Ultimate Failure Test

Figure 88: Cracks Along Base of Doubler at Interface with Aluminum Plate Reveal Disbond Growth Around Corner Radius

Figure 89: Strains in the Aluminum Substrate Up To Specimen Failure

Strain Gage Number	Gage Location	Maximum Strain at 25,060 lbs. ($\mu\epsilon$)
1	Vertical Leg - Alum.	6,058 $\mu\epsilon$
5	45° on Corner - Alum.	10,197 $\mu\epsilon$
8	67° on Corner - Alum.	14,145 $\mu\epsilon$ *
16	90° on Corner - Alum.	8,243 $\mu\epsilon$
17	Horizontal Leg - Alum.	6,335 $\mu\epsilon$

* extrapolated value - gage lost at 23,369 lbs. load

Table 8: Summary of Maximum Strains Measured in the Aluminum Plate for the Ultimate Failure Test (P=25,060 lbs)

Figure 90 shows the strains measured in the aluminum angle (Channel 19) which stiffened the top of the C-section test article. Compressive strains were measured in excess of $-8,800 \mu\epsilon$. The maximum strain recorded prior to failure of the test article was $-7,400 \mu\epsilon$ at a load of 25,060 lbs. Shortly after this measurement, the specimen failed and this stiffening angle bracket buckled.

Figure 90: Compressive Strain Along Top of Reinforcing Aluminum Angle (Ultimate Failure Test of "C Section")

6.3.3 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 91 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 indicated in the figure, the nonlinearities in the curves probably stem from the yielding in the aluminum substrate and the corresponding nonlinear 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.

The vertical and horizontal legs, away from the 5.3" radius of curvature, have the lowest strains. The maximum values, at 25,060 lbs. load, range from 2441 $\mu\epsilon$ in the vertical leg (Channel 2) to 2169 $\mu\epsilon$ in the horizontal leg (Channel 18). The largest strain occurred at the 67° location around the doubler's radius (Channel 9) and had a maximum value of 6979 $\mu\epsilon$ at 25,060 lbs. load. The maximum strain levels measured in the composite doubler just prior to the failure load of 25,660 lbs. (last measurement at 25,060 lbs.) are listed in Table 9 below. The maximum strains in the composite are roughly 35-50% of the maximum strains in the adjacent aluminum plate. This ratio is 5-10% higher than in the 150% Limit Load Test indicating that the doubler is taking a greater percentage of the load as the aluminum yields. Figure 92 compares the aluminum and Boron-Epoxy strains for the critical area around the cut-out radius in the C- section.

Strain Gage Number	Gage Location	Maximum Strain at 25,060 lbs. ($\mu\epsilon$)	Percent of Strain From Adjacent Aluminum Plate
6	45° on Corner - Composite	3635 $\mu\epsilon$	35.6% (vs. Ch. 5)
9	67° on Corner - Composite	6979 $\mu\epsilon$	49.3% (vs. Ch. 8)
18	Horizontal Leg - Composite	2169 $\mu\epsilon$	34.2% (vs. Ch. 17)

Table 9: Summary of Maximum Strains Measured in the Composite Doubler for the Ultimate Failure Test (P=25,060 lbs.)

6.3.4 Rosette Strain Elements in the Aluminum and Composite

Results recorded by the two Rosette strain gages are shown in Figure 93. The same observations presented in Section 6.1 also apply here. Rosette 1 (Channels 10, 11, and 12) experienced tension loads. The lateral strain component (Channel 12) perpendicular to the load, P, was negligible while the axial component (Channel 10) in the direction of the load and the 45° shear component (Channel 11) were approximately equal throughout

the test. These strain levels indicate that the Principal Strain direction is equidistant between the axial and shear direction, or at 67° . Rosette 2 (Channel 13, 14, and 15) experienced compression loads, however, the strain magnitudes were relatively small. This indicates that the neutral axis for this structure in bending is between Rosette 1 and Rosette 2 and is closer to Rosette 2. Strain levels along the perimeter of the door corner ($7,000 \mu\epsilon$; Channel 9) were much larger than the strains at the center of the doubler ($-2,000 \mu\epsilon$; Channel 13). This supports the design assumption and analyses results 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.

Figure 91: Strains in Composite Doubler Up To Specimen Failure

Figure 92: Comparison of Critical Strains in Aluminum and Composite Material Showing Load Transfer into Doubler

Figure 93: Rosette 1 and 2 Strain Elements (Ultimate Failure Test);
Rosette 1 (10,11,12) on Aluminum;
Rosette 2 (13,14,15) on Composite

6.3.5 Maximum Principal Stresses and Directions in Aluminum and Composite

The principal stresses were determined from strain data obtained from the Rosette strain gages. Figure 80 shows the locations of the two Rosette strain gages employed in the Subscale Door Corner Test. In these three-element rectangular rosettes, gages were placed at 0° , 45° , and 90° orientations corresponding to strains ϵ_1 , ϵ_2 , and ϵ_3 , as shown in Figure 30. From these individual strains, the maximum and minimum principal stresses were calculated using equations (1) and (2). The orientation of the maximum principal stress $\sigma_{\max}$ is defined by the angle θ (see Figure 30) which is determined from equation (15).

The maximum principal stresses for Rosettes R1 (Channels 10, 11, and 12) and R2 (Channels 13, 14, and 15) during the Ultimate Failure Test are plotted in Figure 94. The stress in the aluminum plate at specimen failure was 86 KSI. The material began to yield

after a load of 15,600 lbs., however, the maximum stress in the composite-reinforced aluminum plate was greater than the Mil Handbook value listed for the ultimate tensile stress of 7075-T6 material: 83 KSI. Thus, the composite doubler was able to provide sufficient strength to allow the aluminum to survive ultimate stress levels without fracture.

Figure 95 shows the direction of the principal stress for the two Rosette gages. Rosette 1, on the aluminum plate, consistently measured a principal angle of $22^\circ - 23^\circ$. This means that strain gages mounted tangential to the 67° mark ($90^\circ - 23^\circ = 67^\circ$) around the door radius (i.e. Channels 7-9) measured the principal strains at their location. This was the location of maximum stress/strain predicted by the finite element model (see Section 6.4 below). Rosette 2 was located on the center (interior) portion of the composite doubler's horizontal leg. As Figure 93 shows, most of the strains in this area were lateral strains produced by the bending loads. The principal angle at the Rosette 2 location was approximately 80° . A 90° principal angle would correspond to pure lateral strains along the X axis (see Figure 80 for axis).

Figure 94: Maximum Principal Stresses at Rosette Gages 1 and 2

Figure 95: Direction of Principal Stress at Rosette Gages 1 and 2

6.4 Comparisons with Finite Element Model

A finite element model (FEM) was produced for the Subscale Door Corner test article in order to: 1) determine if and when strain levels could be achieved to simulate in-flight loads from fuselage downbending and internal pressure, 2) establish the location for critical strain gages, 3) compare analytical and experimental results for the structure's response up to failure, 4) verify composite doubler analysis techniques, and 5) analytically assess the potential for delaminations and disbonds in extremely thick composite doubler installations when subjected to large shear stresses. A stress contour plot produced by the FEM of the test article is shown in Figure 96. The stresses correspond to those generated by the 150% Limit Load (15,241 lbs.) condition.

Qualitatively it can be seen that the FEM predicted the highest stresses to occur at the 67° mark around the door corner radius. The large stresses are fairly localized and the magnitude reduces from 80 KSI to below 60 KSI in a span of less than 0.5". The composite doubler also has its maximum stress around the 67° mark and stresses in the doubler are 30% - 50% of those generated in the aluminum substrate. A comparison between the FEM stresses in Figure 96 and the test stresses in Figure 94 show agreement on the maximum stress at 67° around the door corner. Using Mil handbook values for Young's Modulus ($E_{ALUM} = 10.3 \text{ MSI}$) and the strain values in Table B-8, Channels 7 and 8 measure maximum stresses between 76 KSI and 78 KSI. The FEM stress contour plot

shows stresses of between 70 and 80 KSI. Similarly, the FEM contour plot shows a maximum stress in the composite doubler of 30 - 40 KSI while the strain gage in this area (Channel 9) measures a maximum stress of 34 KSI.

Figure 96: Stress Contour plot Generated by Finite Element Model for 150% Limit Load Condition (P=14,420 lbs)

Figure 97 compares the experimental and analytical stresses in the composite doubler (Rosette 1 location) for increasing load up to 13,000 lbs. Analytical results presented for loads greater than 13,000 lb. were approximated by a linear extrapolation. This is because the model predicts yielding to initiate at 13,000 lbs. The items of note are: 1) the FEM correctly traces the stress magnitude in the aluminum plate, 2) the FEM correctly locates the principal stress direction, and 3) comparisons with Figure 86 show that the FEM closely predicts the onset of yielding between 12,000 and 13,000 lbs.

Figures 98 through 100 are plots comparing the strain gage measurements with the strain field predicted by the FEM. Analytical strains in both the aluminum substrate and composite doubler compared quite well with the 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 10 summarizes the comparison between the FEM analytical predictions and the strains measured during the load tests. The comparisons are made for the maximum load prior to yielding in the aluminum plate.

Strain Gage Number	Finite Element Model - Analytical Strains	Strain Gage Measurements - Experimental Strains	Percent Difference Between Experimental and Analytical Strains
5	5311 $\mu\epsilon$	5067 $\mu\epsilon$	4.8 %
7	6153 $\mu\epsilon$	6045 $\mu\epsilon$	1.8 %
8	6153 $\mu\epsilon$	6305 $\mu\epsilon$	2.4 %
9	2912 $\mu\epsilon$	2595 $\mu\epsilon$	12.2 %
16	5225 $\mu\epsilon$	4215 $\mu\epsilon$	24 %
19	-2874 $\mu\epsilon$	-3351 $\mu\epsilon$	14.2 %

Table 10: Summary of Comparison Between Finite Element Model and Experimental Strains at P = 13,000 lbs.

Figure 97: Comparison Between Composite Doubler (Rosette 1) Stresses and Values Predicted by Finite Element Model

Figure 98: Comparison Between Aluminum/Composite Strains and Values Predicted by Finite Element Model

Figure 99: Comparison Between Aluminum Strains and Values Predicted by Finite Element Model

Figure 100: Comparison Between Strains Measured in Angle Bracket and Those Predicted by Finite Element Model

6.5 Post-Failure Nondestructive Inspection

Nondestructive inspection (NDI) of the test article was conducted after each load test. Using the visual, Pulse-Echo ultrasonic, and Thru-Transmission ultrasonic inspection methods described in Section 2.5 and Ref. [18], it was determined that there was no disbond growth during either the Limit Load or 150% Limit Load tests. Thus, the composite doubler design began the Ultimate Failure Test having successfully demonstrated a resistance to flaw initiation and growth while enduring shear loads which exceed design levels.

After the failure test, TTU and PE ultrasonics were applied to the test article in order to define the range of disbonds, delaminations, and porosity/voids generated in the specimen during ultimate failure. Both PE ultrasonics and TTU agreed on the boundaries of the resulting flaw. Figure 101 shows the TTU inspection scan of the test article after ultimate failure. The dotted line traces out the shape of the composite doubler. The black and pink areas inside the perimeter of the dotted line represent little or no ultrasonic signal

transmitted through the specimen. This indicates the presence of a flaw through the thickness of the composite-aluminum lay-up. PE ultrasonics used time-of-flight information to detect the same flaw. It also determined that the flaw was a disbond in the adhesive layer. The brightly colored area (mostly red) in Figure 101 indicates the remaining unflawed portion in the specimen. Thus, the failure test produced a disbond which included the entire horizontal leg and the area around the door corner. The stress and strain field analysis showed that these were the areas with the highest peel stresses. A comparison between TTU images before and after ultimate failure testing clearly traces the growth of the disbond flaw. Figure 101 shows the bonded doubler prior to testing along side the failed specimen which contained the large, disbonded area. The post-failure photographs in Figures 87 and 88 show how this disbond produced a separation between the edge of the doubler and the aluminum substrate beneath it.

**Figure 101: Through Transmission Ultrasonic C-Scan of Composite Doubler
After Ultimate Failure Test**

Boron Epoxy Patch after Simulated Flight Bending/Shear Test

Ultrasonic Through Transmission Signal Amplitude Variations

Figure 102: Comparison of Ultrasonic Inspection Results Before and After Structural Testing

PART III: FOUR POINT BENDING TESTS

This Page Left Intentionally Blank

7.0 Four-Point Bending Tests

7.1 Objective of Test

The purpose of this test was to experimentally assess the potential for delaminations and disbonds in an extremely thick (0.40"; 72 plies) composite doubler installation when subjected to pure bending stresses. Bending of the fuselage skin is an important consideration and this test focused on worst case conditions for pure bending loads. The loads in this test did not simulate a particular flight condition. Rather, the bending loads were increased until the specimen failed. In this manner, it was possible to assess the design margin and mode of failure associated with high bending strains.

7.2 Four Point Bending Test Specimen

For this test series, the specimen was a beam comprised of a 72 ply Boron-Epoxy laminate (lay-up as per L-1011 doubler repair drawing LCC-7622-378) 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 103 below shows the test specimen. Three similar specimens were tested so that the Boron-Epoxy laminate could be evaluated in both compression (laminate on top) and tension (laminate on bottom) modes.

Figure 103: Composite Doubler Four-Point Bending Test Specimen

8.0 Structural Test Loads and Procedures - Four-Point Bending Tests

Prior to testing, NDI was applied to each test specimen. Pulse Echo ultrasonics was used to check for any pre-existing flaws which may affect the test results. During the structural tests, upper and lower load beams were used to support the specimen and transfer the load as indicated in the test set-up shown in Figure 103. Figure 104 is a photograph of the test specimen mounted in the uniaxial test machine. The brittle white coating around the perimeter, used to visually detect delaminations, is visible.

**Figure 104: Photograph of Four Point Bending Specimen
Mounted in Test Fixture**

The beam bending flexure test was modeled after the ASTM 53-600, method 330 for four point bending. The bottom reaction loads were spaced 12" apart (S) and the P/2 machine loads were spaced 6" apart (S/2). The bending test load, P, was applied in 100 to 200 lb. increments. At each load level, the test was halted to visually check for disbonds/delaminations and to check for indications of yielding or failure (i.e. load drop off). Strain gage measurements were also acquired. The load was uniformly and continuously increased until structural failure occurred. Failure was defined as an inability to sustain an increasing load (i.e. a "knee" in load vs. displacement curve).

The test was repeated three times. Test 1 (Sample 4B-1) applied the load with the composite laminate on the bottom. In Test 2 (Sample 4B-2), the laminate was placed in

compression on the top of the beam. Figure 105 shows the difference in these two test configurations. Test 3 (Sample 4B-3) repeated the configuration which showed the lowest failure load from Tests 1 and 2: composite laminate on bottom.

Figure 105: Pure Bending Test Configurations Showing Laminate on Top and Laminate on Bottom Set-Ups

8.1 Instrumentation

Uniaxial strain gages were installed in the center of the test specimen (Fig. 103) to monitor the strains in the pure bending area of the aluminum skin and the composite doubler. Strain gage 1 measured the X-direction strains at the edge of the aluminum beam. Strain gages 2 and 3 measured the X-direction strains in the composite material. These two gages were mounted on each side (edge) of the beam's 1" width. This was done to monitor strain levels at the onset of a delamination which could occur on either side of the beam. A brittle white coating was applied to the entire perimeter/free edge of the doubler in the thickness direction. This provided a visual tool to aid the detection of disbonds or delaminations at any point during the tests.

9.0 Four-Point Bending Test Results

Pre-test inspections did not reveal any flaws in the test specimens. The maximum moment generated in the test specimen is given by the relation:

$$M_{\max} = PS/8 \quad (19)$$

The stress produced by this moment can be calculated using the following equation:

$$\sigma = M y/I \quad (20)$$

Where:

I = moment of inertia of cross section

M = moment in in.-lbs.

P = load in lbs.

S = distance between load supports

y = distance from neutral axis to point of interest

σ = stress in PSI

Using an ultimate stress value of 65 KSI for the composite material and Equations (19) and (20), it was possible to estimate the load, P , necessary to induce failure in the composite laminate. In each of the four point bending tests the estimated failure load, based on the resulting stress levels in the composite laminate, was $P=1,200$ lbs. All three test specimens surpassed this load level before failing.

9.1 Strains and Failure in the 4B-1 Test Specimen

The 4B-1 specimen was tested with the composite laminate on the bottom. Thus, the plies in the laminate experienced tension loads. Figure 106 plots the strains in the 4B-1 specimen during its four point bending test. The strains at the top (aluminum) and bottom (composite), although opposite in direction, are equal in magnitude. The maximum strains reached $8,000 \mu\epsilon$ before the specimen failed at $P = 4,058$ lbs. Note that these strains are 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 also below one of the load application points. Figure 107 shows the delamination failure in the 4B-1 specimen. Post test thickness measurements indicated that the approximate locations (i.e. ply number) of the delaminations were as listed in Table 11(a).

Delamination	Delamination Height from Aluminum Substrate	Approximate Ply Number
1	0.035"	5
2	0.030"	52
3	0.320"	55
4	0.330"	58
5	0.364"	63

Table 11(a): Measured Delamination Layers in Specimen 4B-1

It was possible to visually inspect the surfaces between the delaminated plies and to check the ply orientations at the points of failure. The delaminations appeared to occur primarily at interfaces to 45° and 90° plies. This is reasonable since 45° and 90° plies have less strength than the 0° plies in the X-direction (i.e. direction of bending strains).

Figure 106: Bending Strains for Sample 4B-1 Test with Composite on Bottom

Figure 107: Delamination Failure Mode of Four Point Bending Specimen 4B-1 with Composite Laminate on Bottom

9.2 Strains and Failure in the 4B-2 Test Specimen

The 4B-2 specimen was tested with the composite laminate on the top. Thus the plies in the laminate experienced compression loads. Figure 108 plots the strains in the 4B-2 specimen during its four point bending test. Again, the strains at the top (composite) and bottom (aluminum) are equal in magnitude but opposite in direction. The maximum strains reached 12,000 - 13,000 $\mu\epsilon$ before the specimen failed at $P = 6,030$ lbs. These strains are on the order of the strains experienced by the Subsize Door Corner Test Article during its Ultimate Failure Test. Thus, the resistance to delamination for this bending direction greatly exceeds the desired design margins for the door installation. In this test, the aluminum substrate began to yield before failure. The nonlinear strain response in all three strain gages began at approximately 9,000 $\mu\epsilon$.

As in specimen 4B-1, the failure mode was interply delamination, however, there was a single delamination, covering several adjacent plies. The delamination traveled the entire 13" length of the beam. After delamination, the laminate fractured at one of the load application points. Figure 109 shows the delamination failure in the 4B-2 specimen. It can be seen that the predominant ply orientation along the failed surface is 45° .

Figure 108: Bending Strains for Sample 4B-2 Test with Composite on Top

Figure 109: Delamination Failure Mode of Four Point Bending Specimen 4B-2 with Composite Laminate on Top

9.3 Strains and Failure in the 4B-3 Test Specimen

In order to repeat the test with the lowest failure load, specimen 4B-3 was tested with the composite laminate on the bottom. All aspects of the test - failure load and mode - were similar to the 4B-1 test. Figure 110 plots the strains in the 4B-3 specimen during its four point bending test. As in the 4B-1 specimen test, the maximum strains reached $8,000 \mu\epsilon$ before the specimen failed at $P = 4,300$ lbs. 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 near both of the load application points. Figure 111 shows the delamination failures in the 4B-1, 2, and 3 four point bending specimens. Post test thickness measurements indicated that the approximate locations (i.e. ply number) of the delaminations were as listed in Table 11(b).

Figure 110: Bending Strains for Sample 4B-3 Test with Composite on Bottom

Delamination	Delamination Height from Aluminum Substrate	Approximate Ply Number
1	0.060"	9
2	0.297"	51
3	0.318"	55
4	0.356"	61

Table 11(b): Measured Delamination Layers in Specimen 4B-3

Each of these delaminations are within several plies of a delamination found in specimen 4B-1 (see Table 11(a)). Once again, the delaminations appeared to occur primarily at interfaces to 45° and 90° plies.

Figure 111: Failed Four Point Bending Specimens 4B-1, 2, and 3

PART IV: COMPOSITE DOUBLER INSTALLATION ON A COMMERCIAL AIRCRAFT

This Page Left Intentionally Blank

10.0 L-1011 Composite Doubler Installation

The best proof that the composite doubler repair technique is viable on commercial aircraft was provided by the installation of a large doubler on a Delta Air Lines L-1011. The aircraft is currently operating in the Delta fleet (#755, S/N 193Y-1185) and was undergoing a Heavy Maintenance Visit (HMV) during the time of the installation. 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 as per the approved process specifications. The aircraft returned to its trans-Atlantic service (Atlanta and New York to Europe routes) and subsequent inspections following 45 days, 6 months, and one year of activity did not reveal any flaws in the installation.

The door corner doubler represents the first application of an FAA-approved bonded composite doubler (non-decal) on a U.S. commercial aircraft. FAA approval was obtained on all engineering data and procedures supporting this technology. The installation was driven by the series of documents produced by this program and summarized earlier in this report. The AANC will continue to inspect this doubler in order to accumulate history regarding the long-term doubler endurance under actual flight conditions.

10.1 Composite Doubler Repair of Delta L-1011 Aircraft

Figure 112 shows the Delta aircraft undergoing maintenance and the doubler installation staging area at the right side, P3 passenger door. Figure 113 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 [43]. The doubler was placed in an autoclave (150°F, 80 psi) to debulk/densify the doubler and to bleed out excess resin. The 72 ply Boron-Epoxy composite laminate (post-debulk) is shown in Figure 114. 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.

A fiberglass environmental protection ply and a copper mesh lightning protection ply were installed on top of the Boron-Epoxy laminate. Figure 115 shows the complete doubler lay-up as it was positioned on the L-1011 door frame. The doubler was placed on the fuselage followed by bleeder cloths, heater blankets, insulation and a vacuum bag assembly. Textron applied 12 thermocouples and Sandia installed 10 more to monitor the temperature field around the top and perimeter of the doubler, as well as, the back side of the fuselage skin and substructure elements.

A. Candidate L-1011 Aircraft Undergoing a Heavy Maintenance Visit

B. Work Stand Positioned Next to the Right Side Passenger Door

Figure 112: Delta L-1011 Aircraft During Installation of Composite Doubler

Figure 113: Anodized Door Corner Prepared for Doubler Bonding Process

Figure 114: Boron-Epoxy Laminate Following Autoclave Debulk Process

Figure 115: Composite Doubler Positioned on Door for Cure Process

After the heat blankets were turned on, the temperature ramp-up process, which included the placement of heat lamps on the back side of the skin and insulation over heat sink areas, took 2 1/2 hours. Figure 116 shows the vacuum bag arrangement and the Heat Con blanket controller during the cure process. Figure 117 shows the inside structure of the door corner area and the heat lamps that were positioned to minimize the effects of the various heat sinks. In order to optimize the uniformity and stability of the temperature field, the 190°F, six-hour cure cycle was used instead of the 210°F, four-hour cure cycle. Temperature fluctuations were less than 5°F at any location inside and outside the aircraft. The maximum and minimum temperatures were separated by approximately 25°F. A close-up view of the completed composite doubler installation is shown in Figure 118. To contrast this doubler with the conventional approach, Figure 119 shows the series of riveted, metallic plates that are traditionally used to repair the L-1011 door corner. The light colored strip running diagonally across the doubler is a rain gutter. Finally, Figure 120 shows overall views of the doubler immediately following its installation and after the application of the Delta paint scheme.

Figure 116: Cure Process - Doubler Beneath Heat Blanket, Insulation, and Vacuum Bag Assembly; Heat Controller is Shown in Foreground

Figure 117: Heat Lamps Positioned Inside Aircraft to Help Maintain Cure Temperature on Back Skin

**Figure 118: Composite Doubler Following Cure Process
(Prior to Application of Delta Paint Scheme)**

**Figure 119: Conventional Door Corner Repair with Four, Riveted Metal Plates;
Close-Up View (inset) Shows Intricate Rivet Pattern and Complex Shape of
Quadrupler Plates**

B. Composite Doubler Immediately Following Installation

B. View of Doubler After First 45 Days of Operation

Figure 120: Overall View of Composite Doubler on Delta Aircraft Fuselage

10.2 Nondestructive Inspection of Door Surround Structure and Composite Doubler

The NDI techniques deployed in this effort were capable of detecting disbonds, delaminations, and porosity in the composite doubler (ultrasonics), as well as cracks in the parent aluminum material [18, 27-28, 45-46, 51]. The door surround structure was inspected with visual (optical magnification), ultrasonic (UT), X-ray and eddy current (EC) NDI techniques before and after the composite doubler installation. Before the doubler was installed, a drawing was made of the area covered by the doubler footprint. It included the location of all rivets or surface structures which could provide a source of false flaw indications. This drawing was then used as a "template" map to aid the inspections performed after the doubler was installed.

Inspections for cracks in the parent material were carried out using an eddy current procedure applied to both the inside and outside of the structure. Bolt hole eddy current (BHEC) and eddy current surface scan (ECSS) inspections were performed prior to the doubler installation. The aluminum material adjacent to the doubler was also inspected using ECSS after the doubler was installed. Both EC procedures followed existing call-outs in the L-1011 Nondestructive Testing Manual [51]. X-ray inspections, currently in use in the door corner region, were also conducted using the procedure listed in the L-1011 NDT manual.

After the doubler was installed, disbond and delamination inspections were performed using the Sandia Labs AANC ultrasonics inspection procedure for bonded composite doublers, AANC-UT-Comp-5521/4-004 [45]. This procedure is required by reference [42] for the L-1011 application. The doubler was inspected using pulse-echo ultrasonics (UT) and an X-Y scanner system [18]. Most of the flaw detection effort was focused on the critical 1.5" wide strip around the perimeter of the doubler. A 0.5" diameter flaw is the allowable flaw size in this load transfer region. Scans were also produced in the internal region to assure that there were no flaws in excess of the 14 in.² allowed by the damage tolerance assessment. Figure 121 shows the ultrasonic scanner system inspecting the upper, tapered region of the door corner doubler. A typical scan obtained during the aircraft doubler inspection is shown in Figure 122. A disbond flaw would show up as a red area on the scan (black area on the gray scale image shown) corresponding to a loss of signal. The inspection did not reveal any flaws in the L-1011 doubler. Contrast Figure 122 (no flaws) with the Figure 123 pulse-echo ultrasonic scan of a reference standard containing engineered flaws. The standard was one of several developed to support this inspection. In Figure 123 the flaws are clearly seen as distinct dark regions (signal loss) in the C-scan image.

Figure 121: Ultrasonic Scanner System Inspecting a Tapered Region of the L-1011 Door Corner Doubler

Figure 122: Sample Ultrasonic C-Scan from L-1011 Doubler Inspection (approx. 14" x 4" scan window) - Dark Patches, Not Seen Here, Would Indicate Flaws

**Figure 123: Pulse-Echo C-Scan of Composite Doubler Reference Standard
Showing Ultrasonic Flaw Imaging**

10.3 Quality Assurance Measures

An overall approach to managing the implementation of composite doubler technology is proposed in reference [36]. Reference [36] suggests the use of an Engineering Standard to guide all design, analysis, and QA issues. A series of quality assurance (QA) measures were included in this project's composite doubler installation process [43] to assure: 1) sufficient strength in the adhesive layer, 2) sufficient strength in the Boron-Epoxy laminate, 3) proper surface preparation to allow the best opportunity for complete adherence of the doubler, and 4) the detection of any flaws in the composite doubler. The first four QA devices involve test specimens which are generated at the same time the

doubler is installed. All test specimens are produced using the same temperature and pressure profile as the aircraft doubler cure. The final QA mechanism is nondestructive inspection which is used for the initial acceptance of a composite doubler installation and for continued surveillance over the life of the doubler.

- a. Wedge Test - Two aluminum strips are bonded to the parent structure immediately adjacent to the doubler. A phenolic wedge is used to pry the strips off the aircraft. If adhesive is found on both the aluminum strip and the aircraft skin, this indicates that the adhesive fractured (cohesive failure) rather than disbonded (adhesive failure). Thus, the surface preparation is good and the full adhesive strength has been achieved. Figure 124 depicts this surface preparation QA test and the two potential failure modes.
- b. Lap Shear Test - These tests utilize one inch wide aluminum coupons with a bonded lap joint. The specimens are pulled to failure to determine the ultimate strength of the adhesive layer. The minimum strength requirement is 3,000 psi. Six specimens were tested for the L-1011 door corner doubler. They produced the following strength values: 1) 4,720 psi, 2) 4,500 psi, 3) 4,440 psi, 4) 4,880 psi, 5) 4,740 psi, and 6) 4,390 psi.
- c. Short Beam Shear - One inch wide Boron-Epoxy laminate coupons (15 ply unidirectional lay-up) are tested for laminate shear strength as per ASTM specifications. Three specimens were tested for the L-1011 laminate and an average shear value of 10.7 ksi was determined. The cured Boron-Epoxy shear strength should be at least 10.5 to 11.0 ksi.
- d. Four Point Bend - One inch wide Boron-Epoxy laminate coupons (15 ply unidirectional lay-up) are tested for ultimate tensile strength as per ASTM specifications. Three specimens were tested for the L-1011 laminate and an average tensile strength of 212 ksi was determined. The established minimum strength requirement is 180 ksi.
- e. Nondestructive Inspection - The tests outlined above determine the strength properties of the installation. However, it is still necessary to detect any flaws in the installation. Initially, the status of flaws in the doubler and bondline must be ascertained to accept the installation. Thereafter, the flaw status of the doubler, bondline, and parent material must be periodically measured. Nondestructive inspection provides the last line of defense in this regard. NDI is the only means for determining if the structural integrity of the repair area changes over time. The NDI methods introduced here and described in detail in Ref. [18] were used to evaluate the composite doubler installation. Four post-installation inspections, spanning one year of aircraft operation, have shown the doubler to be free of flaws. No flaws were detected in the L-1011 composite doubler or the doubler-to-aircraft bond line.

Two Potential Bondline Failure Modes:

Adhesive Failure

Cohesive Failure

Figure 124: Quality Assurance Wedge Test for Surface Preparation

10.4 FAA and Industry Approvals

The other important products from this program 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.

An important product of the results cited above is the Lockheed-Martin Service Bulletin 093-53-278 which allows the door corner composite doubler to be installed on all L-1011 aircraft. This installation also facilitated the formal introduction of composite doubler technology into the Delta maintenance program. Delta Air Lines, the AANC, and

Textron developed the Engineering Repair Authorization (ERA) document which must accompany any new Delta maintenance activity. The ERA generated all of the job cards necessary to support the installation and inspection of the door corner doubler.

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 L-1011 Aging Aircraft Airworthiness Directive (94-05-1). The FAA letter granting AMOC concludes that the engineering data and installation/inspection specifications listed above provide an "equivalent level of safety" as the metallic doubler called out in Service Bulletin 093-53-237.

<u>Approval Form</u>	<u>Document Number</u>
1. Lockheed L-1011 Service Bulletin	093-53-278
2. Delta Engineering Repair Authorization	340137-14AD
3. FAA Alternate Means of Compliance	1/19/97 letter ref. L96N1088

With the successful completion of the L-1011 door corner application, the FAA and AANC are now conducting a program with Boeing and Federal Express to develop, certify, and install a more generic set of composite doubler applications for a variety of common aircraft fuselage skin repairs.

11.0 Benefits of Composite Doublers on Commercial Aircraft

The process of repairing airplane structures is time consuming and must be carefully engineered. 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. The use of rivets to apply conventional doublers exacerbates these problems. When repairing multi-site or wide-spread fatigue damage using conventional methods, the close proximity of a large number of mechanically fastened repairs can lead to a compromise in the global damage tolerance of the structure. Numerous articles have addressed the myriad of concerns associated with repairing aircraft structures. Many of these repair difficulties can be addressed through the use of bonded composite doubler repairs. The benefits of composite doublers are described below. The discussion is broken down into technical and economic considerations. Table 12 compares the key features of composite doubler repairs with existing metallic doubler repair technology.

11.1 Engineering Advantages

The aircraft repair process using bonded composite doublers has numerous advantages over conventional, mechanically fastened repairs. Adhesive bonding eliminates stress concentrations caused by additional fastener holes. Composite laminates can be easily formed to fit the contour of a fuselage section. Also, the doubler can be tailored to meet specific anisotropy needs thus eliminating the undesirable stiffening of a structure in directions other than those required. Other advantages include corrosion resistance, a high strength-to-weight ratio, and potential time savings in installation.

Engineering Advantages:

1. **Adhesive bonding eliminates stress concentrations caused by additional fastener holes** - Rivet holes create stress intensity factors (magnification) of 3 times the uniform stress field away from the metal doubler. The adhesive layer produces a gradual load transfer into the doubler and eliminates all stress risers. Adhesive bonding also makes repairs possible on structures that cannot tolerate additional fastener holes.
2. **Crack Mitigation Performance** (improved fatigue life of structure) - Comprehensive fatigue testing has determined that the structure's fatigue life (resistance to crack growth) is improved 2.5 times greater than with metal doublers.
3. **Strength-to-Weight Ratio** - Modulus and strength values are three times that of aluminum yet the material is 50% lighter. Equivalent composite doublers can be up to 50% thinner than metal repairs. This reduces aircraft drag.
4. **Flexibility in Design** - The unidirectional strength of individual composite plies allows doublers to be tailored to meet specific directional strength needs. This eliminates undesirable stiffening of a structure in directions other than those required.

This feature works with item (1) to improve the fatigue life of the structure being repaired

5. **Corrosion Resistance** - Boron-Epoxy material does not corrode and will not induce corrosion in the parent material. Metal doublers corrode over time. Corrosion beneath doublers, accelerated by water entrapment, is a common problem. Corrosion inspection, removal, and repair activities induce tremendous costs in the multi-billion dollar commercial aircraft maintenance industry.
6. **Formability** - Composite laminates are easily formed to fit the contour of fuselage sections and tight radius areas such as engine cowlings and leading edges of wings. Metal doublers can be formed to fit curved surfaces but this requires an additional machining process and incurs additional costs.

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. Since it was a very complex application as compared to most repairs, it serves to emphasize the feasibility of composite doublers repairs for metal aircraft structures.

11.2 Economic Advantages

Cost-Benefit Assessment - A complete validation process must also include an assessment of the cost effectiveness of the new maintenance technique in light of the engineering advantages. This includes an analysis of the implementation costs represented by dollars, time, and resources that are used to carry out the maintenance practice (in this case, aircraft repair and subsequent inspection).

In addition to the engineering benefits noted above, bonded composite doublers can provide economic advantages. The economic advantages stem primarily from time savings in installation and the secondary effect of reduced aircraft downtime. Exact dollar values depend on the complexity of the repair installation and the number of repairs installed. In general, data accumulated to date using demonstration installations have indicated that a 50% - 60% savings in labor can be obtained. When all types of potential composite doubler repairs are considered, the commercial aviation industry could realize extensive maintenance and operation cost savings using this repair technique.

One of the most common aircraft repairs is the application of a doubler to a cracked, corroded, or dented surface skin (scab repairs). Composite doublers are particularly well suited to these type of repairs. Many of these repairs can be completed without accessing the inside of the aircraft structure. This can produce a large time savings if the

comparable metallic doubler requires inside access to install the fasteners. These type of surface skin scab repairs can be found many times on a single aircraft. Thus, economies of scale come into play and the cost savings can be substantial when applied over a carrier's entire fleet.

An important by-product of the reduced man-hours needed to effect a composite doubler repair is that it may be possible to return an aircraft to service earlier. In some cases, a composite doubler may allow for an overnight repair and eliminate any loss of service for an aircraft. Revenue loss for aircraft down time can be upwards of \$80,000 per day. With approximately 6,000 aircraft flying in the U.S. commercial fleet, reduced aircraft downtime may represent the greatest potential for cost savings.

Due to the engineering challenges discussed above, this L-1011 door corner application does not provide the optimal cost-benefit case study. However, Delta's maintenance program indicates that the composite doubler installation and inspection, although not streamlined by repetition, produced a significant savings in man-hours versus the installation of the conventional metallic repair. This savings in labor could potentially be realized six times (one per each access door) on every L-1011 in the Delta fleet.

Aircraft Repair Feature	Bonded Composite Doublers	Riveted Metal Doublers	Advantage of Composites & Notes
Stress Field	<ul style="list-style-type: none"> • No need for additional fastener holes in structure • Bond provides a uniform stress field 	<ul style="list-style-type: none"> • New holes produce stress risers and fatigue crack initiation sites • Load transfer occurs exclusively at edge of doubler 	<ul style="list-style-type: none"> • Produces gradual load transfer and more uniform stress field • Eliminates stress risers (stress magnification of 3 found in riveted doublers)
Fatigue Life	<ul style="list-style-type: none"> • Composite doubler can be tailored to provide stiffness only in the required directions • Bonded doubler provides uniform stress reduction in immediate vicinity of flaw 	<ul style="list-style-type: none"> • Isotropic material produces uniform & sometimes undesirable stiffening in all directions 	<ul style="list-style-type: none"> • Longer Fatigue Life: crack mitigation tests show less than half the crack growth over the same number of fatigue cycles (fatigue life improved by factor of 2.5) • Improved damage tolerance
Corrosion Resistance	<ul style="list-style-type: none"> • Boron-Epoxy material does not corrode or induce galvanic reaction in the parent aluminum material • Adhesive bonding process seals off material beneath it from all moisture 	<ul style="list-style-type: none"> • Metal doublers will corrode over time • Installation provides location for water entrapment between doubler and parent aluminum structure; this accelerates corrosion process 	<ul style="list-style-type: none"> • May eliminate follow-up maintenance costs (inspection, corrosion removal, replacement of metal doubler) • Avoids aggravation of initial flawed area
Aerodynamics	<ul style="list-style-type: none"> • Higher strength at reduced ply thickness allows for thinner doublers 	<ul style="list-style-type: none"> • Typical repairs are two to four plates thick (0.125" to 0.375") 	<ul style="list-style-type: none"> • Up to 50% decrease in thickness improves aerodynamics

Table 12: Comparison Between Bonded Composite Doubler Repair Technique and Riveted Metallic Doubler Aircraft Repairs

Aircraft Repair Feature	Bonded Composite Doublers	Riveted Metal Doublers	Advantage of Composites & Notes
Strength-to Weight	<ul style="list-style-type: none"> • Modulus = 28 msi • Tensile Strength = 225 ksi • Density = .066 lbs/in³ 	<ul style="list-style-type: none"> • Modulus (alum/steel) = 10 / 30 msi • Tensile Strength (alum/steel) = 64 / 80 ksi • Density (alum/steel) = 0.100 / 0.283 lbs/in³ 	<ul style="list-style-type: none"> • Strength properties exceed aluminum and steel • Improved fuel efficiency through reduction in aircraft weight (50% - 70% reduction in weight per doubler)
Method of Attachment to Aircraft	<ul style="list-style-type: none"> • Adhesive bonding 	<ul style="list-style-type: none"> • Mechanical fasteners, rivets 	<ul style="list-style-type: none"> • Certain structures, such as wing spars, cannot tolerate the addition of new holes (must be replaced rather than repaired) • Proper surface preparation and adhesive bonding processes are crucial to composite doublers
Formability	<ul style="list-style-type: none"> • Hand pressure can readily shape doubler to contoured surfaces (e.g. engine cowlings, wing leading edges) 	<ul style="list-style-type: none"> • Machining process must be employed to provide proper contour on metal doublers in tight radii areas 	<ul style="list-style-type: none"> • Eliminates additional step and associated costs

Table 12: Comparison Between Bonded Composite Doubler Repair Technique and Riveted Metallic Doubler Aircraft Repairs (continued)

Aircraft Repair Feature	Bonded Composite Doublers	Riveted Metal Doublers	Advantage of Composites & Notes
Installation Time	<ul style="list-style-type: none"> • Typical 1 ft.² fuselage skin repair (12 man-hours) 	<ul style="list-style-type: none"> • Typical 1 ft.² fuselage skin repair (40 man-hours) 	<ul style="list-style-type: none"> • Decreased aircraft down time • Maintenance cost savings due to reduced man-hours required
Material Cost	<ul style="list-style-type: none"> • Cost depends on size of doubler and number of plies • Typical 1 ft.² skin repair doubler: \$800 (20 plies) 	<ul style="list-style-type: none"> • Depends on number of plates & rivets, metal type, and forming required • Typical 1 ft.² skin repair doubler: \$300 (including machining) 	<ul style="list-style-type: none"> • Costs approximately 2.5 times comparable metal doublers • Greater material costs can be offset by savings in man-hours & decreased aircraft downtime

Table 12: Comparison Between Bonded Composite Doubler Repair Technique and Riveted Metallic Doubler Aircraft Repairs (continued)

PART V: CONCLUSIONS

This Page Left Intentionally Blank

12.0 Conclusions from Full Scale Test Validation

As a result of this program, FAA certification has been received for the use of bonded composite doublers 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 addressed these uncertainties for general composite doubler installations. In addition, a proof-of-concept aircraft installation was completed in order to demonstrate the successful consideration of all "cradle-to-grave" issues including: design, analysis, installation, damage tolerance, quality assurance, nondestructive inspection, and FAA/industry oversight. All tests described in this report were performed in extreme combinations of flaw scenarios (sizes and combinations) and excessive flight load spectrums. Thus, the performance results presented here were arrived at in a conservative manner.

Full-Scale Fuselage Testing - In this test series, a custom facility was used to evaluate the response of composite doublers on fuselage structure. The tests simulated maximum cabin pressure loads and maintained the proper proportion between hoop and axial loads. The full-scale fuselage test series validated composite doubler technology using realistic structures and loads. It showed that a properly designed and installed composite doubler is able to enhance fatigue life, transfer load away from damaged structure, and avoid the introduction of new stress risers (i.e. eliminate global reduction in the fatigue life of the structure).

This test series also demonstrated that composite doublers are able to produce extensive stress reductions in the parent aluminum structure especially in critical areas around fuselage cut-outs. Comparisons with test data obtained prior to the doubler installation revealed that stresses in the test article were reduced 30% - 50% through the use of the composite doubler.

Load transfer was another parameter used to assess the doubler's performance. This parameter is directly related to the stress reduction described above and highlights key regions in the doubler. Load transfer values indicated that the majority of the load is shifted to the doubler along the outer, tapered edge of the doubler. As a result, the allowable flaw size - which drives nondestructive inspection requirements - is smallest around the doubler's perimeter. Load transfer values measured in the fuselage panels test agreed with all test data accumulated to date [17, 19, 48-49] and the stress reduction values noted above. The load transfer into the doubler, and away from the aluminum, was consistently in the 40% - 60% range.

Strain gage data from common locations on the inside and outside of the test article showed that, during cabin pressurization, out-of-plane bending accounts for up to 50% of the total strain induced in the fuselage structure. Thus, bending effects should be a prime consideration in composite doubler designs.

Finally, this test series validated a finite element model (FEM) approach to the stress analysis of composite doubler designs and applications [1]. The use of proper constitutive properties for the Boron-Epoxy composite laminate and the adhesive layer, as per Ref. [47], is key to obtaining relevant analytical results. The FEM of the L-1011 fuselage section was able to accurately (within 10%) map out the stress field in the door corner region and correctly locate the Principal Stress directions. Stress concentrations, stress reductions, and out-of-plane bending components were also correctly identified. The validated FEM was able to assess the damage tolerance and crack mitigation capabilities of the doubler using the complete flight load spectrum. 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.

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 stiffness would attract undesirably large loads).

Nondestructive Inspection - Before the use of composite doublers can be accepted by the civil aviation industry, it is imperative that methods be developed which can quickly and reliably assess the integrity of the doubler. Primary among inspection requirements for these doublers is the identification of disbonds, between the composite laminate and aluminum parent material, and delaminations in the composite laminate. Surveillance of cracks in the parent aluminum material beneath the doubler is also a concern.

Periodic field 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. The Sandia Labs AANC has used ultrasonics (UT) to detect both interply delaminations as well as disbonds at the laminate-to-aluminum interface.

Pulse-Echo ultrasonics can be easily implemented on an aircraft using hand held inspection devices. Anomalies in A-Scan signals can be used to detect laminate flaws although signal fluctuations, caused by material nonuniformities, can create interpretation difficulties. The best compromise between field deployment and ease of signal interpretation is achieved through the use of Pulse-Echo C-Scan ultrasonics. C-Scan views are area maps of the inspection surface. They 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, time-of-flight values and signal waveforms.

Extensive testing has shown that the two-dimensional, color coded images produced by manual and automated scanners are able to reliably show flaws on the order of 0.25" 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 thickness.

A UT inspection procedure was developed by the AANC and approved for use on L-1011 aircraft by Lockheed-Martin, Delta Air Lines and the FAA [45]. This procedure was carried out to certify the initial L-1011 doubler installation and to assess the condition of the doubler following 45 days, 6 months, and 1 year of service in Delta's fleet. The sensitivity of the technique to detect 1/4 to 1/2 inch diameter disbonds and delaminations was demonstrated.

Crack detection in the parent aluminum material can be accomplished using conventional eddy current and X-ray techniques. The success of the eddy current technique is primarily dependent upon two installation factors: 1) lift-off effects due to the thickness of the composite doubler, and 2) signal disruption from other conductive medium such as copper mesh lightning protection. Initial testing conducted by the AANC on fatigue crack doubler 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.

The AANC completed 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 doublers (72 ply) [18]. Radiography was demonstrated to be a very effective inspection method to interrogate the interior of the parent material covered by the composite doubler. X-ray inspections are as effective as before a doubler is installed. The required damage detection threshold for cracks under the doubler is 1.0". X-ray images showed the ability to detect fatigue cracks on the order of 0.38" in length beneath 0.40" thick (72 ply) Boron-Epoxy doublers.

Component Level Tests - A C-section test article was used to study doubler delamination and disbond in extreme bending and peel stress conditions. This test specimen was a subsize mock-up of a skin cut-out with a corner reinforced by a composite doubler. Specimen geometry and load conditions were engineered to exceed design strains corresponding to the worst-case flight conditions of fuselage downbending and pressurization.

The test specimen survived the first two load scenarios, up to 15,600 lbs. (150% Limit Load) without failure. The aluminum plate, composite doubler, and aluminum angles all survived strain levels which exceeded the design allowable values. In performing its

reinforcing duties, the doubler absorbed approximately 40% to 50% of the load from the aluminum structure beneath it. Strain levels along the perimeter of the door corner cut-out were much larger than the strains at the center of the doubler. This supports the design assumption and analyses methods 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.

Strain gage results revealed a large plastic region in the corner of the aluminum plate. This plastic zone caused the strain rate in the doubler to increase. Tests to failure demonstrated that the bondline is able to transfer plastic strains into the doubler and that the parent aluminum skin must experience yield strains before any damage to the doubler will occur. Furthermore, composite-reinforced aluminum plates are able to withstand stresses in excess of the Mil Handbook ultimate tensile stress values listed for the aluminum material.

In the ultimate failure test, the stress in the composite doubler reached 92.3 KSI before failure occurred. Thus, the stresses required to fail this doubler were found to be 4.8 times greater than the stresses measured during the full-scale fuselage tests (simulated cabin pressure tests). The parent aluminum material beneath the doubler experienced a maximum stress level of 86 KSI. This means that the composite-reinforced aluminum plate was able to withstand stresses in excess of the Mil Handbook ultimate tensile stress value listed for the parent 7075-T6 aluminum material (83 KSI). Thus, the aluminum stresses measured at doubler failure were 5.6 times greater than the stresses experienced during the full-scale fuselage tests. The C-section test article withstood a maximum load, or failure load, which was 2.5 times the limit load for this composite application. These results show that the safety factor for this doubler design exceeded the minimum goal of 1.5.

General Conclusions Regarding Composite Doubler Technology - 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 ground rules 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 can successfully address both of these concerns.

REFERENCES

1. Jones, K.M., and Shah, S., "Composite Repair - Upper Forward Corner of P-3 Door - Model L-1011 Aircraft, Strength and Damage Tolerance Analysis", Report No. LG95ER0157, Part of Documentation Package for FAA Atlanta Aircraft Certification Office Project No. SP1798AT-Q, analysis plan December 1995, final report October 1996.
2. Baker, A.A., "Bonded Composite Repair of Metallic Aircraft Components", AGARD-CP-550 Composite Repair of Military Aircraft Structures, 1994.
3. Lynch T.P., "Composite Patches Reinforce Aircraft Structures", Design News, April 1991.
4. Sandow, F.A. and R. K. Cannon, "Composite Repair of Cracked Aluminum Alloy Aircraft Structure", Air Force Wright Aeronautical Laboratories Report AD-A190 514, Sept. 1987.
5. Baker, A.A. and Jones, R., Bonded Repair of Aircraft Structures, Martinus Nijhoff Pub., The Netherlands, 1988.
6. Atluri, S.N., Park, J.H., Punch, E.F., O'Donoghue, P.E., Jones, R., "Composite Repairs of Cracked Metallic Aircraft Structures", Dept. of Transportation Report No. DOT/FAA/CT-92/32, May 1993
7. Elvidge, R., "Structures in Service - An Airline Point of View," Canadian Aeronautics and Space Journal, Vol. 30, December 1984
8. Jones, R. and Callinan, R., "A Design Study in Crack Patching," Fibre Science and Technology, Vol. 14, 1981.
9. Bakuckas, Jr, J.G., Chen, C.C., Yu, J., Tan, P.W., and Bigelow, C.A., "Engineering Approach to Damage Tolerance Analysis of Fuselage Skin Repairs," Dept. of Transportation Report No. DOT/FAA/AR-95/75, November 1996.
10. Davis, M., "The Development of an Engineering Standard for Composite Repairs," AGARD-CP-550, Proceedings of AGARD Meeting, Seville, October 1994.
11. Christian Jr., T., Hammond, D., and Cochran, J., "Composite Material Repairs to Metallic Airframe Components," Journal of Aircraft, Vol. 29, 1992.
12. Roach, D., "DC-10 Composite Doubler Repair Status Report," AANC-FAA document, July 1998.
13. Belason, E.B., "Status of Bonded Boron-Epoxy Doublers for Military and Commercial Metallic Aircraft Structures," Airframe Finishing, Maintenance and Repair Conference, SAE Paper 951145, Bellevue, WA, March 1995.
14. Sivam, T.P., Edwards, D., Mason, S., and Guy, P., "The Evolution of Composite Patch Repair Technology at CTAS in Meeting the Challenges of the C141-B Drop-In-Repair Program for WRALC," SAE Airframe Maint. and Repair Conf, SAE Paper No. SAE-961255, August 1995.
15. Baker, A.A., and Chester, R.J., "Recent Advances in Bonded Composite Repair Technology for Metallic Aircraft Components," Proc. Int'l Conf. on Adv. Composite Materials, 1993.

16. Molent, L., Callinan, R.J., and Jones, R., "Design of an All Boron/Epoxy Doubler Reinforcement for the F-111C Wing Pivot Fitting: Structural Aspects," *Composite Structures*, Vol. 11, 1989.
17. Roach, D.P., "Damage Tolerance Assessment of Bonded Composite Doublers for Commercial Aircraft Applications", Sandia National Laboratories/ Dept. of Energy Report No. SAND98-1016, August 1998
18. Roach, D.P., and Walkington, P., "Development and Validation of Nondestructive Inspection Techniques for Composite Doubler Repairs on Commercial Aircraft", Sandia National Laboratories/ Dept. of Energy Report No. SAND98-1014 , May 1998.
19. Roach, D.P., "Results from FAA Program to Validate Bonded Composite Doublers for Commercial Aviation Use," SAE Paper 972622, SAE Airframe Maintenance and Repair Conference, August 1997.
20. Roach, D.P. "Use of Composite Doublers to Repair DC-10 Aircraft Structure," AANC proposal to the FAA, November 1997.
21. Baker, A.A., "Fatigue Studies Related to Certification of Composite Crack Patching for Primary Metallic Aircraft Structure," FAA-NASA Symposium on Continued Airworthiness of Aircraft Structures, Dept. of Transportation Report No. DOT/FAA/AR-97-2,I, July 1997.
22. Fredell, R.S., "Damage Tolerant Repair Techniques for Pressurized Aircraft Fuselages", PhD Dissertation, Delft University, 1994.
23. Rice, R., Francini, R., Rahman, S., Rosenfeld, S., Rust, S., Smith, S., and Broek, D., "Effects of Repair on Structural Integrity", Dept. of Transportation Report No. DOT/FAA/CT-93/79, December 1993.
24. Jones, R., Chiu, C., Paul, J., "Designing for Damage Tolerant Bonded Joints," *Composite Structures*, Vol. 25, 1993
25. Chiu, W.K., Rees, D., Chalkley, P., and Jones, R., "Designing for Damage Tolerant Repairs," ARL Aircraft Structures Report 450, August 1992.
26. Grills, R.H., and Mullis, T., "C-141 Weep Hole External Inspection of Bonded Boron Patches", Air Force 3rd Aging Aircraft Conference, Sept. 1995.
27. Roach, D.P., Moore, D., and Walkington, P., "Nondestructive Inspection of Bonded Composite Doublers for Aircraft", Proceedings of SPIE Conference on Nondestructive Evaluation of Aging Aircraft, December 1996.
28. Gieske, J.H., Roach, D.P., Walkington, P.D., "Ultrasonic Inspection Technique for Composite Doubler/Aluminum Skin Bond Integrity for Aircraft," SPIE Nondestructive Evaluation Techniques for Aging Infrastructure & Manufacturing Conf., vol. 3258, April 1998.
29. Fredell, R.S., van Barnveld, W., and Vlot, A., "Analysis of Composite Crack Patching of Fuselage Structures: High Patch Modulus Isn't the Whole Story", SAMPE Int'l Symposium 39, April 1994.
30. Rose, L.R., "Influence of Disbonding on the Efficiency of Crack Patching", *Theoretical Applied Fracture Mechanics*, vol. 7 , 1987.
31. Baker, A.A., "Growth Characterization of Fatigue Cracks Repaired with Adhesively Bonded Boron/Epoxy Patches," Proc. Of Int'l Conf. Fracture, ICF-9, April 1997.

32. Rose, L.R., "A Cracked Plate Repaired by Bonded Reinforcements," *Int'l Journal of Fracture*, Vol. 18, 1982.
33. Sun, C.T., Klug, J., and Arendt, C., "Analysis of Cracked Aluminum Plates Repaired with Bonded Composite Patches," *AIAA Journal*, Vol. 34, February 1996.
34. Callinan, R.J., Rose, L.R., and Wang, C.H., "Three Dimensional Stress Analysis of Crack Patching," *Proc. Of Int'l Conf. Fracture, ICF-9*, April 1997.
35. Rose, L.R., "Design Analysis and Validation for a Bonded Composite Repair to Primary Aircraft Structure," *Proc. Of Int'l Conf. Fracture, ICF-9*, April 1997.
36. Davis, M.J., "A Call for Minimum Standards in Design and Application Technology for Bonded Structural Repairs", *Int'l Symposium on Composite Repair of Aircraft Structures in concert with ICCM-10*, August 1995.
37. Fredell, R.S., and Marr, J., "An Engineering Approach to the Design and Analysis of Fuselage Crack Patching with the Computer Program CalcuRep for Windows", *Int'l Symposium on Composite Repair of Aircraft Structures in concert with ICCM-10*, August 1995.
38. Xiong, X., Raizenne, D., "A Design Methodology and PC-Based Software for Bonded Composite Repair in Aircraft Structure", *Int'l Symposium on Composite Repair of Aircraft Structures in concert with ICCM-10*, August 1995.
39. Schweinberg, W., Jansen, R., and Fiebig, J., "Advanced Composite Repairs of the C-141 Wing Structure", *Int'l Symposium on Composite Repair of Aircraft Structures in concert with ICCM-10*, August 1995.
40. Klemczyk, C., and Belason, E.B., "Analysis of Maximum Stresses Associated with a Boron/Epoxy Doubler Bonded to Aluminum", *Boeing Report under contract 6-1171-10A3397R4*, January 1994
41. Rutherford, P., Berg, S., Miller, M. and Mazur, C. "Boron Epoxy Field Repair Doubler for Commercial Aircraft," *Boeing Report under contract 6-1171-10A3397R4*, January 1995.
42. Herderich, D., Shah, S., and Izquierdo, I., "Doubler-Composite Reinforcement, P-3 PAX Door, UPR FWD Corner, Composite Reinforcement", *Lockheed-Martin Aeronautical Systems Drawing No. LCC-7622-378*, April 1996.
43. Berg, S.D., "Process Specification for the Fabrication and Application of Boron-Epoxy Doublers onto Aluminum Structures", *Textron Specialty Materials Specification No. 200008-001* (may also be referenced as the Boeing Specification D658-10183-1 which was written for Textron), dated November 30, 1995, *Textron Specialty Materials*, Lowell, MA, 01851.
44. Huang, J., Reeve, S., and Shah, S., "Boron/Epoxy Material Allowables", *Report No. LG95ER0193*, Part of Documentation Package for FAA Atlanta Aircraft Certification Office Project No. SP1798AT-Q, November 1995.
45. Walkington, P., and Roach, D., "Ultrasonic Inspection Procedure for Bonded Boron-Epoxy Composite Doublers," *Sandia Labs AANC Specification AANC-PEUT-Comp-5521/4-004*, *Sandia National Laboratories*, Albuquerque, NM; also included in FAA Document SNL96ER0007 under Atlanta ACO Project SP1798AR-Q, FAA approval January 1997.
46. Roach, D., Swanson, M., and Walkington, P., "Nondestructive Inspection Procedure for Bonded Boron-Epoxy Composite Doublers Using the Eddy Current Sliding Probe

- Technique," Sandia Labs AANC Specification AANC-EC-Comp-5521/4-003, January 1995, Sandia National Laboratories, Albuquerque, NM.
47. Huang, J., Reeve, S., and Shah, S., "Boron/Epoxy Material Allowables for L-1011 Repair", Report No. LG95ER0193, Part of Documentation Package for FAA Atlanta Aircraft Certification Office Project No. SP1798AT-Q, November 1995.
 48. Belason, E.B., "Fatigue and Static Ultimate Tests of Boron-Epoxy Doublers Bonded to 7075-T6 Aluminum with a Simulated Crack", Int. Conf. on Aeronautical Fatigue, Melbourne, Australia, May 1995.
 49. Roach, D.P., "Performance Analysis of Bonded Composite Doublers on Aircraft Structures", International Conf. on Composite Materials, August 1995.
 50. Bullock, L.C., "Airframe Ultimate Load and Pressure Tests - L-1011 Static Test Airplane, S/N 0180," LR 25351, Lockheed California Co., Burbank, CA, December 1972.
 51. Nondestructive Testing Manual," Chapter 53, Lockheed California Co., Burbank, CA

This Page Left Intentionally Blank

Appendix A

Engineering Drawings of Aircraft Door Surround Structure Biaxial Test Facility and Instrumentation Scheme

This Page Left Intentionally Blank

Drawing Number	ANC-4
Revision	1
Classification	UNCLASSIFIED
File	1901-1211
Page	1
Sheet	3 of 3

TABULATION									
GAGE #	ORIENTATION	AXIAL DIMENSION	HOOP DIMENSION	USE/AGE LOCATION	STRAIN GAGE TYPE	CHI	CI-2	CHI-3	
R1	SEE CONFIG B	3.06 ± .25	24.25 ± .25	OUTSIDE	CEA-13-125UT-350	1	2	3	
R2	SEE CONFIG B	5.50 ± .25	14.00 ± .25	OUTSIDE	CEA-13-125UT-350	12	13	14	
R3	SEE CONFIG A	27.38 ± .25	1.30 ± .25	OUTSIDE	CEA-13-125UT-350	15	16	17	
R4	SEE CONFIG A	17.38 ± .25	1.38 ± .25	OUTSIDE	CEA-13-125UT-350	24	25	26	
R5	SEE CONFIG B	7.36 ± .25	1.36 ± .25	INSIDE	CEA-13-125UT-350	27	28	29	
R6	SEE CONFIG A	1.25 ± .25	2.00 ± .25	OUTSIDE	CEA-13-125UT-350	30	31	32	
R7	SEE CONFIG B	1.25 ± .25	2.00 ± .25	INSIDE	CEA-13-125UT-350	33	34	35	
R8	SEE CONFIG B	10.63 ± .25	5.38 ± .25	OUTSIDE	CEA-13-125UT-350	36	37	38	
R9	SEE CONFIG A	0.63 ± .25	5.38 ± .25	INSIDE	CEA-13-125UT-350	39	40	41	
R10	SEE CONFIG B	18.88 ± .25	2.63 ± .25	OUTSIDE	CEA-13-125UT-350	48	49	50	
R11	SEE CONFIG B	22.69 ± .25	8.50 ± .25	OUTSIDE	CEA-13-125UT-350	53	54	55	
R12	SEE CONFIG B	16.00 ± .25	14.00 ± .25	OUTSIDE	CEA-13-125UT-350	56	57	58	
B1	SEE CONFIG C	22.59 ± .25	14.00 ± .25	OUTSIDE	CEA-13-125UT-350	4	5		
B2	SEE CONFIG C	13.00 ± .25	15.50 ± .25	OUTSIDE	CEA-13-125UT-350	6	7		
B3	SEE CONFIG C	5.50 ± .25	15.50 ± .25	OUTSIDE	CEA-13-125UT-350	9	9		
B4	SEE CONFIG C	13.00 ± .25	14.00 ± .25	OUTSIDE	CEA-13-125UT-350	10	11		
B5	SEE CONFIG C	20.50 ± .25	3.50 ± .25	OUTSIDE	CEA-13-125UT-350	18	19		
B6	SEE CONFIG C	19.00 ± .25	3.50 ± .25	OUTSIDE	CEA-13-125UT-350	20	21		
B7	SEE CONFIG C	13.00 ± .25	1.38 ± .25	OUTSIDE	CEA-13-125UT-350	22	23		
B8	SEE CONFIG C	5.50 ± .25	1.38 ± .25	OUTSIDE	CEA-13-125UT-350	42	43		
B9	SEE CONFIG C	11.65 ± .25	7.50 ± .25	OUTSIDE	CEA-13-125UT-350	44	45		
B10	SEE CONFIG C	13.35 ± .25	7.50 ± .25	OUTSIDE	CEA-13-125UT-350	46	47		
B11	SEE CONFIG C	16.00 ± .25	8.50 ± .25	OUTSIDE	CEA-13-125UT-350	51	52		
B12	SEE CONFIG C	0.63 ± .25	14.00 ± .25	OUTSIDE	CEA-13-125UT-350	56	57		
B13	SEE CONFIG C	17.75 ± .25	14.00 ± .25	OUTSIDE	CEA-13-125UT-350	61	62		
B14	SEE CONFIG C	5.50 ± .25	22.85 ± .25	OUTSIDE	CEA-13-125UT-350	63	64		
B15	SEE CONFIG C	5.50 ± .25	24.20 ± .25	OUTSIDE	CEA-13-125UT-350	65	66		
B16	SEE CONFIG C	22.69 ± .25	27.00 ± .25	OUTSIDE	CEA-13-125UT-350	67	68

DRAWING NUMBER: AANC-6
 DRAWING DESCRIPTION: UNCLASSIFIED
 DATE: 10/21/13
 SCALE: NONE
 SHEET: 2 OF 2

NOTES:

1. MATERIAL: STEEL, A56, 1.000 PLATE OR EQUIVALENT. MATERIAL MUST HAVE ULTIMATE STRENGTH OF 80KSI OR BETTER AND A YIELD STRESS OF 36KSI OR BETTER. FOR BASE AND SIDE PLATES. SEE CUT DRAWINGS FOR INTERNAL SUPPORT MATERIAL.
 2. WELDING RODS (MATERIAL) MUST HAVE A FINAL ULTIMATE STRENGTH OF 80KSI AND A YIELD STRENGTH OF 36KSI.
 3. FOLLOW CURRENT ASTM WELDING SPECS FOR ALL WELDS. INSPECTION AND LEAK TESTING NOT REQUIRED.
 4. USE SQUARE TUBING SUPPORTS (A) FOR LIFTING, USING STRAPS.
- ▶ SUPPORTS (A), (B) AND (C) TO BE WELDED AT 45° IN LOCATIONS SHOWN.
- ▶ WELD "ALL AROUND" SQUARE TUBING SUPPORTS WHERE POSSIBLE.
- ▶ INTERNAL CHAMBER DIMENSION.

8. RADIUS ON BOTH AXIAL SIDE PLATES MUST HAVE A CONCENTRICITY OF $\pm .060$.

DELIVERY DATE: OCTOBER 30, 1995

TOTAL WEIGHT APPROX. = 3 TONS

DELIVER TO: SANDIA NATIONAL LABS
AREA: III, BLDG: 6730

QUESTIONS CALL:
TOM RICE, 845-9846
DENNIS ROACH, 844-6078

TITLE	
VACUUM CHAMBER, WIDE BODY AIRCRAFT TEST FACILITY. (L1011)	
DATE	NAME
09/24/95	T. M. RICE, 2741
REV.	NUMBER
B	WBAT1006
SCALE: 1/20	SHEET 1 OF 2

Appendix B

Tabulated Data From Full Scale Structural Tests:

- 1) Aircraft Fuselage Structure Cabin Pressurization Tests**
- 2) Door Corner “C-Section” Tests**
- 3) Four-Point Bending Tests**

This Page Left Intentionally Blank

Pressure	Hoop Load	Axial Load	LCE1	LCW2	LCE3	LCW4	LCE5	LCW6	LCE7	LCW8	LCE9	LCW10	LCE11	LCW12	LCE13	LCW14	LCE15
(psi)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)
0	66.5	2.5	29	0	0	0	15	0	0	0	15	15	0	14	15	15	0
1.5	15090	10189	1062	870	1222	957	1037	800	1149	891	951	931	1039	968	892	858	931
2	19442	13132	1382	1165	1527	1214	1304	1011	1396	1192	1268	1280	1291	1221	1154	1149	1207
2.5	22809	15925	1556	1305	1702	1371	1481	1147	1571	1419	1525	1542	1502	1432	1385	1396	1455
3	25724	18357	1644	1375	1833	1471	1630	1253	1731	1615	1766	1789	1712	1628	1600	1644	1702
3.5	28457	20547	1745	1418	1964	1543	1763	1358	1891	1796	1992	2007	1909	1825	1815	1876	1920
4	31125	23152	1847	1446	2080	1614	1881	1449	2036	1977	2219	2225	2105	2021	2015	2109	2153
4.5	33804	25499	1964	1488	2211	1700	2030	1555	2182	2174	2445	2429	2316	2218	2231	2342	2371
5	36377	27888	2051	1516	2327	1757	2148	1660	2327	2355	2657	2647	2498	2414	2431	2575	2589
5.5	38936	29931	2153	1558	2444	1829	2267	1766	2473	2536	2868	2851	2681	2611	2646	2807	2807
6	41553	32230	2255	1586	2560	1900	2385	1857	2604	2717	3079	3055	2877	2807	2862	3055	3040
6.5	44061	34481	2342	1614	2676	1957	2504	1962	2749	2898	3291	3258	3074	3004	3062	3287	3258
7	46475	36807	2415	1642	2793	2014	2607	2053	2895	3064	3502	3462	3256	3200	3262	3535	3462
7.5	48910	39014	2502	1670	2895	2071	2726	2143	3025	3245	3713	3665	3453	3396	3477	3767	3680
8	51275	41513	2575	1698	2996	2129	2830	2249	3171	3411	3909	3855	3649	3593	3677	3985	3884
8.5	53665	43472	2662	1712	3098	2186	2948	2340	3302	3577	4106	4044	3832	3789	3892	4233	4102
9	56037	45938	2735	1726	3200	2243	3052	2445	3447	3758	4302	4247	4028	3986	4092	4465	4320
9.5	58260	48092	2807	1754	3302	2300	3156	2536	3578	3925	4498	4436	4211	4182	4292	4698	4509
10	60557	50021	2909	1796	3404	2357	3274	2626	3709	4091	4694	4625	4407	4379	4492	4931	4727

Pressure	LCW16	LCE17	LCW18	LCE19	LCW20	LCE21	LCW22	LCE23	LCW24	LCE25	LCW26	LCE27	LCW28	LCE29	LCW30
(psi)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)
0	0	0	0	0	0	0	0	0	0	0	0	15	0	0	0
1.5	859	940	873	875	856	931	916	1047	814	1011	856	989	800	889	1025
2	1126	1193	1120	1177	1109	1251	1207	1353	1095	1249	1123	1236	1081	1096	1263
2.5	1363	1418	1353	1449	1333	1513	1469	1600	1333	1432	1361	1425	1304	1259	1432
3	1600	1628	1585	1706	1544	1745	1716	1818	1544	1600	1558	1585	1467	1378	1558
3.5	1837	1825	1804	1947	1754	1964	1949	2036	1754	1754	1740	1731	1644	1496	1656
4	2059	2035	2022	2189	1965	2196	2182	2240	1951	1909	1923	1862	1807	1615	1754
4.5	2296	2232	2240	2445	2175	2415	2400	2444	2147	2063	2091	2007	1956	1719	1825
5	2548	2428	2444	2687	2386	2633	2633	2647	2344	2204	2260	2138	2104	1807	1909
5.5	2785	2625	2647	2928	2596	2836	2865	2851	2540	2330	2428	2269	2252	1896	1979
6	3022	2835	2865	3170	2821	3069	3098	3040	2737	2484	2611	2400	2400	1985	2063
6.5	3259	3032	3069	3426	3032	3287	3331	3244	2933	2611	2765	2531	2533	2059	2119
7	3496	3228	3273	3653	3242	3491	3564	3433	3116	2751	2919	2647	2652	2133	2175
7.5	3733	3411	3476	3894	3453	3709	3782	3622	3298	2877	3074	2764	2785	2207	2232
8	3956	3593	3680	4136	3663	3913	4000	3811	3467	3004	3228	2865	2919	2267	2288
8.5	4207	3789	3898	4362	3874	4116	4218	3985	3649	3144	3368	2982	3037	2341	2330
9	4444	3986	4102	4604	4070	4335	4436	4189	3818	3256	3509	3098	3156	2400	2372
9.5	4667	4168	4291	4815	4267	4524	4655	4349	3972	3382	3649	3185	3259	2444	2428
10	4889	4365	4480	5042	4463	4727	4858	4509	4126	3495	3789	3287	3378	2489	2470

Table B-1: Test Pressure Versus Resulting Hoop and Axial Loads Generated in Fuselage Test Structure; Before Doubler Installation

Pressure	LCE31	LCW32	LCN1	LCS2	LCN3	LCS4	LCN5	LCS6	LCN7	LCS8	LCN9	LCS10	LCN11	LCS12
(psi)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)
0	0	0	0	20	12	-12	0	-11	0	0	0	0	0	-4
1.5	933	1007	1723	1800	1679	1666	1693	1689	1753	1637	1684	1691	1647	1715
2	1185	1259	2136	2223	2153	2158	2207	2194	2244	2122	2230	2243	2136	2217
2.5	1348	1437	2590	2696	2652	2685	2666	2667	2712	2608	2624	2620	2615	2715
3	1467	1556	2961	3068	3065	3105	3070	3076	3082	2959	3063	3073	3030	3162
3.5	1570	1630	3322	3450	3381	3441	3430	3431	3474	3335	3412	3410	3427	3581
4	1659	1704	3766	3913	3832	3896	3889	3872	3910	3760	3846	3831	3828	3960
4.5	1748	1748	4168	4315	4196	4268	4283	4227	4258	4111	4321	4303	4207	4340
5	1837	1793	4581	4727	4598	4676	4676	4625	4749	4597	4661	4639	4559	4688
5.5	1911	1837	4901	5069	4926	5035	5047	4980	5064	4912	5019	4989	4896	5024
6	1985	1881	5231	5401	5303	5419	5495	5431	5467	5324	5413	5370	5234	5372
6.5	2044	1911	5633	5823	5717	5850	5834	5754	5761	5615	5798	5743	5635	5798
7	2089	1926	6046	6236	6070	6210	6205	6109	6219	6064	6152	6097	5996	6209
7.5	2133	1941	6387	6578	6435	6570	6609	6464	6567	6404	6551	6492	6370	6601
8	2178	1970	6840	7040	6812	6953	6992	6851	7025	6864	6980	6923	6762	6984
8.5	2222	1985	7150	7362	7177	7337	7331	7163	7308	7156	7294	7232	7105	7328
9	2252	2000	7521	7734	7602	7757	7778	7604	7744	7580	7706	7650	7488	7711
9.5	2281	2000	7903	8127	7979	8152	8128	7937	8092	7944	8046	7982	7835	8059
10	2311	2015	8213	8439	8271	8452	8478	8239	8386	8247	8382	8322	8186	8427

Table B-1: Test Pressure Versus Resulting Hoop and Axial Loads Generated in Fuselage Test Structure; Before Doubler Installation (continued)

Press.	Ch. 1	Ch. 2	Ch. 3	Ch. 12	Ch. 13	Ch. 14	Ch. 24	Ch. 25	Ch. 26	Ch. 27	Ch. 28	Ch. 29	Ch. 30	Ch. 31	Ch. 32
(psi)	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$
0	-1	-2	0	0	1	0	1	0	0	1	1	0	1	1	1
1.5	8	77	179	2	138	217	78	83	-17	67	-12	-56	-38	167	177
2.01	-7	104	234	-3	194	266	122	114	-27	109	-8	-81	-46	218	226
2.5	-19	136	284	-5	251	309	170	148	-40	155	-3	-109	-56	269	276
3	-33	168	333	-12	306	349	216	181	-51	200	1	-138	-64	322	328
3.5	-47	200	381	-20	360	389	260	214	-61	242	3	-167	-72	375	380
4	-59	234	430	-24	416	428	307	248	-72	287	7	-196	-79	432	434
4.51	-73	265	480	-30	471	469	352	281	-84	331	9	-225	-85	491	488
5	-84	299	528	-35	525	508	396	314	-94	375	12	-252	-89	552	542
5.5	-98	330	578	-40	579	548	440	346	-104	416	16	-280	-92	616	597
6	-109	363	629	-45	634	587	483	379	-114	458	20	-307	-94	680	651
6.51	-120	396	678	-49	689	628	528	411	-124	502	24	-334	-96	750	707
7.01	-133	427	728	-54	744	667	571	444	-134	544	27	-360	-96	822	761
7.5	-145	458	777	-58	798	706	615	476	-143	584	31	-386	-96	895	817
7.99	-154	490	824	-62	852	743	655	507	-152	626	35	-409	-95	966	869
8.51	-167	521	874	-69	906	781	697	539	-162	667	39	-435	-93	1043	925
9.01	-179	553	923	-73	961	819	743	572	-171	712	45	-460	-90	1123	980
9.49	-191	583	970	-78	1015	855	784	603	-181	753	50	-483	-87	1198	1030
9.99	-205	614	1019	-86	1068	892	824	635	-189	792	56	-505	-83	1275	1080

Press.	Ch. 33	Ch. 34	Ch. 35	Ch. 36	Ch. 37	Ch. 38	Ch. 39	Ch. 40	Ch. 41	Ch. 48	Ch. 49	Ch. 50	Ch. 53	Ch. 54	Ch. 55
(psi)	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$
0	1	1	1	1	0	1	0	0	0	0	0	0	1	0	0
1.5	171	-64	186	-103	0	337	40	-65	-112	73	-31	105	72	63	72
2.01	227	-80	189	-127	-8	432	64	-74	-131	110	-36	131	110	85	89
2.5	280	-98	179	-152	-11	530	81	-82	-144	152	-37	152	157	117	102
3	333	-115	170	-178	-12	631	95	-89	-154	193	-35	170	204	153	115
3.5	381	-131	168	-204	-11	728	108	-98	-164	232	-33	189	250	188	127
4	427	-147	168	-232	-9	830	115	-108	-171	274	-32	207	297	222	137
4.51	472	-162	178	-259	-6	930	116	-121	-177	313	-30	226	341	254	148
5	511	-173	198	-286	-1	1027	114	-137	-182	352	-29	243	385	285	156
5.5	547	-182	224	-313	6	1125	109	-154	-184	388	-27	261	426	314	165
6	580	-188	255	-342	11	1223	102	-176	-183	425	-26	278	468	343	171
6.51	611	-191	288	-371	20	1321	92	-200	-181	463	-24	296	510	372	179
7.01	638	-192	316	-399	27	1420	82	-226	-178	500	-23	313	550	397	185
7.5	663	-190	343	-429	37	1518	72	-249	-174	536	-22	331	588	423	190
7.99	683	-188	363	-457	45	1615	59	-272	-168	569	-20	347	628	447	195
8.51	705	-184	384	-488	56	1716	47	-294	-163	606	-18	365	667	472	200
9.01	726	-181	400	-519	66	1817	31	-312	-153	643	-17	383	705	493	201
9.49	743	-177	409	-550	78	1914	17	-328	-143	678	-15	399	743	516	204
9.99	757	-174	416	-583	90	2015	3	-339	-132	712	-14	416	779	539	208

Table B-2: Strain Data from Fuselage Structure Test Prior to Composite Doubler Installation

Press.	R. 1	R. 1	R. 1	R. 1	R. 1	R. 2	R. 2	R. 2	R. 2	R. 2
(psi)	Max Strain	Min Strain	Max Stress	Min Stress	Pr. Angle	Max Strain	Min Strain	Max Stress	Min Stress	Pr. Angle
0	1	-2	5	-19		1	-1	8	-8	
1.5	180	5	2095	682		221	-3	2538	730	
2.01	235	-8	2678	717	88	280	-18	3165	763	78
2.5	284	-20	3207	749	89	337	-35	3769	762	74
3	334	-35	3729	746	87	395	-60	4351	671	71
3.5	383	-51	4246	733	86	453	-87	4929	569	70
4	435	-66	4786	741	84	513	-111	5536	496	68
4.51	487	-83	5328	731	84	573	-137	6138	399	67
5	537	-96	5867	754	83	633	-162	6739	317	67
5.5	589	-112	6412	742	83	692	-188	7334	227	66
6	643	-126	6977	766	82	753	-214	7944	136	66
6.51	694	-141	7524	780	82	813	-238	8557	70	65
7.01	746	-155	8074	791	82	874	-264	9169	-23	65
7.5	798	-170	8624	802	81	933	-290	9763	-112	64
7.99	847	-182	9146	828	81	991	-315	10346	-204	64
8.51	900	-197	9706	843	81	1052	-344	10944	-325	64
9.01	952	-212	10249	846	81	1111	-371	11539	-430	64
9.49	1001	-227	10767	848	81	1169	-398	12116	-544	63
9.99	1053	-244	11304	830	81	1229	-428	12701	-680	63

Press.	R. 4	R. 4	R. 4	R. 4	R. 4	R. 5	R. 5	R. 5	R. 5	R. 5
(psi)	Max Strain	Min Strain	Max Stress	Min Stress	Pr. Angle	Max Strain	Min Strain	Max Stress	Min Stress	Pr. Angle
0	1	0	12	2		1	0	12	2	
1.5	101	-40	1028	-117		70	-58	601	-433	
2.01	148	-53	1519	-102	21	112	-84	1001	-580	-6
2.5	199	-69	2056	-106	19	158	-112	1436	-749	-6
3	249	-84	2577	-108	18	203	-142	1846	-935	-5
3.5	296	-99	3077	-117	18	245	-171	2241	-1118	-5
4	347	-115	3612	-119	17	291	-200	2663	-1302	-5
4.51	397	-131	4131	-133	17	335	-230	3075	-1490	-4
5	446	-146	4639	-136	17	379	-258	3483	-1660	-4
5.5	493	-159	5137	-129	17	421	-285	3868	-1834	-4
6	541	-174	5637	-139	17	463	-313	4264	-2005	-4
6.51	590	-188	6150	-132	16	507	-341	4671	-2174	-4
7.01	637	-202	6652	-130	16	550	-367	5071	-2335	-4
7.5	685	-216	7151	-123	16	590	-393	5452	-2492	-4
7.99	729	-230	7620	-130	16	632	-417	5851	-2625	-4
8.51	776	-244	8110	-129	16	674	-443	6244	-2779	-4
9.01	825	-257	8629	-114	16	718	-469	6667	-2921	-4
9.49	871	-271	9107	-115	16	760	-492	7069	-3043	-4
9.99	915	-285	9574	-119	16	800	-515	7448	-3169	-4

Table B-3: Principal Stress Data from Fuselage Structure Test Prior to Composite Doubler Installation

Press.	R. 6	R. 6	R. 6	R. 6	R. 6	R. 7	R. 7	R. 7	R. 7	R. 7
(psi)	Max Strain	Min Strain	Max Stress	Min Stress	Pr. Angle	Max Strain	Min Strain	Max Stress	Min Stress	Pr. Angle
0	1	1	14	14		1	1	14	14	
1.5	214	-76	2210	-134		422	-67	4638	693	
2.01	277	-98	2856	-177	68	497	-83	5443	757	-43
2.5	340	-121	3503	-220	68	561	-105	6113	732	-41
3	405	-143	4176	-248	68	628	-127	6801	704	-39
3.5	470	-164	4856	-269	68	694	-149	7490	688	-38
4	539	-186	5573	-285	68	761	-169	8199	680	-37
4.51	609	-208	6310	-293	67	833	-188	8965	714	-37
5	680	-230	7048	-300	67	905	-201	9748	814	-37
5.5	753	-252	7817	-299	67	975	-210	10530	958	-37
6	827	-273	8595	-291	66	1045	-215	11311	1131	-38
6.51	905	-298	9408	-304	66	1110	-217	12060	1336	-38
7.01	983	-322	10230	-312	66	1166	-218	12699	1525	-38
7.5	1064	-347	11070	-325	65	1214	-215	13262	1719	-39
7.99	1140	-371	11870	-340	65	1253	-213	13720	1879	-39
8.51	1224	-397	12746	-346	65	1291	-209	14174	2056	-39
9.01	1309	-425	13630	-374	64	1324	-206	14568	2209	-39
9.49	1388	-451	14454	-399	64	1347	-203	14845	2324	-39
9.99	1469	-478	15299	-430	63	1366	-201	15066	2412	-39

Press.	R. 8	R. 8	R. 8	R. 8	R. 8	R. 9	R. 9	R. 9	R. 9	R. 9
(psi)	Max Strain	Min Strain	Max Stress	Min Stress	Pr. Angle	Max Strain	Min Strain	Max Stress	Min Stress	Pr. Angle
0	2	0	22	6		0	0	0	0	
1.5	366	-134	3766	-273		45	-117	117	-1197	
2.01	475	-172	4888	-343	-75	73	-139	359	-1354	-11
2.5	585	-209	6025	-386	-75	92	-155	529	-1469	-12
3	696	-246	7180	-433	-75	109	-168	679	-1563	-13
3.5	803	-282	8291	-477	-75	126	-181	827	-1654	-14
4	914	-319	9441	-519	-75	136	-193	908	-1750	-15
4.51	1021	-355	10558	-557	-75	142	-203	940	-1852	-16
5	1125	-389	11636	-596	-75	147	-215	958	-1968	-18
5.5	1229	-423	12719	-627	-75	151	-225	966	-2074	-19
6	1333	-458	13800	-670	-76	157	-237	989	-2195	-22
6.51	1437	-493	14877	-709	-76	164	-252	1023	-2341	-24
7.01	1542	-527	15963	-743	-76	173	-269	1070	-2501	-27
7.5	1643	-562	17020	-790	-76	184	-285	1138	-2653	-29
7.99	1745	-595	18079	-825	-76	192	-301	1177	-2804	-31
8.51	1850	-630	19160	-869	-77	202	-317	1230	-2955	-33
9.01	1956	-666	20267	-909	-77	208	-330	1259	-3082	-35
9.49	2056	-700	21297	-957	-77	215	-340	1301	-3180	-37
9.99	2159	-736	22366	-1016	-77	220	-348	1330	-3252	-38

Table B-3: Principal Stress Data from Fuselage Structure Test Prior to Composite Doubler Installation (continued)

Press.	R. 10	R. 10	R. 10	R. 10	R. 10	R. 11	R. 11	R. 11	R. 11	R. 11
(psi)	Max Strain	Min Strain	Max Stress	Min Stress	Pr. Angle	Max Strain	Min Strain	Max Stress	Min Stress	Pr. Angle
0	0	0	0	0		1	0	12	2	
1.5	211	-33	2319	346		81	62	1146	1000	
2.01	277	-38	3067	524	-47	118	81	1636	1338	-28
2.5	341	-39	3798	733	-45	159	98	2177	1681	-12
3	398	-37	4467	947	-44	204	113	2744	2012	-4
3.5	455	-37	5129	1155	-42	249	126	3312	2313	0
4	515	-37	5813	1355	-42	296	136	3890	2591	2
4.51	572	-37	6479	1559	-41	340	145	4426	2854	3
5	628	-37	7121	1744	-40	385	153	4967	3099	4
5.5	681	-37	7733	1932	-40	426	161	5476	3334	4
6	736	-38	8366	2113	-39	468	167	5978	3547	4
6.51	791	-38	9001	2305	-39	511	174	6494	3774	5
7.01	845	-38	9626	2494	-39	550	180	6973	3983	5
7.5	899	-38	10247	2673	-39	589	184	7436	4165	5
7.99	949	-38	10817	2846	-38	629	189	7908	4352	5
8.51	1003	-38	11442	3034	-38	668	193	8380	4539	5
9.01	1058	-39	12075	3215	-38	706	195	8820	4688	4
9.49	1109	-39	12657	3390	-38	744	197	9264	4848	4
9.99	1160	-39	13246	3559	-38	781	200	9702	5013	4

Table B-3: Principal Stress Data from Fuselage Structure Test Prior to Composite Doubler Installation (continued)

Pressure	Hoop Load	Axial Load	LCE1	LCW2	LCE3	LCW4	LCE5	LCW6	LCE7	LCW8	LCE9	LCW10	LCE11	LCW12	LCE13	LCW14	LCE15
(psi)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)
0	7	-11	0	-14	15	0	0	0	0	0	0	0	0	0	-15	0	0
1.5	16851	8491	1105	954	1047	1000	1156	1147	1091	1011	966	1004	1109	1025	1015	1062	1062
2	20493	10450	1411	1249	1338	1229	1407	1374	1309	1253	1238	1265	1305	1235	1215	1295	1265
2.5	24920	12107	1745	1600	1687	1500	1733	1645	1571	1555	1555	1585	1572	1488	1477	1585	1542
3	28956	14600	2036	1909	1993	1743	2044	1887	1804	1842	1857	1876	1825	1712	1723	1847	1789
3.5	33188	16241	2284	2204	2255	1986	2311	2158	2036	2143	2174	2167	2119	1951	2000	2124	2080
4	36693	18018	2473	2456	2473	2186	2519	2370	2240	2400	2460	2400	2344	2161	2246	2356	2327
4.5	40185	20355	2662	2681	2676	2386	2726	2596	2429	2672	2717	2647	2582	2358	2508	2589	2589
5	43385	21758	2822	2863	2851	2557	2904	2777	2604	2898	2974	2865	2793	2568	2754	2807	2822
5.5	46468	23916	2953	2975	2996	2671	3067	2943	2778	3109	3215	3084	3018	2765	2985	3040	3055
6	49122	25734	3040	3004	3113	2729	3185	3064	2924	3291	3426	3287	3214	2961	3215	3273	3273
6.5	51843	27732	3113	3018	3215	2800	3319	3185	3069	3487	3653	3491	3425	3158	3446	3520	3491
7	54360	29404	3171	3046	3302	2857	3437	3306	3200	3668	3864	3695	3621	3340	3677	3753	3709
7.5	56767	31295	3229	3060	3389	2914	3541	3426	3345	3834	4060	3884	3818	3551	3892	3985	3927
8	59291	33147	3302	3074	3491	2957	3659	3532	3476	4030	4287	4087	4014	3761	4108	4247	4145
8.5	61641	34793	3375	3074	3564	3000	3763	3653	3607	4196	4498	4291	4211	3972	4338	4495	4349
9	63896	36776	3447	3074	3651	3043	3881	3758	3738	4362	4694	4480	4393	4168	4554	4727	4553
9.5	66224	38437	3520	3088	3738	3086	3985	3864	3869	4528	4906	4684	4589	4379	4769	4975	4771
10	68368	40308	3578	3102	3825	3129	4074	3955	3985	4679	5087	4858	4772	4575	4969	5207	4975

Table B-4: Test Pressure Versus Resulting Hoop and Axial Loads Generated in Test Structure; After Doubler Installation

Pressure	LCW16	LCE17	LCW18	LCE19	LCW20	LCE21	LCW22	LCE23	LCW24	LCE25	LCW26	LCE27	LCW28	LCE29	LCW30
(psi)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)
0	0	0	15	0	0	0	0	0	0	14	0	0	0	0	0
1.5	1022	1081	1033	996	1039	945	1033	945	968	1039	996	1091	1067	1156	1137
2	1230	1263	1222	1192	1249	1149	1280	1135	1179	1221	1207	1309	1304	1393	1347
2.5	1481	1502	1455	1464	1516	1411	1556	1382	1446	1432	1474	1556	1585	1644	1600
3	1719	1740	1673	1736	1740	1658	1818	1600	1684	1628	1712	1789	1852	1852	1825
3.5	1970	1993	1905	2038	1993	1935	2095	1847	1951	1853	1965	2051	2133	2074	2063
4	2193	2218	2080	2309	2189	2167	2313	2065	2161	2035	2175	2255	2356	2237	2274
4.5	2400	2442	2269	2596	2400	2429	2545	2298	2372	2218	2386	2444	2593	2400	2456
5	2622	2653	2458	2868	2596	2662	2764	2516	2582	2386	2582	2633	2800	2533	2625
5.5	2844	2849	2662	3125	2807	2909	2996	2735	2807	2554	2793	2793	3007	2667	2793
6	3067	3046	2851	3366	3032	3127	3215	2924	3004	2695	2975	2938	3185	2770	2919
6.5	3319	3242	3055	3623	3256	3360	3462	3127	3214	2849	3158	3084	3348	2859	3032
7	3541	3439	3258	3864	3467	3578	3680	3316	3396	2989	3312	3215	3496	2948	3116
7.5	3763	3621	3462	4091	3677	3796	3898	3491	3579	3130	3467	3331	3630	3022	3172
8	4015	3818	3680	4332	3902	4015	4145	3680	3775	3256	3621	3447	3778	3096	3228
8.5	4252	4000	3884	4558	4126	4233	4364	3855	3944	3382	3761	3564	3881	3170	3256
9	4489	4182	4102	4785	4323	4436	4582	4015	4112	3509	3902	3665	3970	3230	3284
9.5	4726	4379	4320	5026	4533	4640	4800	4189	4281	3635	4028	3767	4059	3289	3312
10	4948	4561	4524	5223	4730	4844	5004	4364	4435	3747	4154	3869	4148	3348	3326

Table B-4. Test Pressure Versus Resulting Hoop and Axial Loads Generated in Test Structure; After Doubler Installation (continued)

Pressure	LCE31	LCW32	LCN1	LCS2	LCN3	LCS4	LCN5	LCS6	LCN7	LCS8	LCN9	LCS10	LCN11	LCS12
(psi)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)	(lb)
0	0	0	-10	0	0	-12	0	0	0	0	0	0	0	0
1.5	1200	1200	1331	1357	1416	1474	1398	1354	1513	1334	1428	1445	1415	1518
2	1467	1452	1692	1759	1679	1761	1693	1730	1840	1637	1840	1844	1666	1759
2.5	1748	1748	1950	2051	1992	2061	1965	2106	2145	1910	1992	1956	1994	2091
3	1985	2015	2362	2483	2412	2516	2304	2428	2645	2365	2413	2409	2396	2466
3.5	2237	2281	2641	2735	2646	2708	2632	2869	2852	2547	2727	2696	2683	2747
4	2444	2504	2930	3046	2968	3079	2905	3084	3190	2850	3000	3010	2961	3012
4.5	2607	2696	3343	3489	3330	3391	3243	3524	3505	3214	3416	3387	3422	3447
5	2756	2874	3559	3660	3515	3642	3549	3814	3788	3487	3627	3589	3637	3648
5.5	2904	3037	3951	4102	4003	4146	3734	4136	4093	3699	4012	3971	3988	3996
6	2993	3141	4230	4373	4208	4349	4062	4426	4409	4124	4348	4334	4317	4289
6.5	3096	3215	4539	4725	4530	4685	4499	4856	4801	4518	4594	4563	4586	4569
7	3185	3274	4807	4977	4784	4936	4783	5211	5040	4730	4871	4863	4924	4881
7.5	3259	3289	5117	5298	5116	5284	4990	5469	5410	5094	5207	5213	5220	5170
8	3319	3304	5488	5691	5399	5607	5274	5769	5715	5397	5485	5518	5508	5443
8.5	3378	3289	5767	5952	5702	5919	5427	5909	5944	5670	5834	5877	5813	5771
9	3422	3259	5984	6183	5995	6242	5962	6403	6368	6064	6071	6106	6110	6063
9.5	3467	3244	6334	6545	6308	6554	6169	6661	6521	6216	6362	6407	6425	6372
10	3511	3230	6664	6867	6659	6889	6399	6919	6847	6489	6698	6770	6731	6684

Table B-4. Test Pressure Versus Resulting Hoop and Axial Loads Generated in Test Structure; After Doubler Installation (continued)

Press.	Ch. 1	Ch. 2	Ch. 3	Ch. 4	Ch. 5	Ch. 6	Ch. 7	Ch. 8	Ch. 9	Ch. 10	Ch. 11	Ch. 12	Ch. 13	Ch. 14	Ch. 15
(psi)	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue
0	0	0	0	0	1	1	1	1	1	0	1	1	1	1	1
1.5	15	56	155	43	149	53	108	-31	289	35	87	-13	18	146	33
2	-2	73	191	50	178	51	137	-52	336	33	108	-28	27	175	52
2.5	-22	96	238	59	209	53	175	-79	399	34	136	-48	36	208	71
3	-36	120	280	74	234	58	207	-98	453	38	159	-62	48	237	94
3.5	-54	146	328	83	266	63	241	-119	513	41	184	-78	61	268	120
4	-71	173	374	97	288	69	273	-136	567	44	207	-92	75	294	149
4.5	-85	202	418	116	304	77	304	-147	617	50	230	-103	91	318	180
5	-102	230	466	127	319	81	336	-161	669	54	253	-115	106	344	211
5.5	-115	261	512	149	319	90	366	-169	717	60	277	-124	125	366	245
6	-131	290	557	161	319	96	397	-177	766	64	300	-132	143	391	277
6.5	-146	321	603	177	311	104	428	-182	814	70	324	-140	164	415	315
7	-162	353	649	190	304	110	458	-189	862	76	348	-148	183	438	346
7.5	-176	383	692	205	293	118	486	-193	907	81	368	-155	203	459	380
8	-190	417	740	220	281	126	517	-198	955	87	393	-162	224	484	414
8.5	-206	449	786	233	270	133	547	-203	1003	93	414	-168	244	507	445
9	-221	479	831	245	265	141	575	-208	1048	98	436	-175	263	528	479
9.5	-236	510	878	259	260	149	604	-213	1096	105	458	-181	284	551	511
10	-249	540	922	275	256	157	631	-217	1141	112	479	-187	303	573	542

Press.	Ch. 16	Ch. 17	Ch. 18	Ch. 19	Ch. 20	Ch. 21	Ch. 22	Ch. 23	Ch. 24	Ch. 25	Ch. 26	Ch. 27	Ch. 28	Ch. 29	Ch. 30
(psi)	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue
0	0	0	0	0	1	0	0	0	0	0	1	0	0	0	0
1.5	-61	-63	91	-53	54	-60	22	-7	33	48	-9	11	-18	-3	-20
2	-61	-73	121	-62	69	-69	37	-11	57	64	-13	37	-17	-17	-21
2.5	-64	-85	155	-70	87	-81	51	-15	84	80	-20	64	-15	-34	-21
3	-65	-98	191	-80	106	-93	66	-19	110	96	-26	93	-12	-51	-20
3.5	-65	-111	231	-90	127	-106	83	-22	137	112	-33	122	-10	-69	-19
4	-64	-123	273	-99	149	-118	102	-26	165	127	-41	152	-6	-87	-16
4.5	-61	-136	319	-108	173	-132	119	-31	193	142	-51	183	-3	-105	-13
5	-59	-149	363	-118	195	-144	136	-35	221	157	-59	212	0	-122	-9
5.5	-56	-163	413	-129	221	-158	156	-42	250	171	-70	244	4	-141	-6
6	-54	-177	459	-138	246	-170	176	-47	278	187	-79	272	6	-158	-2
6.5	-49	-191	514	-150	275	-184	197	-54	311	203	-90	305	10	-177	4
7	-46	-205	561	-160	300	-197	216	-60	339	218	-100	334	13	-193	8
7.5	-43	-217	610	-170	326	-211	236	-67	367	232	-113	363	17	-210	13
8	-41	-231	661	-180	353	-225	256	-73	397	248	-124	393	20	-228	19
8.5	-39	-244	708	-191	378	-238	274	-80	424	261	-135	422	23	-244	23
9	-35	-257	758	-201	404	-251	293	-88	453	275	-148	453	28	-261	29
9.5	-33	-270	806	-211	430	-265	312	-94	482	289	-161	482	31	-278	35
10	-30	-284	853	-221	454	-279	331	-101	509	304	-170	509	35	-293	40

Table B-5: Strain Data from Fuselage Structure Test After the Composite Doubler Installation

Press.	Ch. 31	Ch. 32	Ch. 33	Ch. 34	Ch. 35	Ch. 36	Ch. 37	Ch. 38	Ch. 39	Ch. 40	Ch. 41	Ch. 42	Ch. 43	Ch. 44	Ch. 45
(psi)	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue
0	0	0	1	0	1	0	0	0	0	-1	1	1	1	1	1
1.5	145	132	78	-44	170	-85	102	255	-23	36	94	-22	46	-13	33
2	185	162	100	-51	183	-102	134	323	-32	48	121	-18	51	-16	34
2.5	236	201	128	-57	200	-127	175	410	-41	60	154	-14	58	-21	36
3	283	236	152	-63	220	-149	212	491	-51	72	184	-8	64	-25	36
3.5	334	273	177	-67	246	-173	252	578	-59	84	216	-4	69	-29	37
4	383	307	201	-71	269	-194	289	658	-67	94	245	3	73	-31	36
4.5	435	341	222	-74	293	-217	326	737	-75	104	273	10	77	-31	32
5	485	375	241	-77	314	-239	362	817	-82	114	301	18	81	-30	30
5.5	539	408	260	-80	333	-262	397	898	-89	122	329	26	83	-24	23
6	589	441	277	-82	349	-284	431	977	-96	129	355	35	86	-19	17
6.5	646	475	294	-83	365	-307	468	1060	-102	137	381	46	88	-9	8
7	697	508	309	-85	379	-330	502	1139	-108	144	407	55	91	-2	1
7.5	750	539	323	-86	392	-352	536	1218	-114	151	430	64	93	7	-8
8	806	573	336	-86	406	-376	571	1300	-120	157	455	76	95	16	-17
8.5	857	605	348	-87	417	-398	605	1382	-125	163	479	85	98	24	-26
9	911	635	360	-87	429	-421	638	1458	-131	169	502	96	99	34	-35
9.5	966	668	371	-87	440	-444	673	1542	-135	176	527	107	102	43	-44
10	1017	697	380	-86	448	-466	706	1618	-141	182	550	118	105	51	-53

Press.	Ch. 46	Ch. 47	Ch. 48	Ch. 49	Ch. 50	Ch. 51	Ch. 52	Ch. 53	Ch. 54	Ch. 55	Ch. 56	Ch. 57	Ch. 58	Ch. 59	Ch. 60
(psi)	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue
0	0	0	0	0	0	0	0	-1	0	0	0	1	1	0	1
1.5	-11	82	-2	-45	19	-50	52	61	90	106	-50	129	-63	-18	36
2	-18	97	8	-59	23	-63	66	96	124	126	-58	155	-77	-22	47
2.5	-29	117	22	-75	27	-78	82	140	168	151	-67	189	-95	-27	62
3	-34	132	35	-89	31	-91	98	179	204	173	-76	219	-111	-32	75
3.5	-41	152	49	-103	35	-105	115	222	244	195	-86	254	-129	-36	90
4	-45	167	63	-115	38	-118	129	263	277	212	-94	283	-145	-41	103
4.5	-45	180	78	-128	40	-129	144	310	310	226	-104	314	-160	-44	115
5	-45	194	93	-139	42	-140	158	355	343	238	-113	345	-175	-47	129
5.5	-39	204	108	-150	44	-151	172	403	377	247	-122	377	-190	-49	141
6	-34	214	123	-158	45	-162	185	449	410	254	-132	408	-204	-51	153
6.5	-23	220	141	-167	46	-172	200	500	442	259	-141	440	-219	-53	166
7	-16	228	158	-175	47	-180	213	546	475	264	-150	471	-233	-53	178
7.5	-6	234	174	-183	47	-190	226	592	504	265	-160	503	-247	-54	190
8	5	242	192	-190	48	-199	240	640	536	267	-169	536	-261	-54	204
8.5	12	249	209	-198	49	-207	254	686	569	269	-179	569	-275	-53	216
9	24	254	228	-205	48	-214	266	731	594	266	-188	598	-287	-53	227
9.5	35	259	246	-213	49	-222	280	777	624	268	-197	631	-301	-53	241
10	45	264	264	-220	49	-229	292	820	652	267	-207	663	-314	-52	254

Table B-5: Strain Data from Fuselage Structure Test After the Composite Doubler Installation (continued)

Press.	Ch. 61	Ch. 62	Ch. 63	Ch. 64	Ch. 65	Ch. 66	Ch. 67	Ch. 68	Ch. 69	Ch. 70	Ch. 71	Ch. 72	Ch. 73	Ch. 74
(psi)	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue	ue
0	1	0	0	0	0	1	0	0	-1	0	0	0	0	0
1.5	-3	64	-39	67	-36	246	-10	30	57	-13	-72	221	24	98
2	6	85	-49	83	-43	297	5	29	99	-24	-88	279	39	119
2.5	15	112	-62	104	-54	365	27	28	150	-38	-109	356	59	147
3	25	136	-74	123	-64	426	49	27	200	-53	-129	428	79	173
3.5	35	163	-88	144	-74	494	77	26	252	-68	-151	506	99	201
4	45	187	-100	162	-84	552	102	28	304	-82	-170	575	119	225
4.5	58	210	-113	180	-94	612	130	30	358	-98	-191	647	139	248
5	69	234	-125	199	-104	673	158	33	410	-113	-211	718	159	271
5.5	79	258	-138	218	-114	735	187	38	468	-130	-232	791	179	294
6	91	280	-149	236	-123	795	217	43	521	-144	-253	861	197	314
6.5	105	303	-162	256	-133	857	250	47	579	-162	-275	936	220	336
7	116	325	-173	274	-142	919	284	51	633	-177	-296	1006	238	356
7.5	127	346	-186	292	-152	979	318	52	687	-193	-318	1077	257	376
8	138	369	-198	311	-162	1043	354	52	743	-209	-341	1151	277	397
8.5	149	391	-209	331	-172	1106	389	54	795	-224	-364	1224	295	417
9	164	411	-221	348	-181	1163	421	54	850	-240	-385	1295	316	436
9.5	175	433	-233	367	-192	1227	457	55	907	-256	-409	1370	336	456
10	186	453	-244	385	-201	1285	490	56	957	-271	-431	1439	353	474

Table B-5: Strain Data from Fuselage Structure Test After the Composite Doubler Installation (continued)

Press.	R. 1	R. 1	R. 1	R. 1	R. 1	R. 2	R. 2	R. 2	R. 2	R. 2
(psi)	Max Strain	Min Strain	Max Stress	Min Stress	Pr. Angle	Max Strain	Min Strain	Max Stress	Min Stress	Pr. Angle
0	0	0	0	0		1	1	16	16	
1.5	161	9	1885	664		160	-27	1981	279	
2	193	-4	2212	619	85	185	-38	2259	224	78
2.5	238	-22	2671	569	85	215	-55	2585	121	81
3	280	-36	3105	557	85	242	-67	2889	71	83
3.5	328	-55	3594	504	89	271	-81	3215	1	84
4	375	-72	4073	462	87	296	-94	3488	-67	86
4.5	420	-87	4549	451	86	319	-104	3752	-107	88
5	470	-106	5052	398	85	344	-116	4031	-163	89
5.5	518	-121	5557	401	84	366	-124	4291	-184	90
6	565	-140	6039	341	84	391	-133	4579	-202	88
6.5	614	-158	6541	304	83	416	-141	4875	-211	87
7	663	-176	7044	266	82	441	-151	5160	-242	86
7.5	710	-193	7520	227	82	463	-159	5419	-262	85
8	761	-211	8051	203	82	490	-168	5730	-271	84
8.5	811	-230	8557	147	81	515	-176	6027	-283	84
9	859	-248	9049	106	81	538	-185	6294	-310	83
9.5	909	-267	9566	68	81	564	-194	6597	-325	82
10	957	-283	10056	42	80	588	-202	6879	-339	82

Press.	R. 3	R. 3	R. 3	R. 3	R. 3	R. 4	R. 4	R. 4	R. 4	R. 4
(psi)	Max Strain	Min Strain	Max Stress	Min Stress	Pr. Angle	Max Strain	Min Strain	Max Stress	Min Stress	Pr. Angle
0	1	0	12	2		1	0	14	2	
1.5	51	-81	311	-761		54	-30	582	-184	
2	69	-91	485	-809	-19	77	-32	872	-124	25
2.5	90	-104	676	-887	-18	103	-39	1188	-105	22
3	113	-116	897	-953	-17	129	-45	1503	-87	19
3.5	139	-130	1159	-1018	-15	156	-52	1831	-64	18
4	170	-143	1460	-1066	-15	184	-60	2166	-66	16
4.5	201	-157	1774	-1112	-14	213	-70	2498	-79	15
5	233	-171	2093	-1163	-13	240	-79	2823	-86	14
5.5	267	-185	2439	-1214	-13	269	-90	3158	-119	13
6	300	-199	2768	-1261	-12	298	-99	3498	-124	13
6.5	338	-214	3158	-1299	-12	331	-111	3887	-147	12
7	370	-229	3482	-1355	-12	360	-122	4219	-177	12
7.5	405	-242	3840	-1389	-11	389	-135	4545	-233	12
8	440	-257	4189	-1442	-11	420	-147	4904	-272	12
8.5	472	-271	4509	-1495	-11	448	-159	5218	-316	11
9	507	-285	4860	-1536	-11	477	-173	5543	-385	11
9.5	540	-299	5192	-1587	-11	506	-185	5878	-435	11
10	572	-313	5510	-1637	-11	535	-196	6205	-460	11

**Table B-6: Principal Stress Data from Fuselage Structure Test
After the Composite Doubler Installation**

Press.	R. 5	R. 5	R. 5	R. 5	R. 5	R. 6	R. 6	R. 6	R. 6	R. 6
(psi)	Max Strain	Min Strain	Max Stress	Min Stress	Pr. Angle	Max Strain	Min Strain	Max Stress	Min Stress	Pr. Angle
0	0	0	0	0		0	0	0	0	
1.5	28	-20	255	-129		173	-61	2022	-112	
2	48	-28	452	-156	-22	217	-75	2535	-132	64
2.5	72	-42	687	-237	-16	273	-93	3197	-141	64
3	100	-58	956	-322	-12	324	-108	3806	-145	63
3.5	129	-75	1227	-424	-11	381	-126	4470	-157	63
4	158	-93	1503	-531	-9	433	-142	5091	-157	62
4.5	189	-111	1796	-627	-8	488	-160	5734	-179	62
5	218	-128	2074	-722	-8	541	-176	6364	-174	61
5.5	250	-147	2377	-828	-7	597	-194	7025	-198	61
6	278	-164	2644	-925	-7	651	-211	7658	-209	60
6.5	311	-183	2959	-1029	-6	709	-230	8345	-228	60
7	340	-200	3237	-1125	-6	764	-247	8988	-235	60
7.5	370	-217	3515	-1219	-6	818	-266	9626	-269	60
8	400	-234	3800	-1321	-6	876	-284	10309	-281	59
8.5	428	-251	4071	-1409	-6	930	-302	10946	-298	59
9	459	-267	4370	-1497	-5	985	-321	11589	-337	59
9.5	488	-284	4648	-1592	-5	1043	-341	12266	-360	59
10	515	-300	4911	-1672	-5	1095	-358	12881	-388	58

Press.	R. 7	R. 7	R. 7	R. 7	R. 7	R. 8	R. 8	R. 8	R. 8	R. 8
(psi)	Max Strain	Min Strain	Max Stress	Min Stress	Pr. Angle	Max Strain	Min Strain	Max Stress	Min Stress	Pr. Angle
0	2	0	22	7		0	0	0	0	
1.5	298	-51	3268	450		256	-85	3007	-110	
2	339	-55	3718	536	-51	324	-104	3823	-82	87
2.5	388	-60	4267	648	-50	412	-129	4873	-66	87
3	437	-65	4816	761	-49	494	-152	5843	-50	86
3.5	492	-69	5440	912	-49	582	-176	6895	-19	86
4	543	-73	6018	1038	-48	662	-198	7857	6	86
4.5	590	-76	6550	1169	-48	742	-221	8804	14	86
5	633	-79	7037	1287	-48	822	-245	9761	27	86
5.5	676	-82	7511	1391	-48	903	-267	10728	46	86
6	710	-84	7904	1491	-48	982	-289	11675	70	86
6.5	744	-85	8290	1598	-47	1066	-313	12677	87	86
7	775	-86	8640	1684	-47	1145	-336	13620	99	86
7.5	802	-87	8949	1769	-47	1225	-359	14562	111	86
8	829	-87	9264	1863	-47	1307	-383	15551	125	86
8.5	853	-88	9541	1938	-47	1389	-405	16530	150	86
9	877	-88	9812	2019	-47	1466	-428	17437	150	86
9.5	899	-88	10072	2097	-47	1550	-452	18436	168	86
10	915	-87	10259	2164	-47	1626	-474	19346	182	86

**Table B-6: Principal Stress Data from Fuselage Structure Test
After the Composite Doubler Installation (continued)**

Press.	R. 9	R. 9	R. 9	R. 9	R. 9	R. 10	R. 10	R. 10	R. 10	R. 10
(psi)	Max Strain	Min Strain	Max Stress	Min Stress	Pr. Angle	Max Strain	Min Strain	Max Stress	Min Stress	Pr. Angle
0	2	-1	19	-5		0	0	0	0	
1.5	94	-23	1002	54		63	-46	643	-356	
2	121	-32	1288	50	-89	90	-59	947	-421	-48
2.5	154	-41	1634	56	-89	124	-75	1320	-493	-46
3	184	-51	1949	51	-89	154	-89	1667	-553	-45
3.5	216	-59	2288	64	-89	187	-104	2035	-619	-44
4	245	-67	2598	78	-89	216	-116	2368	-666	-43
4.5	273	-75	2893	78	-89	248	-129	2723	-718	-42
5	301	-82	3195	100	-89	276	-141	3050	-758	-42
5.5	329	-89	3483	108	-90	304	-152	3370	-791	-41
6	355	-96	3764	124	-90	329	-161	3661	-812	-40
6.5	381	-102	4045	139	-90	358	-171	4001	-834	-40
7	407	-108	4318	161	-89	385	-180	4312	-843	-39
7.5	430	-114	4570	177	-89	410	-190	4608	-868	-39
8	456	-120	4840	188	-89	439	-199	4947	-872	-38
8.5	479	-125	5096	214	-89	466	-207	5260	-883	-38
9	503	-131	5348	229	-88	493	-217	5578	-898	-38
9.5	527	-136	5615	258	-88	521	-227	5912	-915	-37
10	551	-142	5867	274	-88	548	-235	6221	-921	-37

Press.	R. 11	R. 11	R. 11	R. 11	R. 11	R. 12	R. 12	R. 12	R. 12	R. 12
(psi)	Max Strain	Min Strain	Max Stress	Min Stress	Pr. Angle	Max Strain	Min Strain	Max Stress	Min Stress	Pr. Angle
0	0	-1	-2	-12		2	0	25	8	
1.5	107	60	1442	1065		36	-63	221	-683	
2	131	91	1822	1502	70	47	-77	315	-824	-87
2.5	169	122	2371	1995	52	63	-96	443	-1000	-86
3	204	148	2869	2412	42	76	-112	556	-1161	-86
3.5	246	171	3432	2822	35	91	-130	684	-1337	-86
4	285	191	3943	3184	29	105	-146	796	-1496	-85
4.5	328	209	4501	3541	23	117	-161	898	-1646	-85
5	371	221	5048	3839	19	131	-177	1011	-1791	-85
5.5	419	231	5630	4116	17	143	-191	1111	-1939	-86
6	465	238	6191	4358	16	155	-206	1215	-2075	-86
6.5	515	244	6787	4594	14	168	-221	1328	-2219	-86
7	563	248	7350	4805	13	180	-234	1429	-2353	-86
7.5	609	248	7888	4972	12	191	-248	1520	-2491	-87
8	658	249	8452	5154	12	205	-262	1648	-2618	-87
8.5	706	249	9004	5320	12	217	-276	1749	-2753	-87
9	750	247	9511	5446	11	228	-288	1848	-2867	-87
9.5	797	248	10056	5620	11	242	-302	1976	-2995	-88
10	840	246	10549	5747	11	254	-314	2086	-3105	-88

**Table B-6: Principal Stress Data from Fuselage Structure Test
After the Composite Doubler Installation (continued)**

Load	Ch. 1	Ch. 2	Ch. 3	Ch. 4	Ch. 5	Ch. 6	Ch. 7	Ch. 8	Ch. 9	Ch. 10
(lb)	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$
2	1	2	2	3	1	3	2	1	2	1
483	82	-10	109	-17	193	18	221	254	81	100
996	177	-12	223	-34	395	39	453	516	166	205
1499	279	-3	334	-44	594	65	681	773	251	307
1990	381	12	443	-52	787	97	904	1018	339	407
2505	493	31	558	-56	990	136	1138	1272	434	508
2988	600	54	665	-58	1181	176	1358	1508	524	603
3486	711	82	775	-57	1374	218	1582	1748	617	700
3994	829	115	887	-54	1573	265	1812	1992	715	798
4563	960	155	1010	-47	1794	320	2068	2264	826	907
4983	1064	189	1105	-39	1958	363	2259	2463	908	988
5505	1192	233	1222	-28	2164	419	2497	2711	1012	1088
6008	1317	274	1333	-20	2361	471	2728	2950	1109	1184
6501	1440	322	1443	-6	2554	527	2952	3182	1211	1279
7019	1568	372	1557	10	2754	587	3187	3423	1316	1377
7498	1687	420	1663	26	2939	642	3403	3646	1412	1468
7981	1810	470	1770	43	3127	700	3624	3872	1510	1560
8499	1941	523	1885	60	3327	761	3861	4113	1613	1657
9075	2084	583	2009	82	3552	830	4125	4382	1733	1767
9507	2199	634	2109	100	3720	885	4324	4584	1821	1850
10020	2331	691	2223	123	3920	951	4564	4826	1928	1948
10303	2404	724	2285	135	4029	987	4696	4959	1991	2002

**Table B-7: Load, Strain and Stress Data from Limit Load Test of
Subsize Door Corner with $P_{max} = 10,300$ lbs
("C-Section" Test)**

Load	Ch. 11	Ch. 12	Ch. 13	Ch. 14	Ch. 15	Ch. 16	Ch. 17	Ch. 18	Ch. 19
(lb)	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$
2	1	-1	1	2	2	2	1	2	0
483	116	15	-41	-36	7	160	119	40	-139
996	236	30	-85	-73	10	327	244	81	-285
1499	347	44	-126	-105	15	491	365	121	-425
1990	452	56	-164	-131	19	650	483	160	-560
2505	559	69	-203	-156	23	818	608	200	-698
2988	657	81	-238	-176	26	974	724	239	-827
3486	756	93	-271	-197	29	1133	843	279	-955
3994	855	104	-305	-215	32	1298	967	321	-1087
4563	964	117	-341	-235	34	1481	1105	368	-1230
4983	1043	125	-368	-247	38	1616	1206	402	-1339
5505	1141	136	-402	-261	40	1786	1334	444	-1473
6008	1234	146	-437	-278	38	1950	1459	481	-1600
6501	1324	155	-464	-287	43	2110	1580	523	-1725
7019	1418	165	-494	-298	45	2277	1705	567	-1853
7498	1503	173	-521	-308	48	2430	1822	608	-1971
7981	1589	181	-548	-317	50	2588	1942	650	-2092
8499	1680	189	-580	-328	50	2756	2070	692	-2221
9075	1783	198	-612	-339	50	2943	2213	741	-2360
9507	1857	205	-638	-345	52	3084	2321	779	-2470
10020	1947	212	-667	-354	54	3252	2451	824	-2598
10303	1996	216	-682	-358	54	3343	2522	849	-2667

**Table B-7: Load, Strain and Stress Data from Limit Load Test of
Subsize Door Corner with $P_{max} = 10,300$ lbs
("C-Section" Test)**

Load	Ch. 1	Ch. 2	Ch. 3	Ch. 4	Ch. 5	Ch. 6	Ch. 7	Ch. 8	Ch. 9	Ch. 10
(lb)	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$
0	0	0	0	0	0	1	1	0	0	0
1011	183	-11	225	-38	401	34	459	525	163	209
2026	390	10	451	-57	801	95	920	1037	340	415
3040	609	53	673	-64	1197	176	1378	1531	526	614
3997	828	111	885	-59	1570	262	1811	1992	708	800
4993	1066	187	1104	-45	1958	361	2261	2466	902	992
6069	1329	280	1343	-22	2376	477	2747	2973	1114	1198
7004	1561	369	1549	5	2738	583	3171	3409	1302	1377
8005	1811	469	1770	38	3126	701	3625	3876	1505	1567
9028	2071	580	1997	77	3523	828	4092	4353	1716	1763
9990	2319	688	2211	118	3898	951	4534	4801	1917	1947
10486	2449	746	2323	142	4092	1017	4765	5035	2026	2043
11052	2595	812	2449	169	4312	1091	5031	5301	2145	2152
11487	2707	864	2545	191	4480	1149	5247	5516	2243	2238
12034	2848	929	2667	219	4692	1224	5521	5790	2383	2345
12495	2969	985	2770	243	4873	1289	5771	6036	2485	2438
12993	3100	1047	2881	269	5067	1358	6045	6305	2595	2539
13521	3237	1111	2998	299	5274	1432	6356	6608	2705	2650
14019	3368	1175	3111	327	5471	1503	6669	6911	2821	2757
14583	3516	1246	3236	359	5694	1583	7064	7285	2968	2884
15034	3636	1304	3338	385	5873	1650	7370	7577	3083	2986

**Table B-8: Load, Strain and Stress Data from 150% Limit Load Test of
Subsize Door Corner with $P_{max} = 15,600$ lbs
("C-Section" Test)**

Load	Ch. 11	Ch. 12	Ch. 13	Ch. 14	Ch. 15	Ch. 16	Ch. 17	Ch. 18	Ch. 19
(lb)	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$
0	0	0	0	0	0	0	0	0	0
1011	239	31	-89	-78	8	331	247	79	-292
2026	459	57	-170	-136	16	661	491	159	-571
3040	665	82	-243	-180	23	988	733	238	-840
3997	852	102	-309	-214	29	1296	963	314	-1087
4993	1039	123	-373	-244	35	1617	1201	394	-1341
6069	1237	143	-440	-272	40	1963	1461	482	-1614
7004	1406	160	-497	-292	46	2264	1687	560	-1848
8005	1582	175	-555	-310	52	2588	1931	645	-2098
9028	1763	191	-613	-327	56	2920	2185	733	-2353
9990	1930	205	-668	-342	61	3236	2426	819	-2593
10486	2017	211	-697	-348	64	3401	2552	863	-2719
11052	2114	219	-728	-356	67	3586	2694	914	-2859
11487	2190	225	-751	-360	69	3727	2806	953	-2968
12034	2286	233	-781	-366	70	3905	2944	1002	-3105
12495	2369	239	-807	-370	72	4055	3064	1045	-3224
12993	2458	247	-834	-375	75	4215	3192	1088	-3351
13521	2554	254	-863	-380	77	4383	3327	1136	-3486
14019	2648	263	-890	-383	80	4541	3457	1182	-3617
14583	2758	271	-922	-386	82	4716	3603	1231	-3767
15034	2847	279	-947	-389	85	4859	3721	1272	-3887

**Table B-8: Load, Strain and Stress Data from 150% Limit Load Test of
Subsize Door Corner with $P_{max} = 15,600$ lbs
("C-Section" Test)**

Load	Ch. 1	Ch. 2	Ch. 3	Ch. 4	Ch. 5	Ch. 6	Ch. 7	Ch. 8	Ch. 9	Ch. 10
(lb)	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$
0	0	0	0	0	0	0	0	0	-1	0
2141	346	2	412	-55	743	91	867	989	330	402
3184	566	41	638	-64	1146	173	1333	1494	520	606
4221	801	102	866	-60	1546	266	1798	1988	716	807
5171	1025	171	1074	-48	1913	360	2225	2438	901	991
6357	1312	270	1333	-23	2370	488	2758	2993	1135	1220
7219	1523	350	1523	0	2701	586	3147	3395	1309	1383
8323	1800	460	1766	36	3128	717	3647	3909	1534	1595
9443	2085	580	2015	79	3562	855	4159	4432	1767	1811
10461	2345	692	2240	122	3957	986	4624	4906	1980	2008
11282	2555	786	2423	161	4274	1094	5001	5287	2153	2166
12427	2852	921	2677	217	4720	1248	5530	5821	2396	2388
13945	3245	1105	3015	298	5308	1457	6234	6529	2721	2683
15181	3570	1260	3293	369	5791	1632	6816	7108	2988	2926
16357	3882	1410	3559	440	6250	1806	7453	7741	3264	3175
17361	4143	1536	3779	502	6633	1970	8257	8488	3558	3440
18384	4413	1670	4007	568	7036	2145	9101	9285	3869	3738
19351	4669	1792	4219	631	7425	2318	9928	10069	4294	4054
20249	4904	1903	4413	690	7803	2489	10711	10416	4681	4384
21331	5182	2038	4642	762	8290	2715	1594	11309	5167	4828
22373	5442	2170	4855	834	8788	2939	834	12239	5615	5318
23369	5680	2274	5051	902	9290	3181	713	13191	6049	5845
24292	5892	2374	5226	965	9780	3422	668	1343	6576	6398
25059	6058	2441	5365	1019	10197	3635	615	1105	6979	6926
22922	6204	2490	5488	1068	10595	3862	577	969	7779	7484
23694	5660	2199	5031	869	10909	1315	715	945	348	544

**Table B-9: Load, Strain and Stress Data from Ultimate Failure Test of
Subsize Door Corner with $P_{fail} = 25,660$ lbs
("C-Section" Test)**

Load	Ch. 11	Ch. 12	Ch. 13	Ch. 14	Ch. 15	Ch. 16	Ch. 17	Ch. 18	Ch. 19
(lb)	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$	$\mu\epsilon$
0	0	0	0	0	0	0	0	0	0
2141	447	55	-167	-132	17	674	503	162	-595
3184	656	78	-240	-175	24	1011	750	240	-869
4221	854	100	-310	-209	32	1344	995	320	-1135
5171	1031	116	-372	-236	38	1650	1221	394	-1378
6357	1246	137	-445	-262	45	2032	1505	489	-1680
7219	1399	150	-498	-280	51	2310	1713	559	-1897
8323	1594	166	-562	-299	56	2669	1982	650	-2175
9443	1789	180	-627	-316	63	3036	2260	747	-2458
10461	1964	193	-685	-330	68	3370	2515	836	-2715
11282	2104	203	-731	-339	72	3641	2721	909	-2923
12427	2299	214	-795	-352	78	4020	3012	1012	-3211
13945	2555	228	-879	-366	86	4526	3399	1152	-3596
15181	2763	238	-949	-375	94	4939	3717	1267	-3914
16357	2976	250	-1018	-382	101	5316	4022	1377	-4227
17361	3197	264	-1082	-384	109	5615	4281	1475	-4515
18384	3436	270	-1149	-383	117	5921	4547	1578	-4823
19351	3684	269	-1216	-380	128	6217	4800	1674	-5128
20249	3930	255	-1281	-372	138	6502	5036	1763	-5424
21331	4251	228	-1364	-361	153	6858	5321	1860	-5800
22373	4582	180	-1453	-345	171	7225	5599	1948	-6187
23369	4918	114	-1546	-326	192	7604	5867	2033	-6594
24292	5259	29	-1650	-305	223	7991	6120	2110	-7033
25059	5580	-62	-1762	-283	257	2782	6335	2169	-7487
22922	5876	-240	565	-30	-806	1173	6321	224	-8230
23694	222	-360	538	7	-822	1113	6523	206	-8802

**Table B-9: Load, Strain and Stress Data from Ultimate Failure Test of
Subsize Door Corner with $P_{fail} = 25,660$ lbs
("C-Section" Test)**

Load (lb)	Ros. 1 Max Strain	Ros. 1 Min Strain	Ros. 1 Max. Stress	Ros. 1 Min. Stress	Ros. 1 Princ. Angle	Ros. 2 Max Strain	Ros. 2 Min Strain	Ros. 2 Max. Stress	Ros. 2 Min. Stress	Ros. 2 Princ. Angle
0	0	0	0	0		0	0	0	0	
2141	507	-54	5557	1111	26	34	-183	-277	-2083	-74
3184	752	-74	8255	1714	25	42	-256	-460	-2933	-77
4221	987	-89	10866	2347	24	48	-323	-643	-3723	-79
5171	1200	-103	13233	2911	24	51	-383	-825	-4426	-81
6357	1462	-117	16148	3638	23	56	-453	-1033	-5251	-83
7219	1648	-129	18219	4141	23	60	-503	-1176	-5845	-84
8323	1889	-143	20891	4796	23	64	-565	-1367	-6584	-86
9443	2132	-158	23594	5448	22	69	-628	-1542	-7324	-87
10461	2351	-170	26034	6062	22	73	-685	-1709	-7997	-88
11282	2528	-180	28001	6546	22	77	-731	-1826	-8529	-89
12427	2775	-196	30739	7204	21	83	-794	-1995	-9275	90
13945	3101	-216	34364	8084	21	93	-879	-2195	-10255	88
15181	3368	-233	37334	8797	21	103	-951	-2351	-11088	87
16357	3642	-248	40378	9560	20	113	-1021	-2493	-11904	86
17361	3930	-259	43605	10413	20	125	-1089	-2609	-12673	85
18384	4250	-277	47159	11296	20	139	-1161	-2716	-13496	84
19351	4587	-303	50891	12151	19	155	-1234	-2788	-14309	83
20249	4934	-336	54702	12954	19	174	-1306	-2840	-15112	82
21331	5398	-386	59782	13954	18	200	-1400	-2877	-16152	81
22373	5900	-451	65253	14932	18	232	-1502	-2876	-17259	80
23369	6434	-529	71033	15862	17	270	-1611	-2828	-18428	79
24292	6993	-623	77037	16690	16	318	-1732	-2708	-19714	78
25059	7528	-725	82742	17353	16	372	-1863	-2550	-21081	78
22922	8088	-910	88460	17166	15	576	-814	3875	-7651	4
23694	563	-381	5084	-2400	8	559	-840	3573	-8027	6

**Table B-9: Load, Strain and Stress Data from Ultimate Failure Test of
Subsize Door Corner with $P_{fail} = 25,660$ lbs
("C-Section" Test)**

This Page Left Intentionally Blank

DISTRIBUTION:

<p>1 Ron Atmur
FAA-Los Angeles ACO
ANM-12 OL
3960 Paramount Blvd.
Lakewood, CA 90712-4137</p> <p>1 Alan Baker
Aeronaut. & Maritime Research Lab
506 Lorimer St Fishermens Bend
P.O. Box 4331
Melbourne, Victoria, Australia 03001</p> <p>1 Dorenda Baker
DOT/FAA
Northwest Mountain Region
1601 Lind Ave. S. W. ANM-109
Renton, WA 98055-4056</p> <p>1 Oksana Bardygula
Bombardier
120 N. Lasalle Suite 3600
Chicago, IL 60602</p> <p>1 Scott Bays
Boeing Defense and Space Group
P.O. Box 7730, MS K84-40
Wichita, KS 67277</p> <p>1 Bob Bell
Lockheed-Martin
86 South Cobb Dr.
D/73-25 Zone 0160
Marietta, GA 30063-0160</p> <p>1 Cathy Bigelow
FAA Hughes Technical Center
AAR-433
Atlantic City Int'l. Airport, NJ 08405</p> <p>1 Brett Bolan
AFRL/MLSA
2179 Twelfth St., Rm. 122
Wright-Pat. AFB, OH 45433-7718</p> <p>1 John Brausch
AFRL/MLSA
2179 Twelfth St., Rm. 122
Wright-Pat. AFB, OH 45433-7718</p> | <p>1 Al Broz
FAA-New England, ANE 105N
12 New England Exec, Dev. Park
Burlington, MA 01803</p> <p>1 Ronald Cairo
Pratt & Whitney
MS 714-03
P.O. Box 109600
West Palm Beach, FL 33410-9600</p> <p>1 Edgard Cavalheiro
Embraer
Av. Brigadeiro Faria Lima, 2170
12227-901 - S. Jose dos Campos SP
Brasil</p> <p>1 Richard Chester
Aeronaut. & Maritime Research Lab
P.O. Box 4331
Melbourne, Victoria, Australia 03001</p> <p>1 Joe Clark
President, Aviation Partners
Box 81107
Seattle, WA 98108</p> <p>1 Tom Collins
Boeing North American
12214 Lakewood Blvd
MC-AB70
Downey, CA 90242</p> <p>1 Tobey Cordell
AFRL/MLLP
2230 Tenth St.
Wright-Pat. AFB, OH 45433-7817</p> <p>1 Bill Cusato
Federal Express
7401 World Way West
Los Angeles, CA 90045</p> <p>1 Isaac Daniel
Center for Qual. Eng. & Failure Prev.
Northwestern Univ.
2137 N. Campus Dr.
Evanston, IL 60208-3020</p> |
|---|---|

<p>1 Carver Dawkins
c/o Sikorsky Aircraft Corp.
6900 Main St., MS S424A
Stratford, CT 06497</p> <p>1 Gregg Delker
USAir
4001 N. Liberty St.
Winston-Salem, NC 27156</p> <p>1 Capt. Jason Denney
USAF
WL/FIBE, Bldg. 65
2790 D Street, Room 504
Wright-Patt. AFB, OH 45433-7402</p> <p>1 Gerry Doetkott
Northwest Airlines
MS C- Dept. C8840
St. Paul, MN 55111-3034</p> <p>1 Robert Domka
Bell Helicopter
P.O. Box 482
Fort Worth, TX 76101</p> <p>1 Cong Duong
Boeing
Mail Code 71-34
2401 E. Wardlow Road
Long Beach, CA 90807-5309</p> <p>1 Bob Eastin
FAA National Resource Specialist Fatigue
FAA- Los Angeles ACO
ANM-12 OL
3960 Paramount Blvd.
Lakewood, CA 90712-4137</p> <p>1 Jennifer Elmore
NAVAIRSYSCOM
Bldg. 2187, Suite 2380
48110 Shaw Rd. Unit 5
Patuxent River, MD 20670</p> <p>1 John Fabry
FAA Hughes Technical Center
AAR-433
Atlantic City Int'l. Airport, NJ 08405</p> | <p>1 Mike Favaloro
Textron Systems Division
201 Lowell St., MS-5128
Wilmington, MA 01887</p> <p>1 Allen Fawcett
Boeing Commercial Airplane Group
P.O. Box 3707 MC 0K-UC
Seattle, WA 98124-2207</p> <p>1 Jay Fiebig
WR-ALC/TIED
255 2nd St., Ste. 122
Robins AFB, GA 31098-1637</p> <p>1 Tom Flournoy
FAA Hughes Technical Center
AAR-430
Atlantic City Int'l. Airport, NJ 08405</p> <p>1 Stephen Galea
Aeronautical & Maritime Res. Lab
GPO Box 4331
Melbourne, Victoria 3001
Australia</p> <p>5 Dave Galella
FAA Hughes Technical Center
AAR-433
Atlantic City Int'l. Airport, NJ 08405</p> <p>1 Joe Gallagher
AFRL/ML Bldg. 45
2130 8th St, Suite 1
Wright-Pat. AFB, OH 45433-7542</p> <p>1 Wriley Gay
Naval Aviation Depot
Commanding Officer Code 4.3.4.4
PSC Box 8021
Cherry Pt., NC 28533-0021</p> <p>1 Manny Gdalevitch
Bombardier Aerospace
P.O. Box 6087, Station Centre-ville
Montreal, Quebec
Canada H3C 3G9</p> |
|---|--|

- 1 Thomas Gogel
Lufthansa Technik
FRA WA42
Lufthansa Base
60546 Frankfurt
Germany
- 1 Ulf Goranson
Boeing Commercial Airplane Group
Structures Lab & Tech. Standards
P.O. Box 3707, M/S 4510
Seattle, WA 98124-2207
- 1 Kim Graebner
The Boeing Company
1218 Armstrong Ct.
Derby, KA 67037
- 1 Ken Griess
Boeing Customer Service
Box 3707 MS 2J-62
Seattle, WA 98124-2207
- 1 Bob Grills
SAIC Ultra Image International
Two Shaw Cove
New London, CT
- 1 Don Hagemmaier
Boeing
2401 East Wardlow Rd
Mail Code 71-12
Long Beach, CA 90807-5309
- 1 Tom Hahn
UCLA
Mech/Aero Engin. Dept
405 Hilgard Ave.
Los Angeles, CA 90024-1597
- 2 David Hempe
Federal Aviation Administration
AFS 330D
800 Independence Ave. SW
Washington, DC 20591
- 1 Michael J.Hoke
Abaris Training
5401 Longley Lane Suite 49
Reno, NV 89511
- 1 David Hsu
Iowa State University
1915 Scholl Rd.
ASCII
Ames, IA 50011
- 1 Jun Hun
Boeing
3855 Lakewood Blvd.
Mail Code 35-35
Long Beach, CA 90846
- 1 Larry Ilcewicz
FAA Nat. Resource Specialist Composites
1601 Lund Ave. SW
ANM-115N
Renton, WA 98055
- 1 William Jappe
Boeing
2401 East Wardlow Rd
MC C078-0113
Long Beach, CA 90807-4418
- 1 Kevin Jones
Gulfstream
456 Price St.
Savannah, GA 31401
- 1 Rhys Jones
Monash Univ.
Dept. of Mech Engin.
Wellington Rd.,
Clayton, Victoria, Australia 3168
- 1 Jeff Kollgaard
Boeing Commercial Airplane Group
P.O. Box 3707 MS 9U-EA
Seattle, WA 98124-2207
- 1 Ron Lafaro
FAA Hughes Technical Center
AAR-433
Atlantic City Int'l. Airport, NJ 08405
- 1 Bob Lamson
Aviation Partners
Box 81107
Seattle, WA 98108

<p>1 Steve LaRiviere
Boeing
P.O. Box 3707, M/S 9U-EA
Seattle, WA 98136</p> <p>2 Jess Lewis
FAA
P.O. Box 882
Mustang, OK 73064</p> <p>1 Jack Lincoln
USAF/TOGAA
ASC/ENE Bldg. 125
2335 Seventh St., Ste. G
Wright Patterson AFB, OH 45933</p> <p>1 Gordon Lindstrom
Saab Aircraft
S-581 88 Linkoping
Sweden</p> <p>1 John Marshall
Delta Air Lines, Inc.
Dept. 563, 1500 Aviation Blvd.
Hartsfield Atlanta Int'l. Airport
Atlanta, GA 30320</p> <p>1 Jim Mazza
AFRL/MLSA
2179 Twelfth St., Rm. 122
Wright-Pat. AFB, OH 45433-7718</p> <p>1 Glae McDonald
US Airways
Maintenance Hangar
5020 Hangar Rd., P.O. Box 19004
Charlotte, NC 28219</p> <p>1 Matthew Miller
Boeing Commercial Airplane Group
P.O. Box 3707, MS 6M-67
Seattle, WA 98124-2207</p> <p>1 Tommy Mullis
WR-ALC/TIEDM
420 2nd St. Suite 100
Robins AFB, GA 31098</p> | <p>1 George Murphy
Federal Express
7401 World Way West
Los Angeles, CA 90045</p> <p>1 Amir Nasruddin
US Airways
Pittsburgh Int'l Airport
P.O. Box 12346, MC Pit/D345 Hangar 3
Pittsburgh, PA 15231-0346</p> <p>1 Arnold Nathan
Israel Aircraft Industries
Eng. Div. Dept. 4441
Ben Gurion Int'l. Airport, Israel</p> <p>1 Burl Nethercutt
American Airlines
Maintenance & Eng. Center
P.O. Box 582809, MD 23
Tulsa, OK 74158-2809</p> <p>1 Jim Newcomb
FAA Hughes Technical Center
AAR-433
Atlantic City Int'l. Airport, NJ 08405</p> <p>1 Don Oplinger
FAA Hughes Technical Center
ACD-220, Bldg. 210
Atlantic City Int'l Airport, NJ 08405</p> <p>1 John Oxenford
Syncrude Canada Ltd.
9421 - 17 Ave.
Edmonton, Alberta T6N 1H4</p> <p>1 Don Palmer
Boeing
P.O. Box 516
MC 102-1111
St. Louis, MO 63166</p> <p>1 David Rashi-Dian
Boeing
Internal Mail Code 35-35
3855 Lakewood Blvd
Long Beach, CA 90846</p> |
|---|---|

- 1 Stan Rokhlin
Ohio State Univ.
Edison Joining Tech. Center
1248 Arthur E. Adams Dr.
Columbus, OH 43221
- 1 Francis Rose
Aeronaut. & Maritime Research Lab
506 Lorimer St Fishermens Bend
P.O. Box 4331
Melbourne, Victoria, Australia 03001
- 1 Paul Rutherford
Boeing Defense and Space Group
P.O. Box 3999
MS 4X-56
Seattle, WA 98124-2499
- 1 Lt. Jim Ryan
AFRL/VASE
2790 D St., Rm. 504
Wright-Pat. AFB, OH 45433-7402
- 1 Forrest Sandow
AFRL/VASA Bldg. 45
2130 8th St., Suite 1
Wright-Pat. AFB, OH 45433
- 1 Christine Scala
Aeronaut. & Maritime Research Lab
506 Lorimer St Fishermens Bend
P.O. Box 4331
Melbourne, Victoria, Australia 03001
- 1 Bill Schweinburg
WR-ALC/TIED
420 Second St., Ste. 100
Robins AFB, GA 31098
- 1 Paul Sconyers
DOT/FAA
Atlanta Aircraft Cert. Office
1701 Columbia Ave., Ste. 2-160
College Park, GA 30337-2748
- 1 Nick Shah
Boeing
2401 E. Wardlow Rd.
Long Beach, CA 90807-5309
- 1 Surendra Shah
Lockheed-Martin Aero. Systems
1465 Sumter Drive
Marietta, GA 30064
- 2 William Shurtleff
Iowa State University
1915 Scholl Road
Ames, IA 50011
- 1 Bob Singler
Textron Systems Division
201 Lowell St., MS-1113
Wilmington, MA 01887-2941
- 1 Chris Smith
FAA Hughes Technical Center
AAR-430
Atlantic City Int'l. Airport, NJ 08405
- 1 Marti Smith
DynCorp./ARTI
One Ridgmar Centre
6500 West Freeway
Fort Worth, TX 76116-2187
- 2 Fred Sobeck
FAA Flight Standards
AFS 330D
800 Independence Ave. SW
Washington, DC 20591
- 1 Barry Spigel
Southwest Research Institute
P.O. Drawer 28510
San Antonio, TX 78228-0510
- 1 Paul Tan
FAA Hughes Technical Center
AAR-433
Atlantic City Int'l. Airport, NJ 08405
- 1 Bob Thomas
Wayne State University
Institute for Manufacturing
66 W. Hancock 242 Physics
Detroit, MI 48201

- 1 Patrick Walter
Texas Christian University
Engineering Dept., Box 298640
Ft. Worth, TX 76129
- 1 Mike Waddell
AFRL/MLMP
2977 P St., Bldg. 653, Suite 6
Wright-Pat. AFB, OH 45433-7739
- 1 D. S. Wahl
Boeing
P.O. Box 16858 MS P24-16
Philadelphia, PA 19142-0858
- 1 Hans Weber
Weber Technology Applications
7916 Laurelridge Rd
San Diego, CA 92120
- 1 Bud Westerman
Boeing Defense & Space Group
P.O. Box 3707
MC 4X-56
Seattle, WA 98124-2207
- 1 Dwight Wilson
Boeing
2401 East Wardlow Rd
MC C071-0013
Long Beach, CA 90807-5309
- 1 Nancy Wood
Boeing
P.O. Box 516
MC 1-2-1111
St. Louis, MO 63166
- 1 Raymond Worley
Delta Air Lines, Inc.
1500 Aviation Blvd. TOC-1
Dept. 521/NDI
Hartsfield Atlanta Int'l. Airport
Atlanta, GA 30320
- 1 Rich Yarges
FAA-ANM 112
1601 Lind Ave. SW
Renton, WA 98055-4099

- 1 Jin Yu
Boeing
2401 East Wardlow Rd
MC 71-34
Long Beach, CA 90807-5309

1 MS-0958 Carol Adkins, 1472
1 MS-0958 Ernie Correa, 1472
1 MS-0958 Tommy Guess, 1472
1 MS-0724 Joan Woodard, 6000
1 MS-0766 Dori Ellis, 6300
1 MS-0767 Henry Abeyta, 6312
1 MS-0615 Dick Perry, 6352
1 MS-0615 Mike Ashbaugh, 6352
1 MS-0615 Craig Jones, 6352
1 MS-0615 David Moore, 6352
20 MS-0615 Dennis Roach, 6352
1 MS-0615 Kirk Rackow, 6352
1 MS-0615 Mike Valley, 6352
5 MS-0615 Phil Walkington, 6352
1 MS-0555 Mark Garrett, 9133
1 MS-0439 Jim Redmond, 9234
1 MS-0829 Floyd Spencer, 12323
1 MS-9018 Central Technical Files, 8940-2
2 MS-0899 Technical Library, 4916
1 MS-0619 Review & Approval Desk, 15102
For DOE/OTSI
1 MS-0161 Patent and Licensing Office, 11500