P-142

NASA Contractor Report 172358

IN-24

97700

ACEE COMPOSITE STRUCTURES TECHNOLOGY

Papers by Boeing Commercial Airplane Company

Boeing Commercial Airplane Company
P.O. Box 3707
Seattle, Washington 98124

August 1984

(NASA-CR-172358) ACEE COMPOSITE STRUCTURES TECHNOLOGY (Boeing Commercial Airplane Co.) 142 p Avail: NTIS HC A07/MF A01 CSCL 11D

N87-28614

Unclas

G3/24 0097700

National Aeronautics and
Space Administration

Langley Research Center
Hampton, Virginia 23665

FOREWORD

The NASA Aircraft Energy Efficiency (ACEE) Composite Primary Aircraft Structures Program has made significant progress in the development of technology for advanced composites in commercial aircraft. Under NASA sponsorship, commercial airframe manufacturers have now demonstrated technology readiness and cost-effectiveness of advanced composites for secondary and medium primary components and have initiated a concerted program to develop the data base required for efficient application to safety-of-flight wing and fuselage structure. Timely dissemination of technical information acquired in these programs is achieved through distribution of reports and periodic special oral reviews.

The third special oral review of the ACEE Composites Programs was held in Seattle, Washington, on August 13-16, 1984. The conference included comprehensive reviews of all composites technology development programs by ACEE composites contractors — Boeing, Douglas, and Lockheed. In addition, special sessions included selected papers on NASA-sponsored research in composite materials and structures and reviews of several important Department of Defense programs in composites.

Individual authors prepared their narrative and figures in a form that could be directly reproduced. The material is essentially the same material that was orally presented at the conference. The papers were compiled in five documents. Papers prepared by personnel from Boeing Commercial Airplane Company, Douglas Aircraft Company, and Lockheed-California Company are contained in NASA CR-172358, CR-172359, and CR-172360, respectively. Papers on selected NASA-sponsored research are contained in NASA CP-2321. Papers on selected Department of Defense programs are in NASA CP-2322.

The assistance of all authors, contractor personnel, and the Research Information and Applications Division of the Langley Research Center in publishing these proceedings is gratefully acknowledged.

John G. Davis, Jr.
Technical Chairman for
ACEE Composite Structures
Technology Conference
Langley Research Center

PRECEDING PAGE BLANK NOT FILMED

CONTENTS

PRECEDING PAGE BLANK NOT FILMED

ACEE COMPOSITE STRUCTURES TECHNOLOGY CONFERENCE

AGENDA

MONDAY, AUGUST 13, 1984

[1]SESSION 1: OUTLOOK FOR COMPOSITES IN FUTURE AIRCRAFT

SESSION CHAIRMAN: Robert L. James, Jr., Manager, ACEE Project Office, NASA Langley Research Center

AIR TRANSPORTATION SYSTEMS AT THE END OF THE MILLENNIUM — Howard T. Wright, Director for Projects, NASA Langley Research Center

FUTURE COMMERCIAL VIABILITY OF COMPOSITES — Kenneth F. Holtby, Corporate Senior Vice President, The Boeing Company

THE WAY AHEAD — Russell H. Hopps, Vice President and General Manager, Engineering and Development, Lockheed—California Company

COMPOSITE AIRCRAFT MATERIALS — THE FUTURE — John B. DeVault, Vice President, Composite Materials and Structures, Hercules Aerospace

TUESDAY MORNING, AUGUST 14, 1984

SESSION 2: ACEE SECONDARY AND MEDIUM PRIMARY COMPOSITE STRUCTURES — STATUS REPORT

SESSION CHAIRMAN: Andrew J. Chapman, Technical Manager, Composites, ACEE Project Office, NASA Langley Research Center

[2]ADVANCED COMPOSITES ON BOEING COMMERCIAL AIRPLANES — John T. Quinlivan, Boeing Commercial Airplane Company

[3]DAMAGE TOLERANCE AND FAILSAFE TESTING OF THE DC-10 COMPOSITE VERTICAL STABILIZER — John M. Palmer, Jr., Clive O. Stephens, and Jason O. Sutton, Douglas Aircraft Company

[4]INITIAL STRENGTH AND HYGROTHERMAL RESPONSE OF L-1011 VERTICAL FIN COMPONENTS — Anthony C. Jackson, Lockheed-California Company

[5]RESIDUAL-STRENGTH TESTS OF L-1011 VERTICAL FIN COMPONENTS AFTER 10 AND 20 YEARS OF SIMULATED FLIGHT SERVICE — Osvaldo F. Lopez, NASA Langley Research Center

[5]WORLDWIDE FLIGHT AND GROUND BASED EXPOSURE OF COMPOSITE MATERIALS — H. Benson Dexter, NASA Langley Research Center, and Donald J. Baker, U.S. Army Structures Laboratory, NASA Langley Research Center

[5]COMPARISON OF TOUGHENED COMPOSITE LAMINATES USING NASA STANDARD DAMAGE TOLERANCE TESTS — Jerry G. Williams, NASA Langley Research Center, T. Kevin O'Brien, U.S. Army Structures Laboratory, NASA Langley Research Center, and Andrew J. Chapman, NASA Langley Research Center

Footnotes on page x

BOEING

TUESDAY AFTERNOON, AUGUST 14, 1984

SESSION 3: REVIEW OF SELECTED NASA RESEARCH IN COMPOSITE MATERIALS AND STRUCTURES

SESSION CHAIRMAN: James H. Starnes, Jr., Head, Structural Mechanics Branch, NASA Langley Research Center

[5]SYNTHESIS AND TOUGHNESS PROPERTIES OF RESINS AND COMPOSITES — Norman J. Johnston, NASA Langley Research Center

[5]TENSILE STRENGTH OF COMPOSITE SHEETS WITH UNIDIRECTIONAL STRINGERS AND CRACK-LIKE DAMAGE — Clarence C. Poe, Jr., NASA Langley Research Center

[5]IMPACT DYNAMICS RESEARCH ON COMPOSITE TRANSPORT STRUCTURES — Huey D. Carden, NASA Langley Research Center

[5]POSTBUCKLING BEHAVIOR OF GRAPHITE/EPOXY PANELS — James H. Starnes, Jr., NASA Langley Research Center; John N. Dixon, Lockheed-Georgia Company; Marshall Rouse, NASA Langley Research Center

[1]DAMAGE TOLERANCE RESEARCH ON COMPOSITE COMPRESSION PANELS — Jerry G. Williams, NASA Langley Research Center

[5]STUDIES OF NOISE TRANSMISSION IN ADVANCED COMPOSITE MATERIAL STRUCTURES — Louis A. Roussos, Michael C. McGary, and Clemans A. Powell, NASA Langley Research Center

WEDNESDAY MORNING, AUGUST 15, 1984

SESSION 4: ACEE WING KEY TECHNOLOGIES

SESSION CHAIRMAN: Marvin B. Dow, Technical Manager, Composites, ACEE Project Office, NASA Langley Research Center

[2]COMPOSITE WING PANEL DURABILITY AND DAMAGE TOLERANCE TECHNOLOGY DEVELOPMENT — Robert D. Wilson, Boeing Commercial Airplane Company

[2]DESIGN DEVELOPMENT OF HEAVILY LOADED WING PANELS — Peter J. Smith, Boeing Commercial Airplane Company

Footnotes on page x

WEDNESDAY MORNING, AUGUST 15, 1984

[3]THEORY AND ANALYSIS FOR OPTIMIZATION OF COMPOSITE MULTI-ROW BOLTED JOINTS — L. John Hart-Smith, Douglas Aircraft Company

[3]DESIGN AND TEST OF LARGE WING JOINT DEMONSTRATION COMPONENTS — Bruce L. Bunin, Douglas Aircraft Company

[4]COMPOSITE WING FUEL CONTAINMENT AND DAMAGE TOLERANCE TECHNOLOGY DEVELOPMENT — Charles F. Griffin, Lockheed-California Company

[4]COMPOSITE WING FUEL CONTAINMENT AND DAMAGE TOLERANCE TECHNOLOGY DEMONSTRATION — Tom W. Anderson, Lockheed-California Company

WEDNESDAY AFTERNOON, AUGUST 15, 1984

SESSION 5: REVIEW OF SELECTED DOD PROGRAMS

SESSION CHAIRMAN: H. Benson Dexter, NASA Langley Research Center

[6]MANUFACTURING TECHNOLOGY FOR LARGE AIRCRAFT COMPOSITE PRIMARY STRUCTURE (FUSELAGE) — DESIGN SELECTION — Hank R. Fenbert, Boeing Commercial Airplane Company; Harry S. Reinert, U.S. Air Force, MLTN, Wright-Patterson AFB; Vere S. Thompson, Boeing Commercial Airplane Company

[6]MANUFACTURING TECHNOLOGY FOR LARGE AIRCRAFT COMPOSITE WING STRUCTURE — Melvin A. Price, North American Aircraft Operations, Rockwell International Corporation, and D. R. Beeler, U.S. Air Force, AFWAL/MLTN, Wright-Patterson AFB

[6]MANUFACTURING TECHNOLOGY FOR LARGE COMPOSITE FUSELAGE STRUCTURE — Richard L. Circle and R. Dennis O'Brien, Lockheed-Georgia Company

[6]DAMAGE TOLERANCE OF COMPOSITES — John E. McCarty, Boeing Military Airplane Company

[1]ADVANCED COMPOSITE AIRFRAME PROGRAM (ACAP) — Tom Mazza, U.S. Army Applied Technology Laboratory, Fort Eustis

[6]COMPOSITE STRUCTURES — IMPROVED DESIGNS FOR MILITARY AIRCRAFT — Anthony Manno, Mark S. Libeskind, Ramon Garcia, and Edward F. Kautz, U.S. Navy, Naval Air Development Center

[1]COMPOSITE STRUCTURES IN THE JVX AIRCRAFT — Keith Stevenson, Bell Helicopter Corporation

Footnotes on page x

BOEING

THURSDAY MORNING, AUGUST 16, 1984

SESSION 6: ACEE ADVANCED COMPOSITE STRUCTURES TECHNOLOGY

SESSION CHAIRMAN: Jon S. Pyle, Technical Manager, Composites, ACEE Project Office, NASA Langley Research Center

[2]BOEING — PRESSURE CONTAINMENT AND DAMAGE TOLERANCE IN FUSELAGE STRUCTURE — Ronald W. Johnson, Boeing Commercial Airplane Company

[3]DOUGLAS — JOINTS AND CUTOUTS IN FUSELAGE STRUCTURE — D. Joseph Watts, Douglas Aircraft Company

[4]LOCKHEED — IMPACT DYNAMICS AND ACOUSTIC TRANSMISSION IN FUSELAGE STRUCTURE — Anthony C. Jackson, Lockheed-California Company

[4]LOCKHEED COMPOSITE TRANSPORT WING TECHNOLOGY DEVELOPMENT

COVER/RIB CONCEPTS — Arthur M. James, Lockheed-California Company

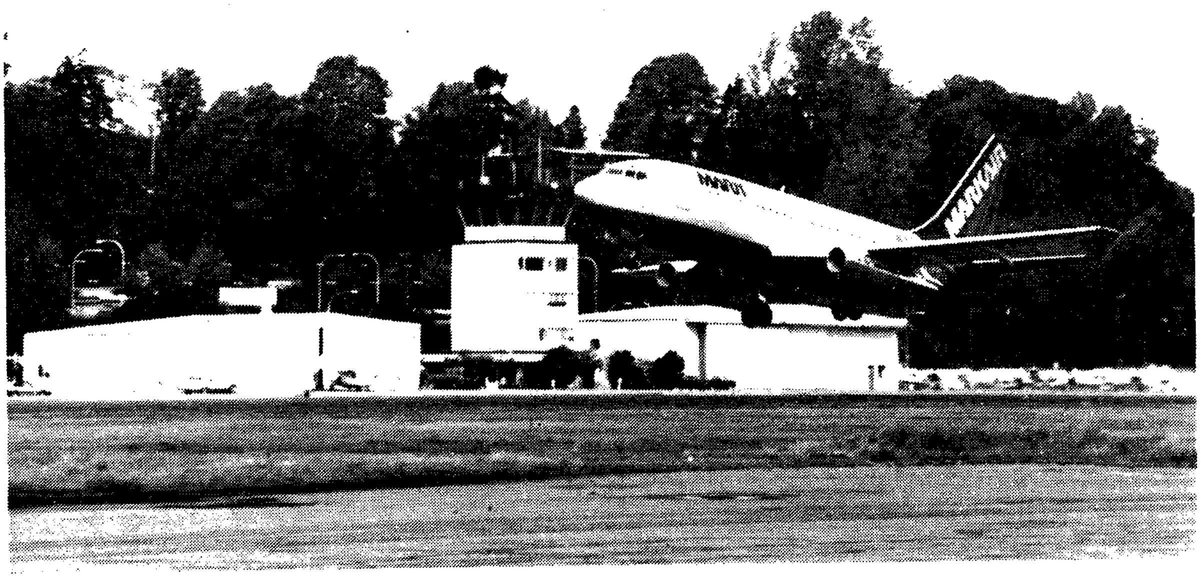
SPAR/ASSEMBLY CONCEPTS — William E. Harvill, Jr., Lockheed-Georgia Company

[1]Oral presentations only
[2]Papers contained in NASA CR-172358
[3]Papers contained in NASA CR-172359
[4]Papers contained in NASA CR-172360
[5]Papers contained in NASA CP-2321
[6]Papers contained in NASA CP-2322

BOEING

ORIGINAL PAGE IS
OF POOR QUALITY

Advanced Composites on Boeing Commercial Airplanes

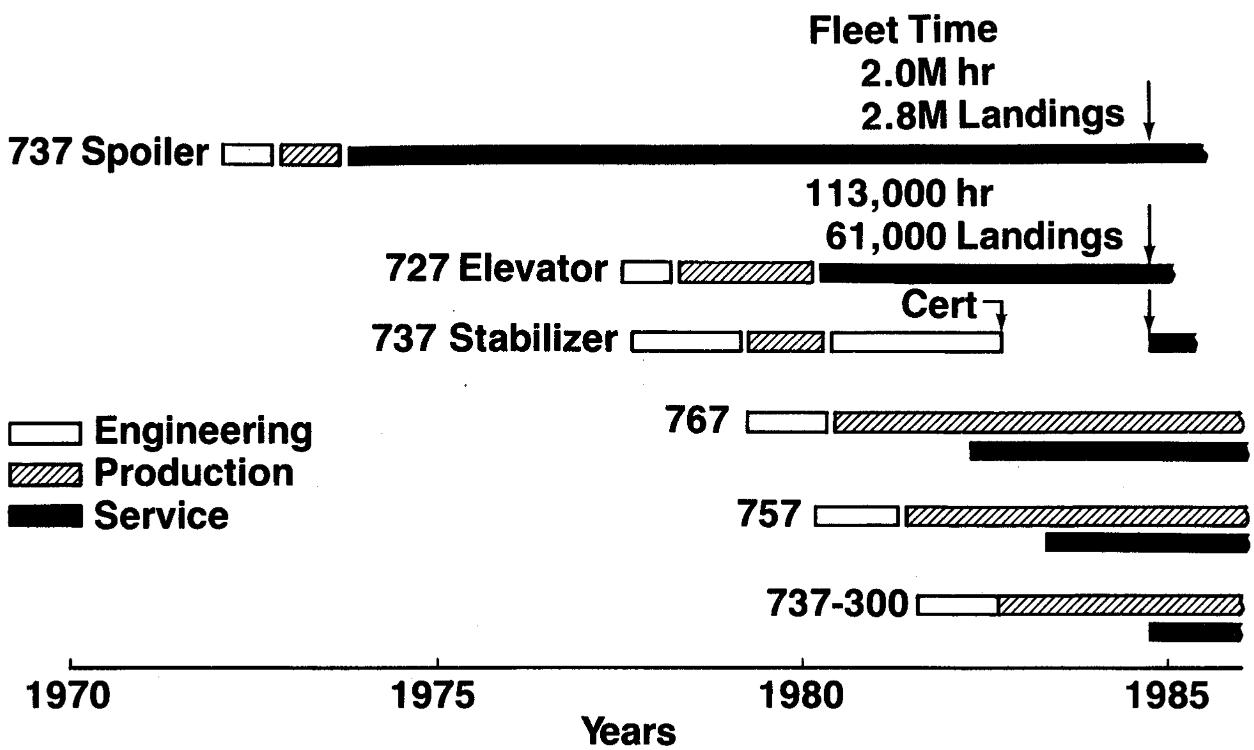


J. Quinlivan
Boeing Commercial Airplane Company
August 1984

BOEING

Major Graphite Components
(Commercial)

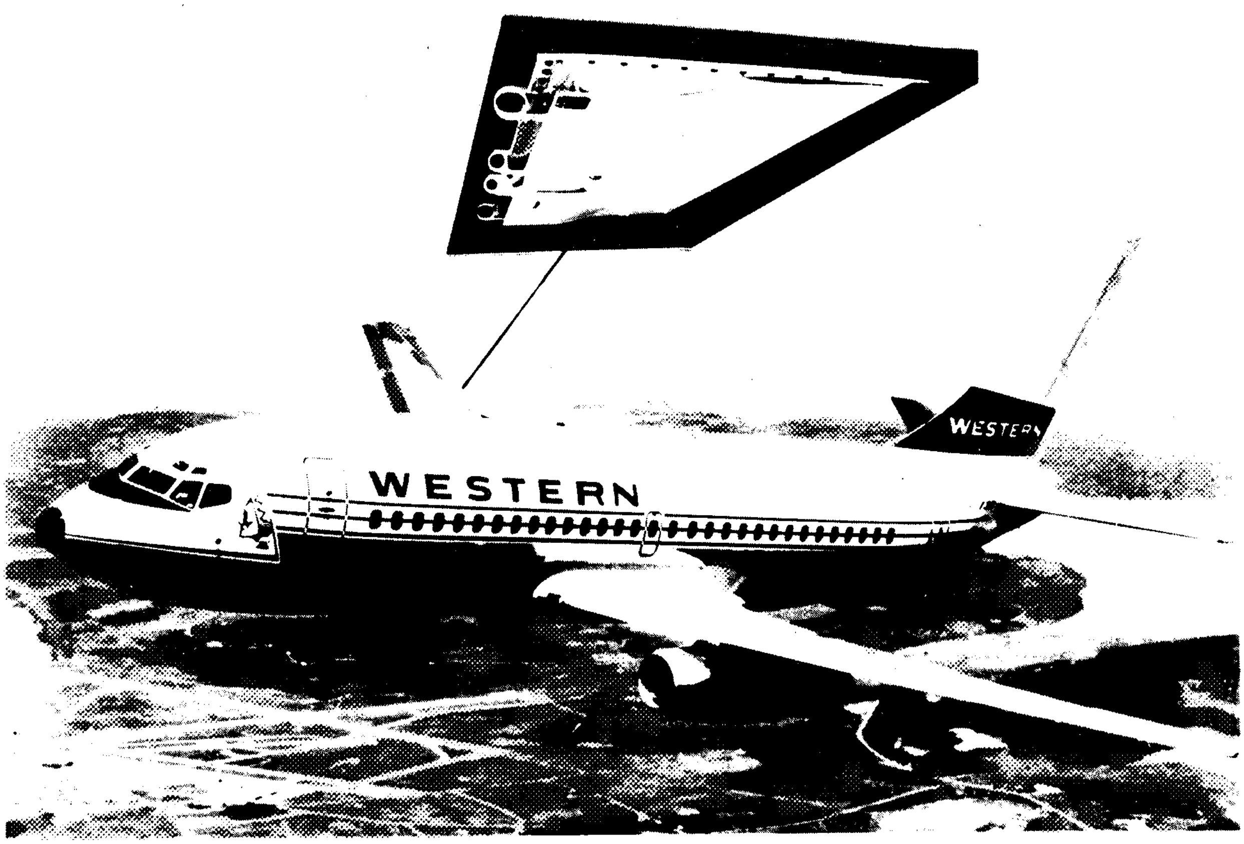
The NASA 737 spoiler and 727 elevator programs provided the engineering production and service basis necessary to commit advanced composite components to modern aircraft. The NASA 737 stabilizer provides the background for future use of composite in primary structure.



BOEING

Graphite-Epoxy Spoiler
In-Service Demonstration

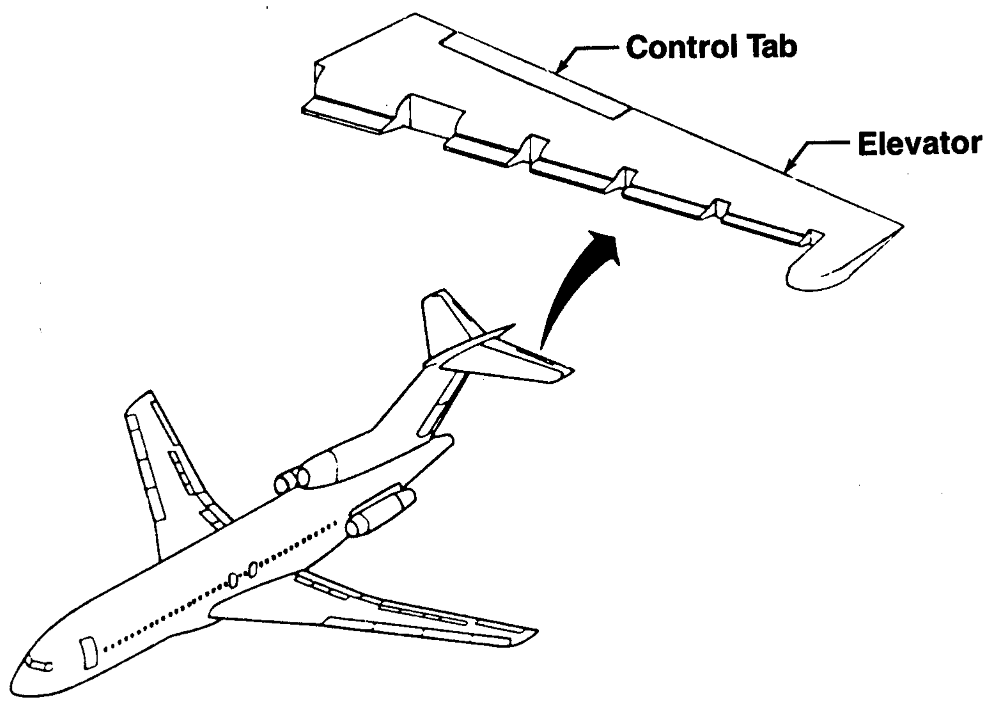
The NASA 737 graphite-epoxy spoiler program provided the first long-term usage information concerning composites in service. The success of the program is exemplified by the more than 2 million hours accumulated by these spoilers in service through the middle of 1984.



BOEING

727 Composite Components

The 727 composite elevator program provided additional information for secondary composite structures. Five shipsets of composite elevators have been introduced into service successfully.

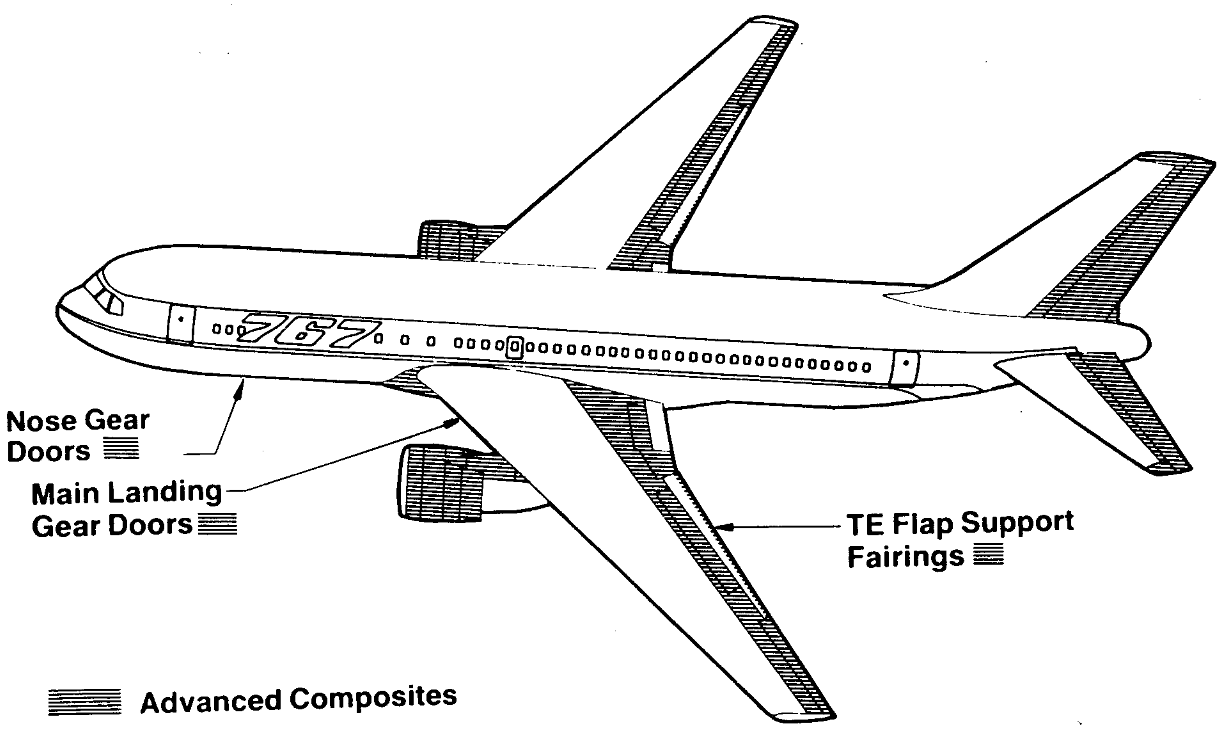


150-lb Weight Savings per Airplane
(26% Weight Savings)

BOEING

767 Advanced Composite Usage

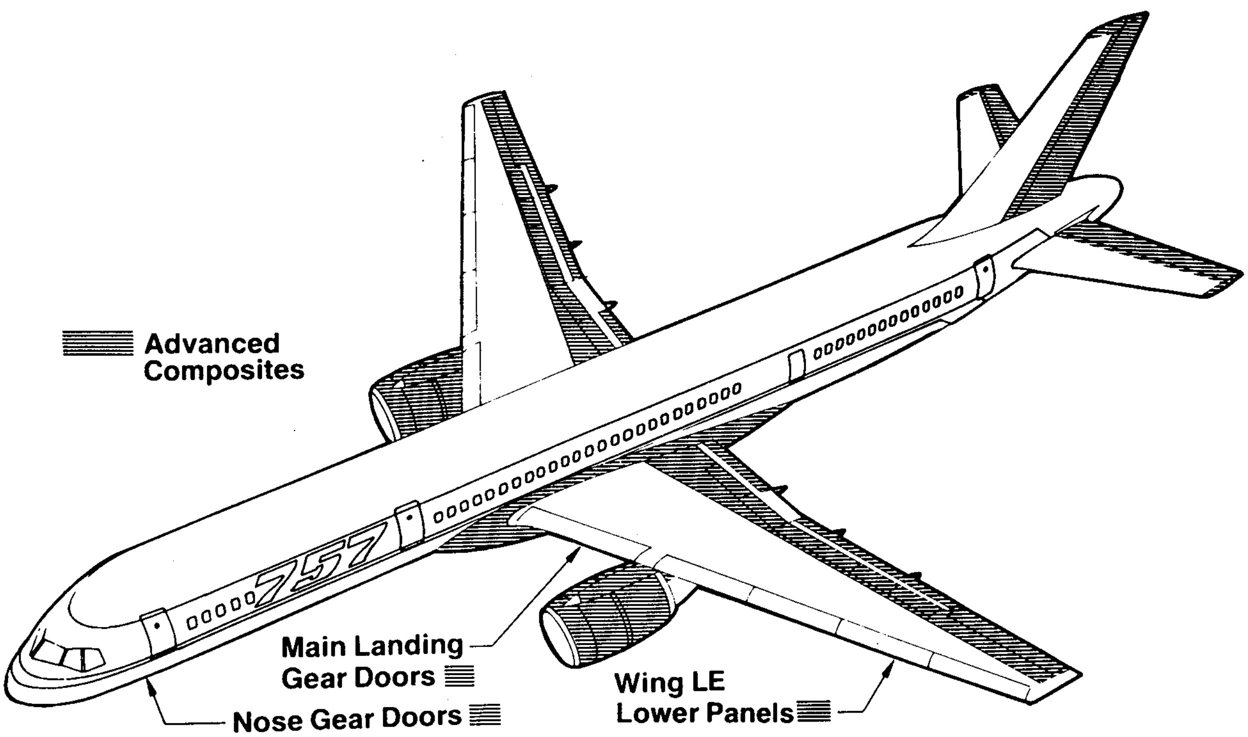
The use of advanced composites on the 767 extends to most of the flight control surfaces, the wing-to-body fairings, landing gear doors, nacelle components, and fixed trailing edges of the wing and empennage. In addition, there are many applications of advanced composites in secondary structures within the fuselage.



BOEING

757 Advanced Composite Usage

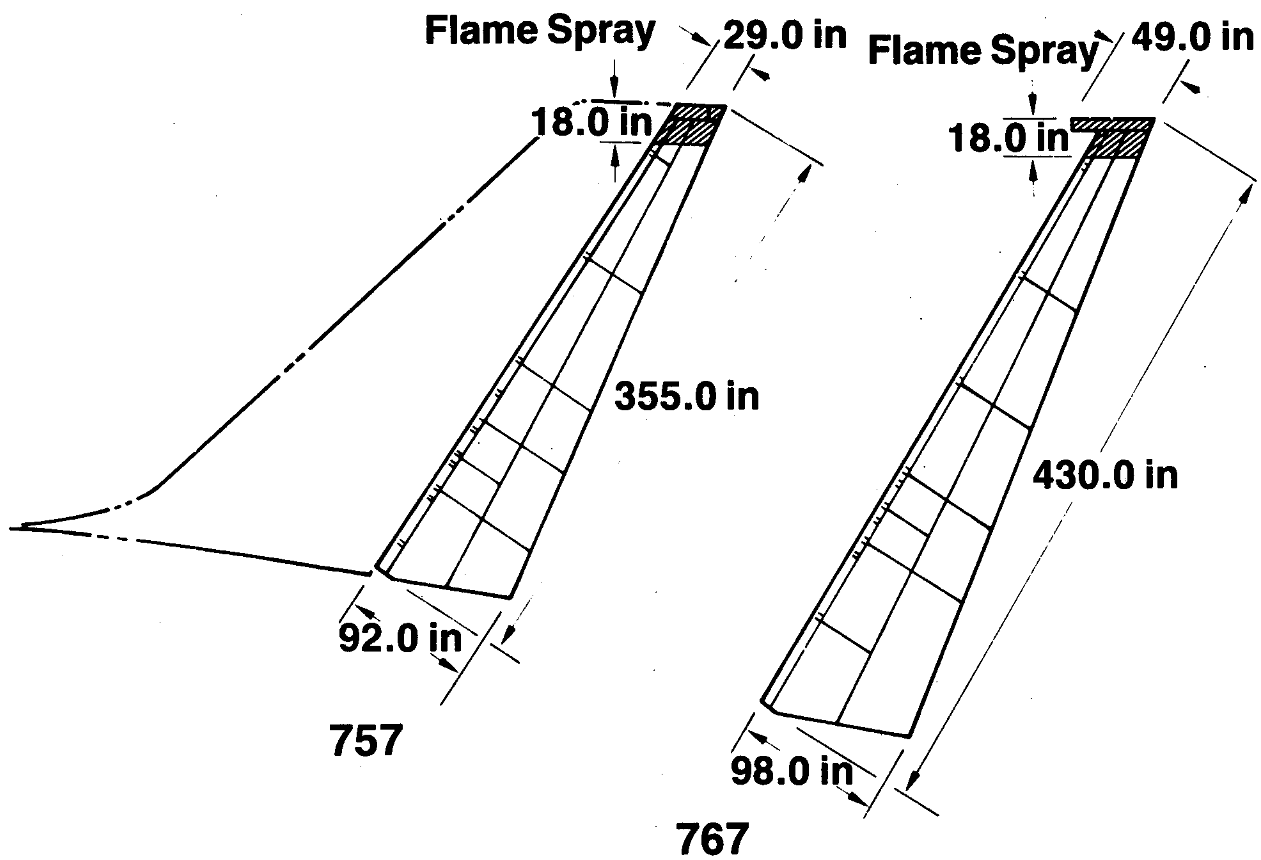
The 757 utilizes a variety of composites. In addition to the applications common to the 767, the trailing-edge flaps of the 757 are made of composites as well.



BOEING

Rudder
Graphite

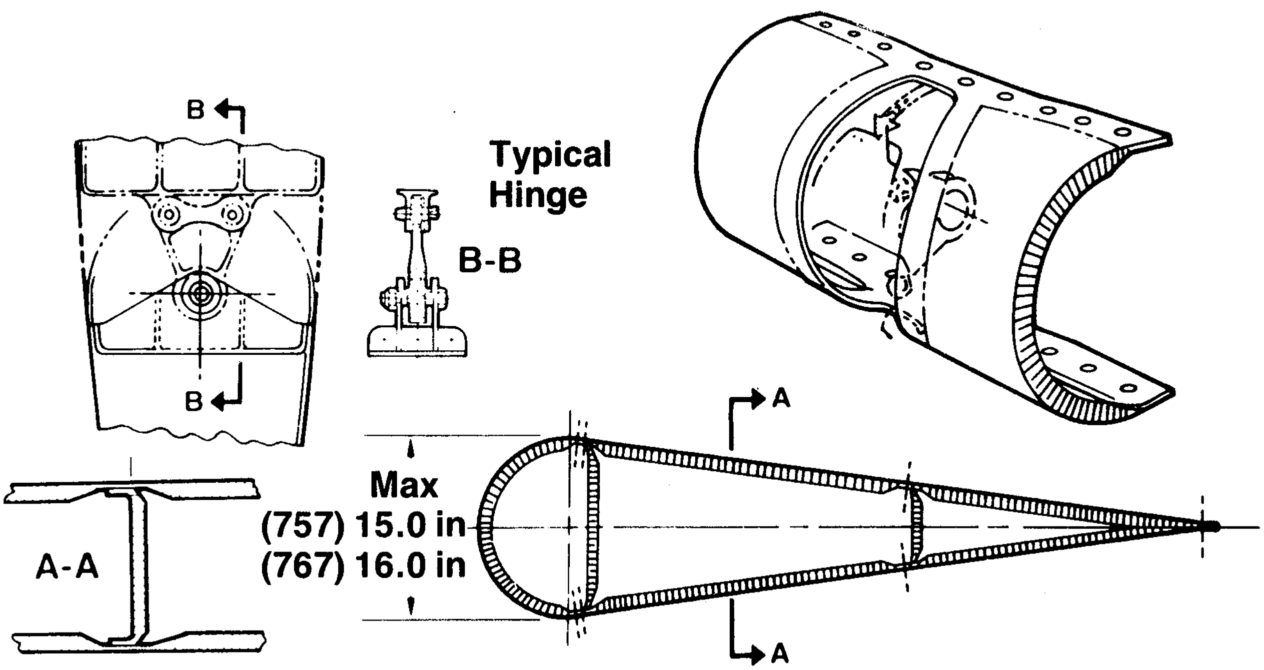
The rudder of the 767 and 757 are shown. These large composite components are constructed of graphite-epoxy materials.



BOEING

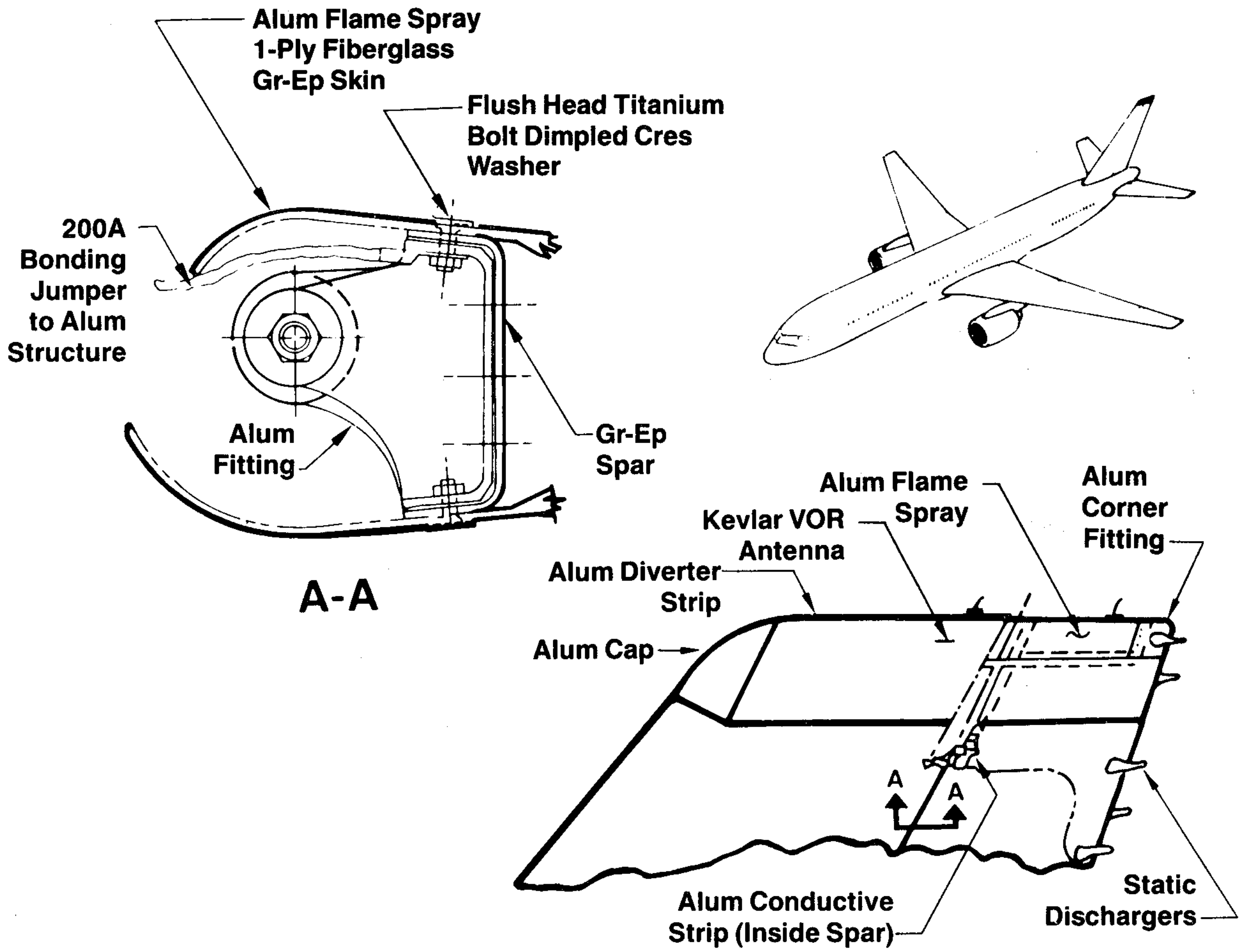
757 and 767 Rudder
Graphite

The structural arrangement selected for the composite rudder utilizes a panelized honeycomb construction. The designs include two spars, several full ribs, as well as partial ribs. The leading edge is also composite material construction. The individual details are mechanically attached.



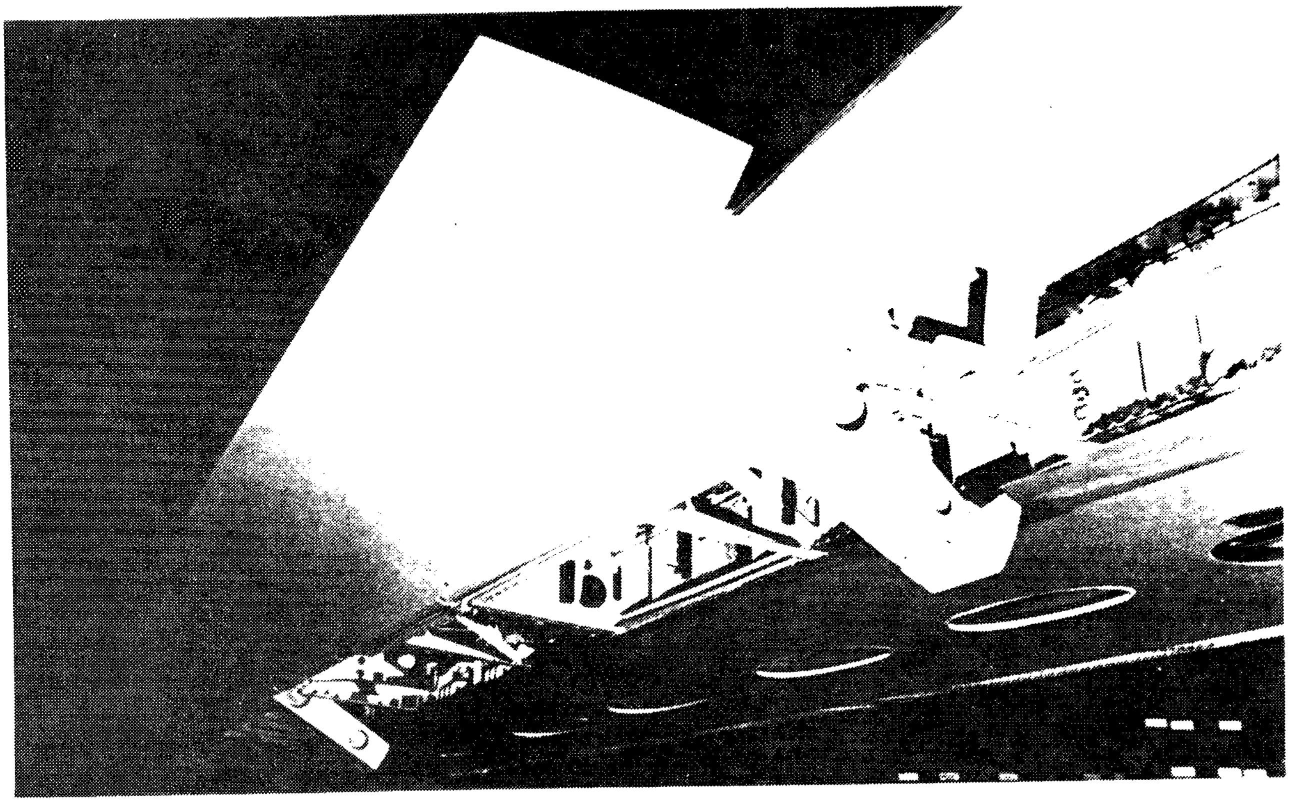
Lightning Protection

Lightning protection for the rudders and ailerons is provided by local aluminum flame spray applied directly to the composite surfaces.



767 Outboard Aileron

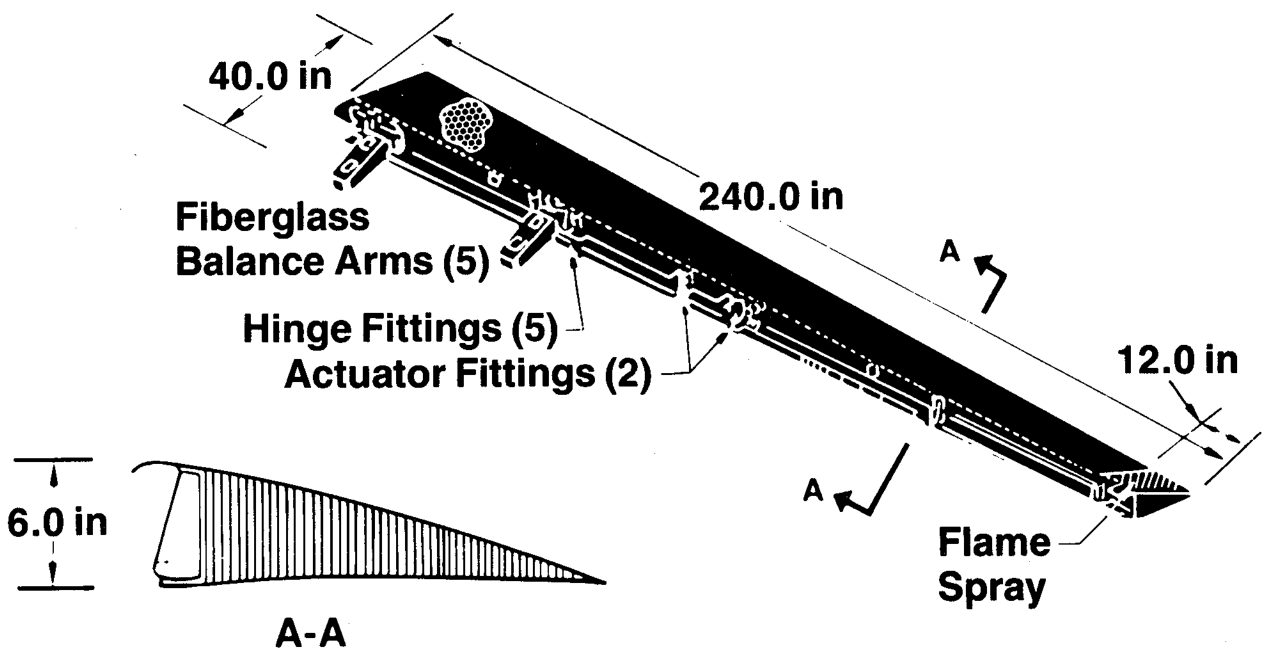
The 767 outboard aileron utilizes a full-depth honeycomb concept.



BOEING

767 Outboard Aileron
Graphite

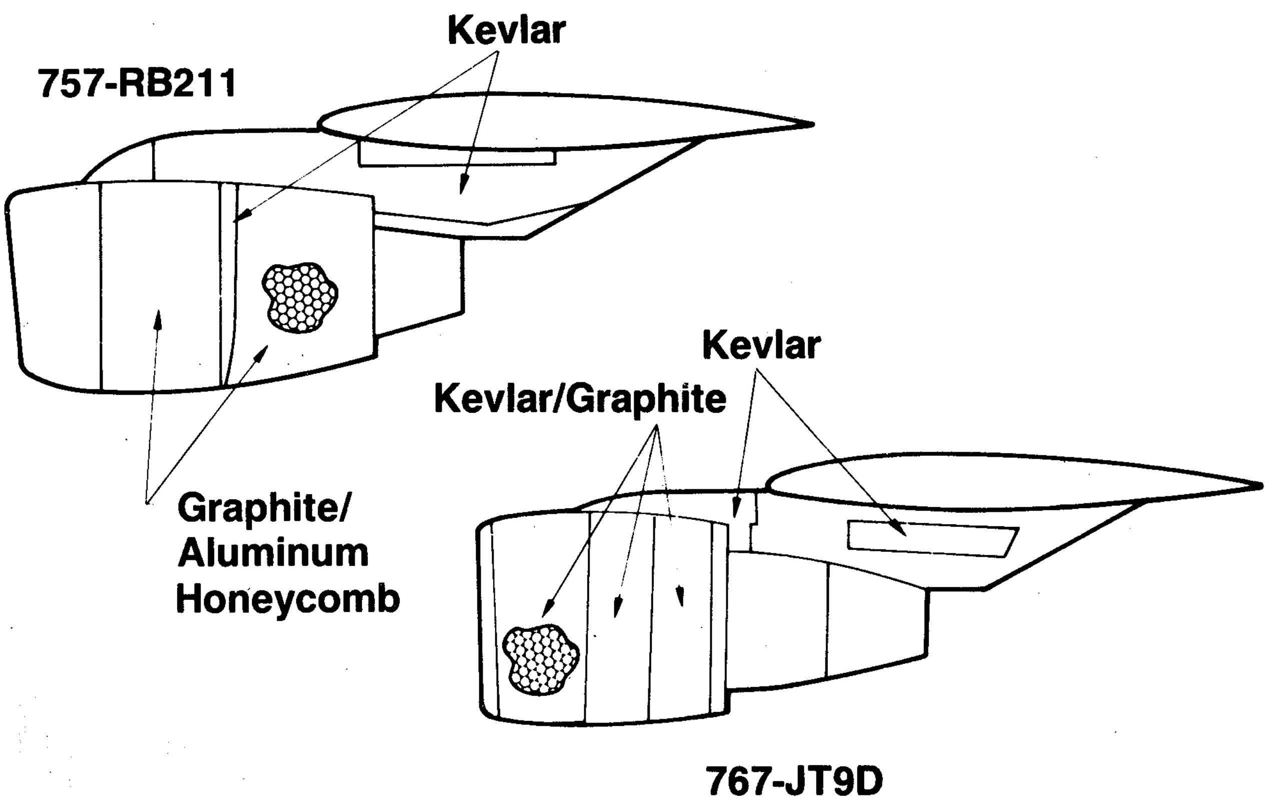
The structural arrangement selected for the composite aileron uses precured skins adhesively bonded to full-depth Nomex honeycomb core.



BOEING

Engine Strut/Pod
Kevlar/Graphite

Engine strut and pod material selections include hybrids of Kevlar and graphite as well as all-graphite or all-Kevlar aluminum honeycomb panels.

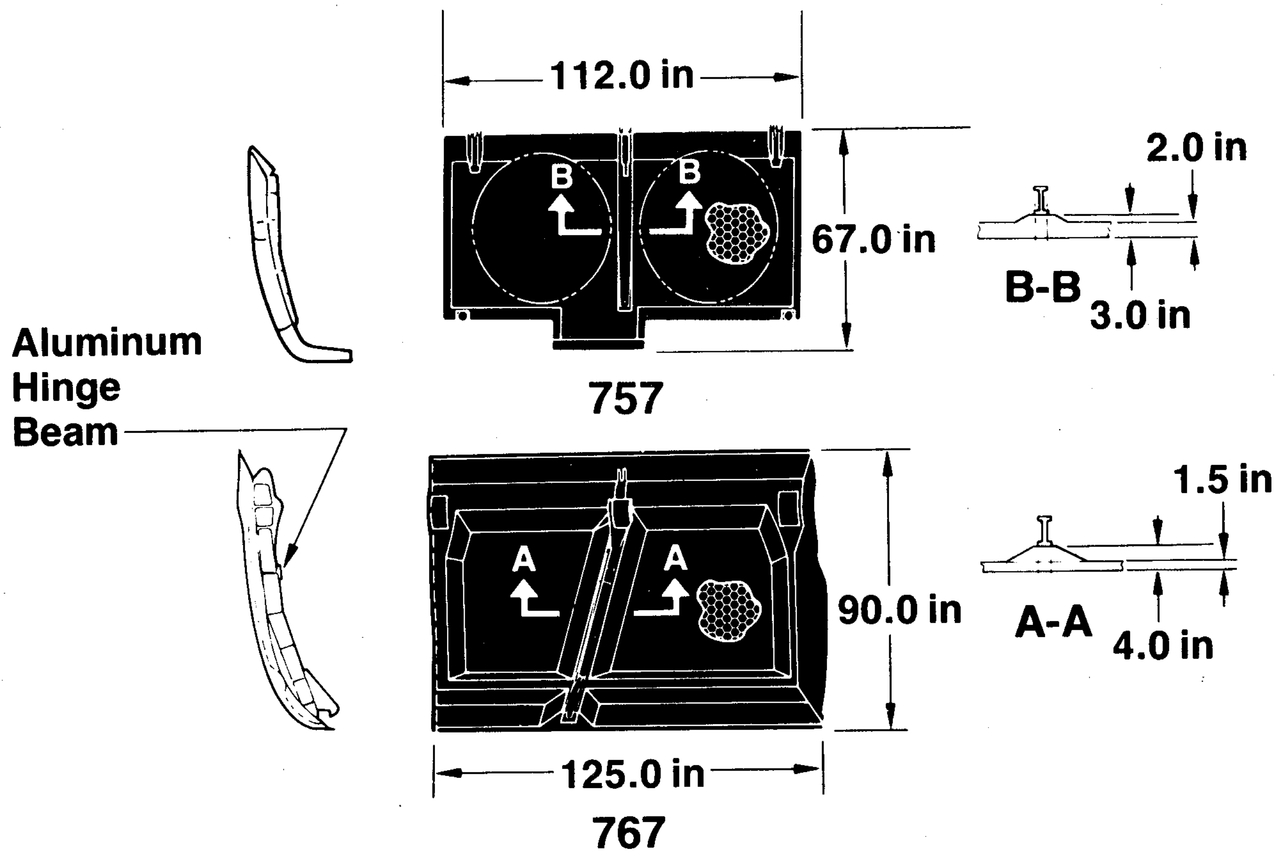


BOEING

Main Landing Gear Doors

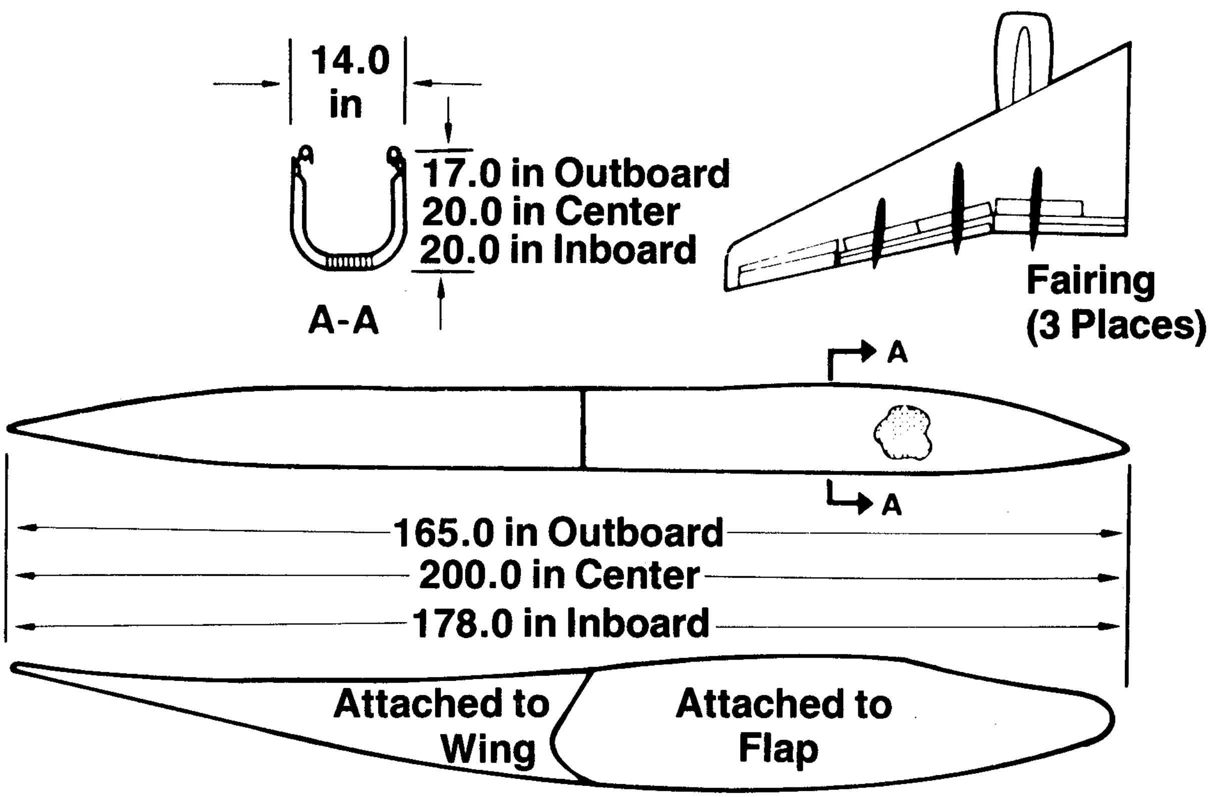
Kevlar/Graphite

The main landing gear doors of the 757 and 767 are two of the larger composite components produced today. These full-depth honeycomb core designs utilize local honeycomb beams for stiffening.



757 Flap Fairing
Kevlar/Graphite

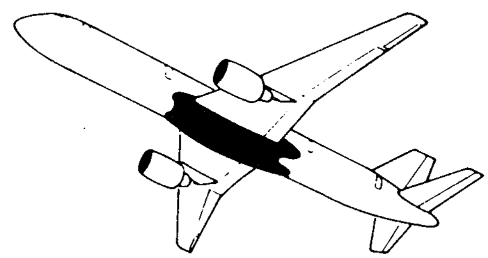
The 757 flap fairing uses a kevlar/graphite hybrid construction.



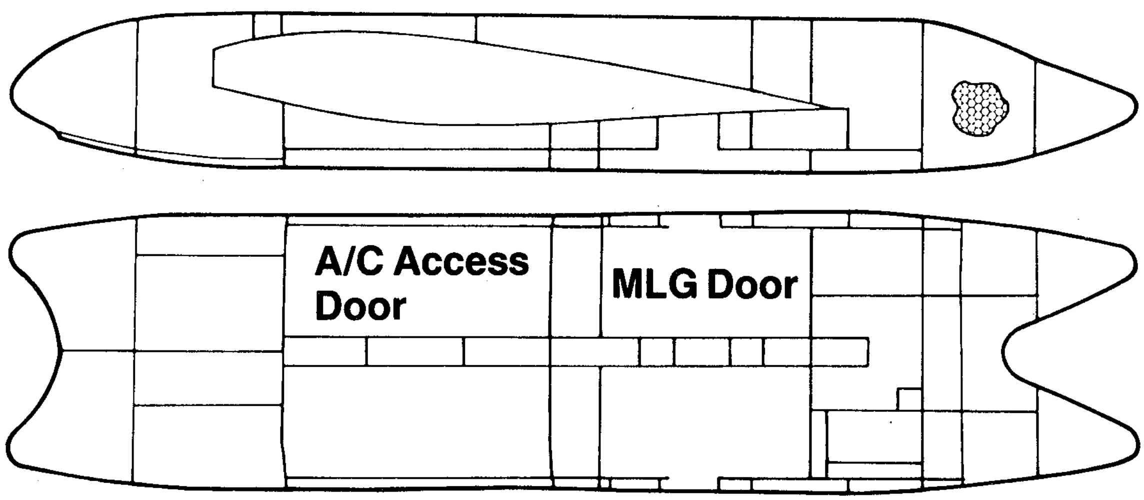
BOEING

757 Wing/Body Fixed Fairing
Kevlar/Graphite

The wing-body fixed fairings are Kevlar/graphite hybrid constructions. These large composite fairings save a substantial percentage of weight compared with the glass fiber constructions of earlier aircraft.

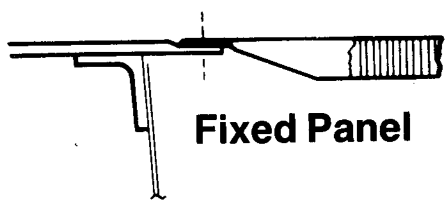


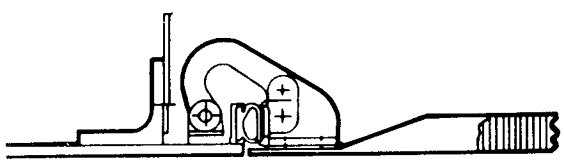
- **36 Panels**
- **0.5 x 21.0 x 28.0 in (Min)**
- **2.0 x 60.0 x 146.0 in (Max)**



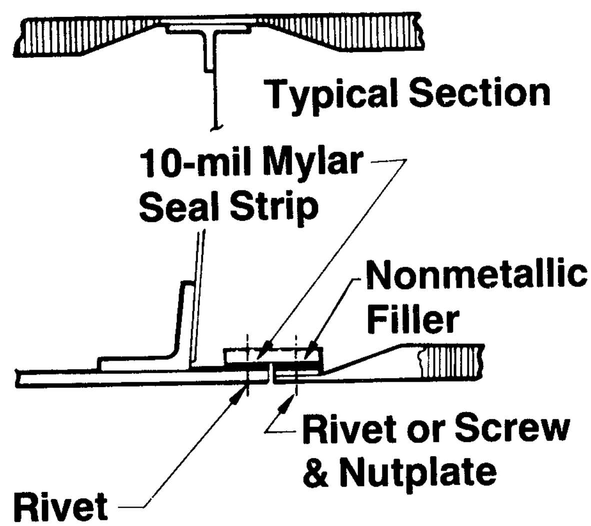
767/757 Wing Trailing Edge Panels
Kevlar/Graphite

Wing trailing-edge panels, wing-to-body fairings, and other simple fairings using all-Kevlar edge bands are mechanically attached to fixed structure using aluminum rivets.





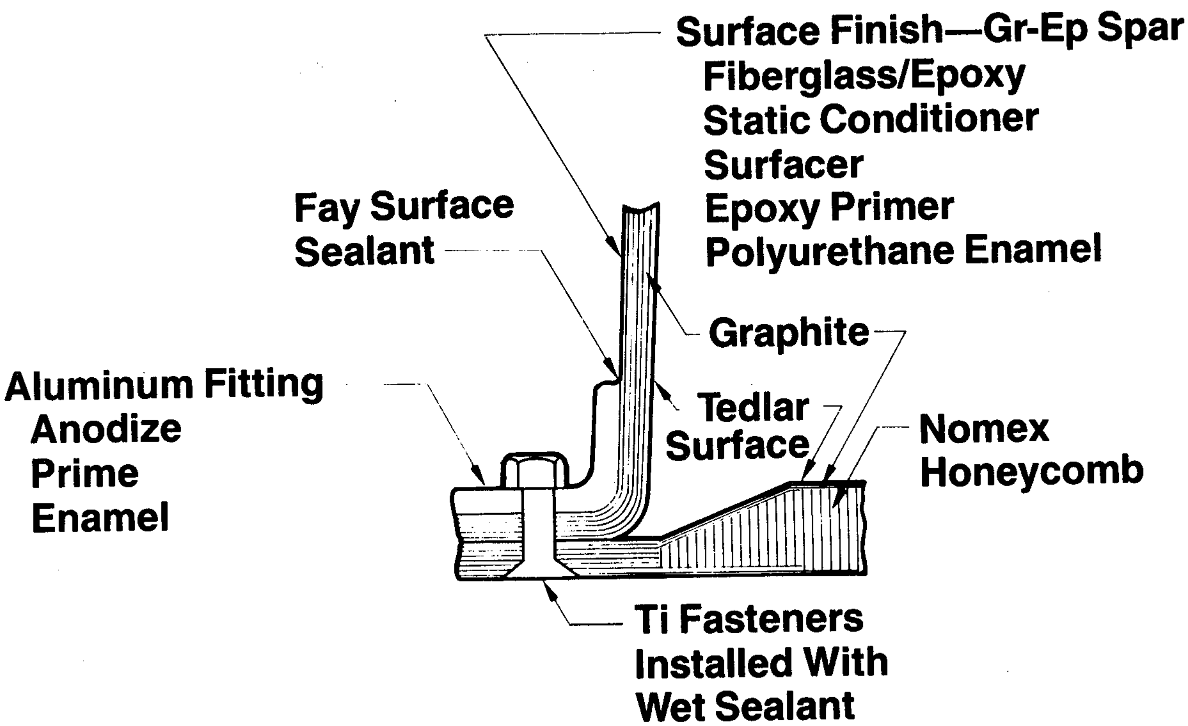
Hinged Panel



Fixed Panel or Removable Panel

Corrosion Prevention
Dissimilar Materials

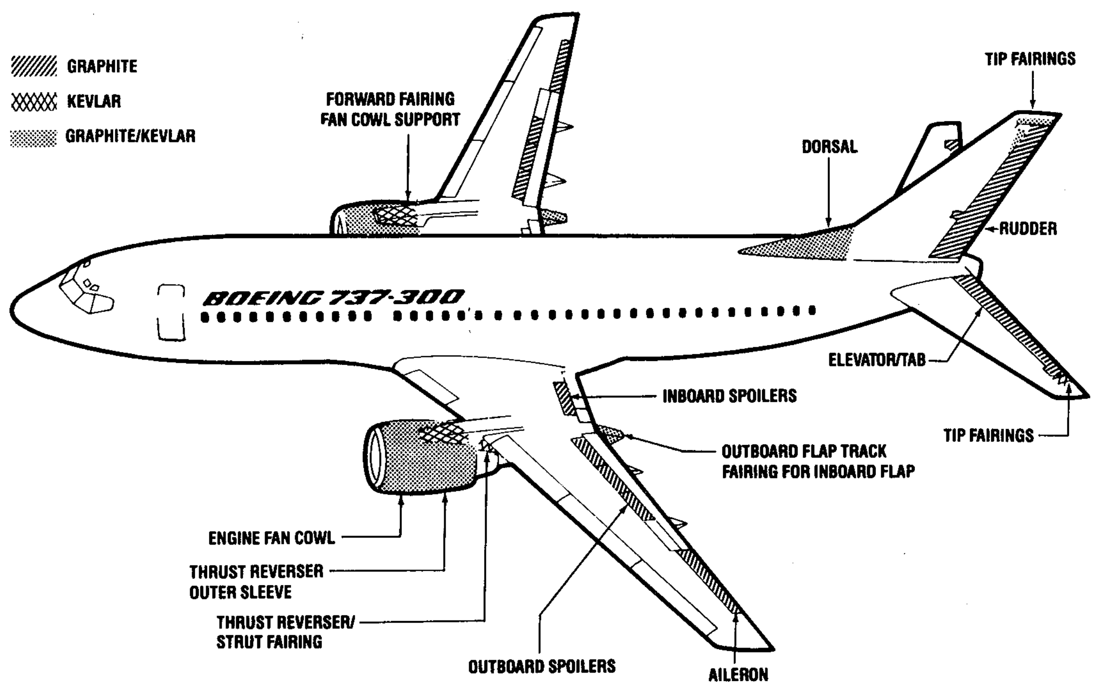
Corrosion protection is provided by isolating the aluminum metal from the composite material.



Advanced Composites Applications

Model 737-300

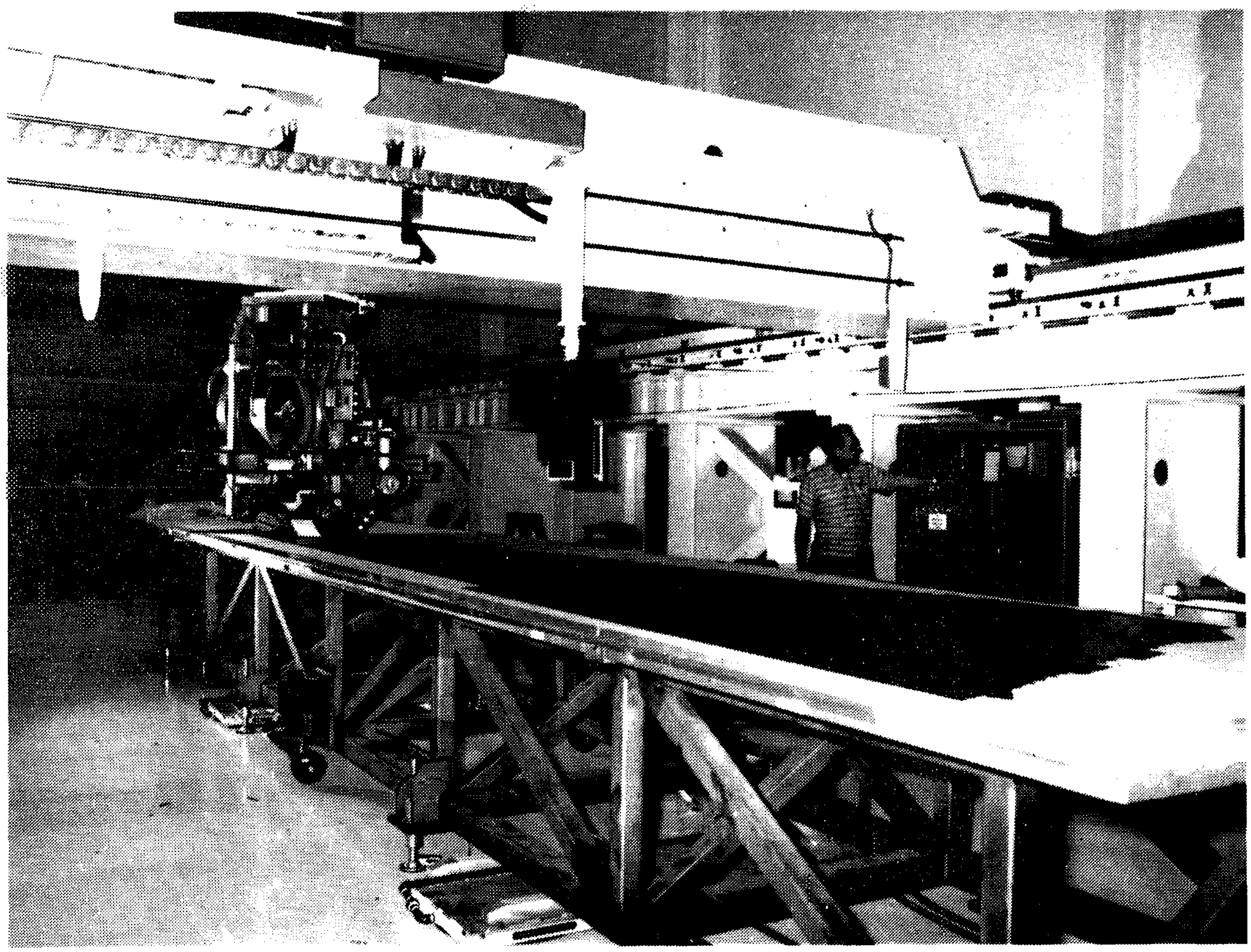
The 737-300 incorporates advanced composites in its design. A primary difference between this airplane and the 757 and 767 is the extensive use of tape materials as opposed to the fabric materials.



BOEING

Automated Flat-Tape Laminating Machine

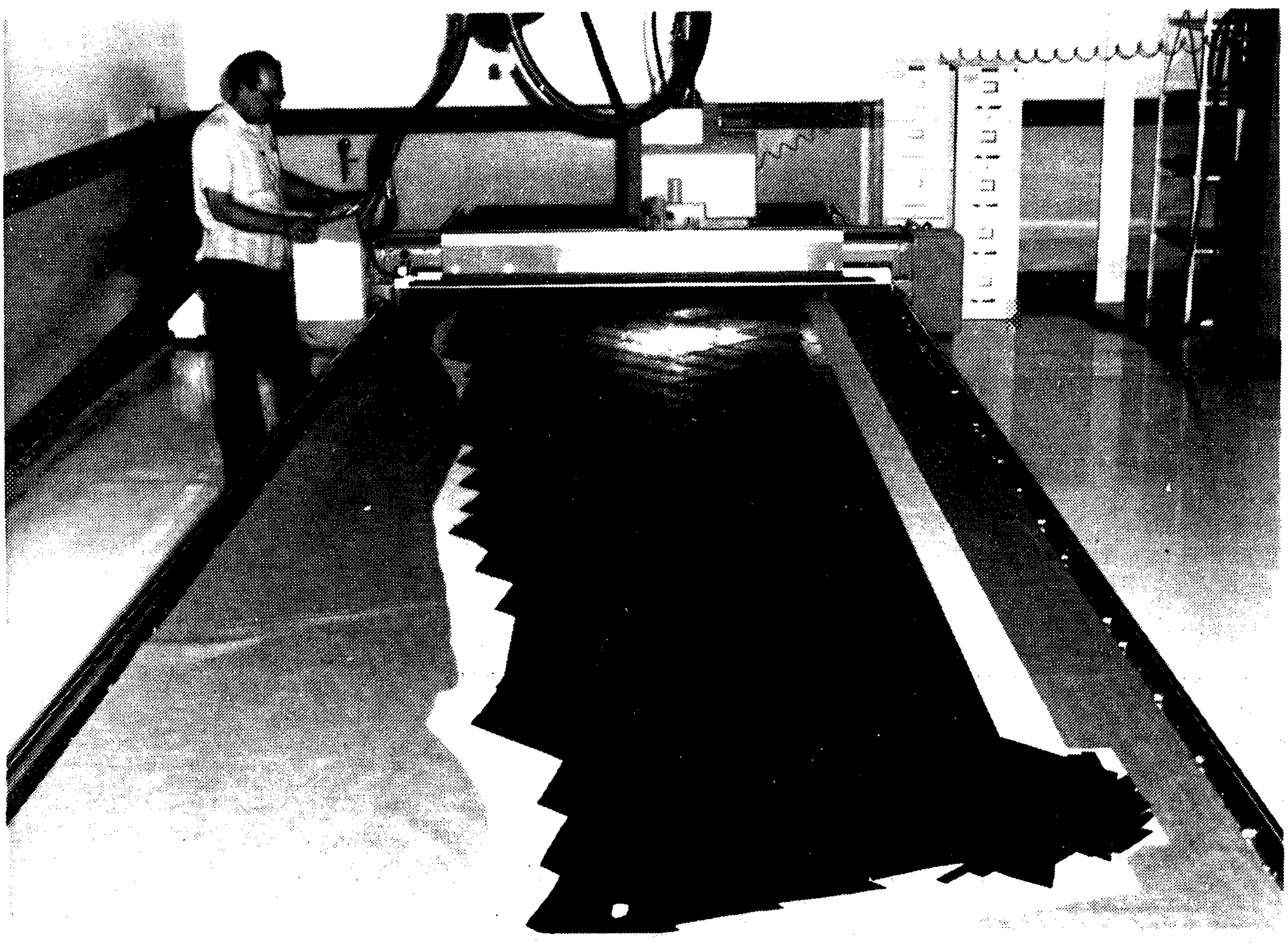
Graphite-epoxy tape is automatically dispensed using a tape laminator. This provides a major cost savings in the fabrication of composite components for the 737-300. The elevator skin is shown here.



BOEING

Numerically Controlled Cutter

The prelaid skins for the 737-300 elevator are cut automatically using a Gerber cutter. Automation is making significant inroads in composite fabrication.



Current Composite Usage

The current composite usage on Boeing commercial airplanes approaches three percent of the aircraft's structural weight.

	737-300	757	767
Total Composite Weight [1]	1500 lb	3340 lb	3380 lb
Total Weight Reduction [2]	600 lb	1490 lb	1400 lb

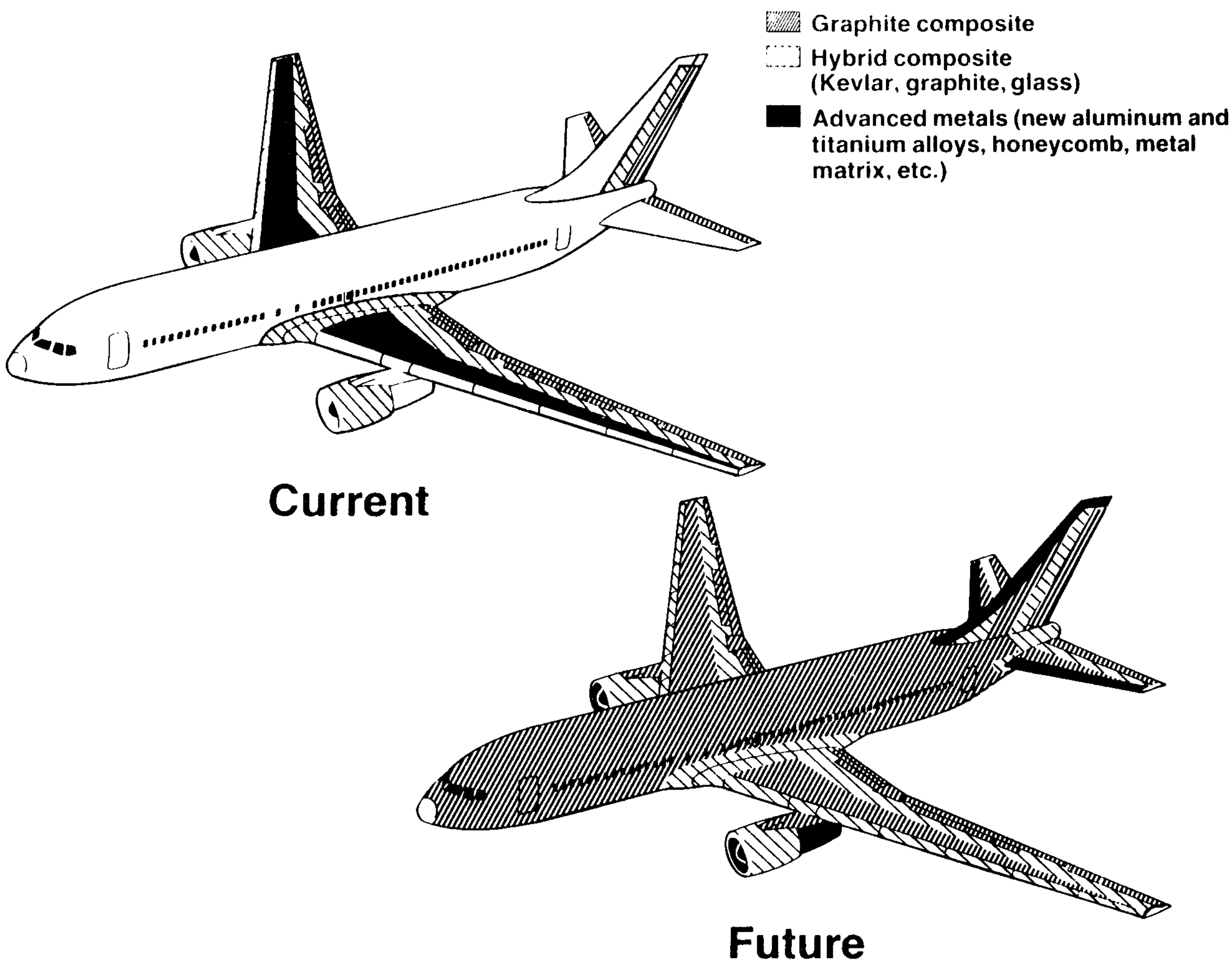
[1] Includes:
- Graphite-Epoxy
- Kevlar/Graphite
- Fiberglass/Graphite
- Fiberglass/Graphite/Kevlar
- Kevlar

[2] Includes Effects on Control Surface Balance Weights

BOEING

Advanced Structures

Current composite applications to large transport aircraft include the secondary structure of the wing and empennage as well as many fairings and nacelle components. Future aircraft can be expected to utilize significant quantities of composites in the wing and fuselage and empennage primary structures as well.



BOEING

737 Graphite-Epoxy Horizontal Stabilizer

The 737 horizontal stabilizer program has reached a major milestone with the introduction of the stabilizers into commercial service.

BOEING

Stabilizer Program Status

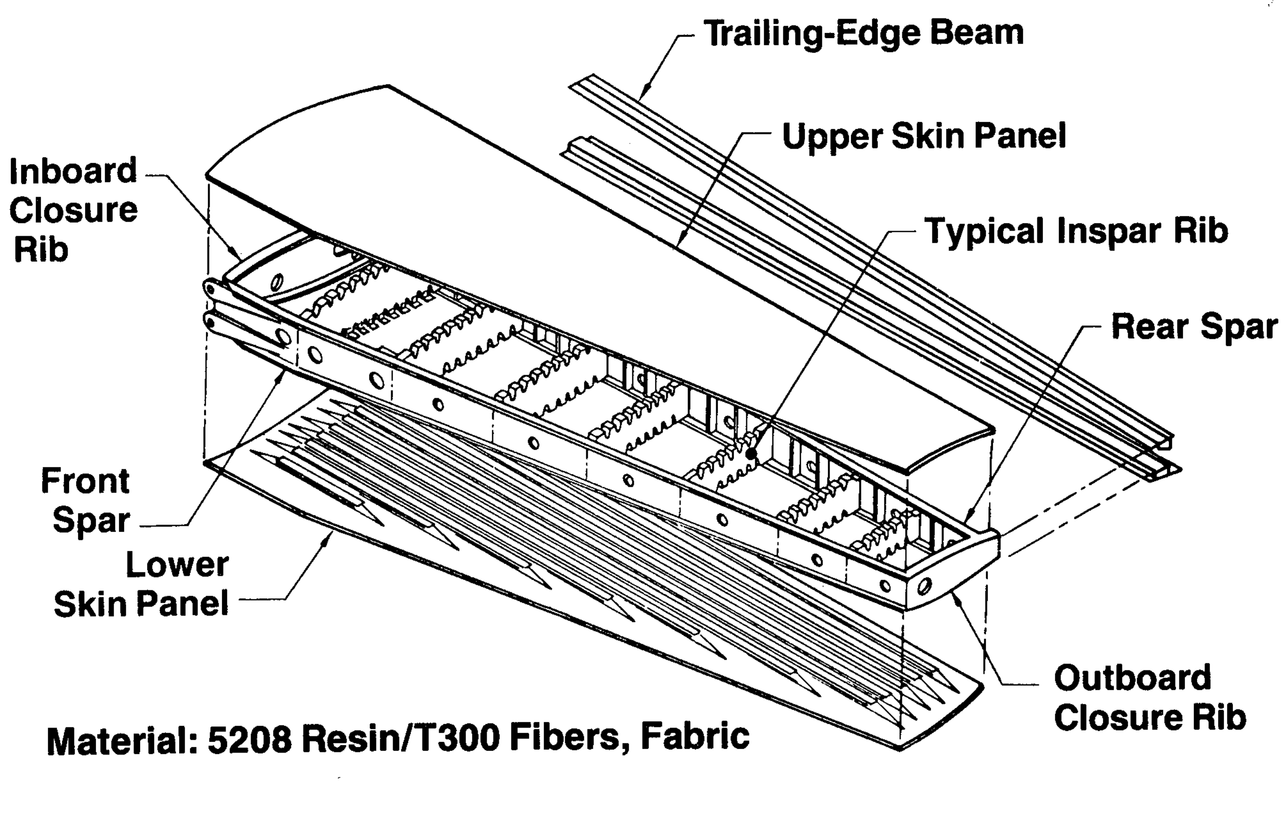
The 737 graphite-epoxy horizontal stabilizer was developed and certified by Boeing as part of the NASA/ACEE Advanced Composite Structures program. Five shipsets have been produced and were introduced into commercial service by mid-1984.

- FAA Certification: August 1982
- Five Shipsets Fabricated
- Production Service Scheduled
  - Delta (Two Units)
  - Mark-Air (Three Units)

BOEING

Structural Arrangement

The structural arrangement selected for the graphite-epoxy horizontal stabilizer uses a cocured, integrally stiffened skin and laminate front and rear spars. The skin is supported on seven inspar honeycomb ribs and laminate inboard and outboard closure ribs. The trailing-edge beam is laminate construction. The composite stabilizer was designed to match the existing interfaces for the aluminum center section structure.



Material: 5208 Resin/T300 Fibers, Fabric

737 Horizontal Stabilizer Highlights

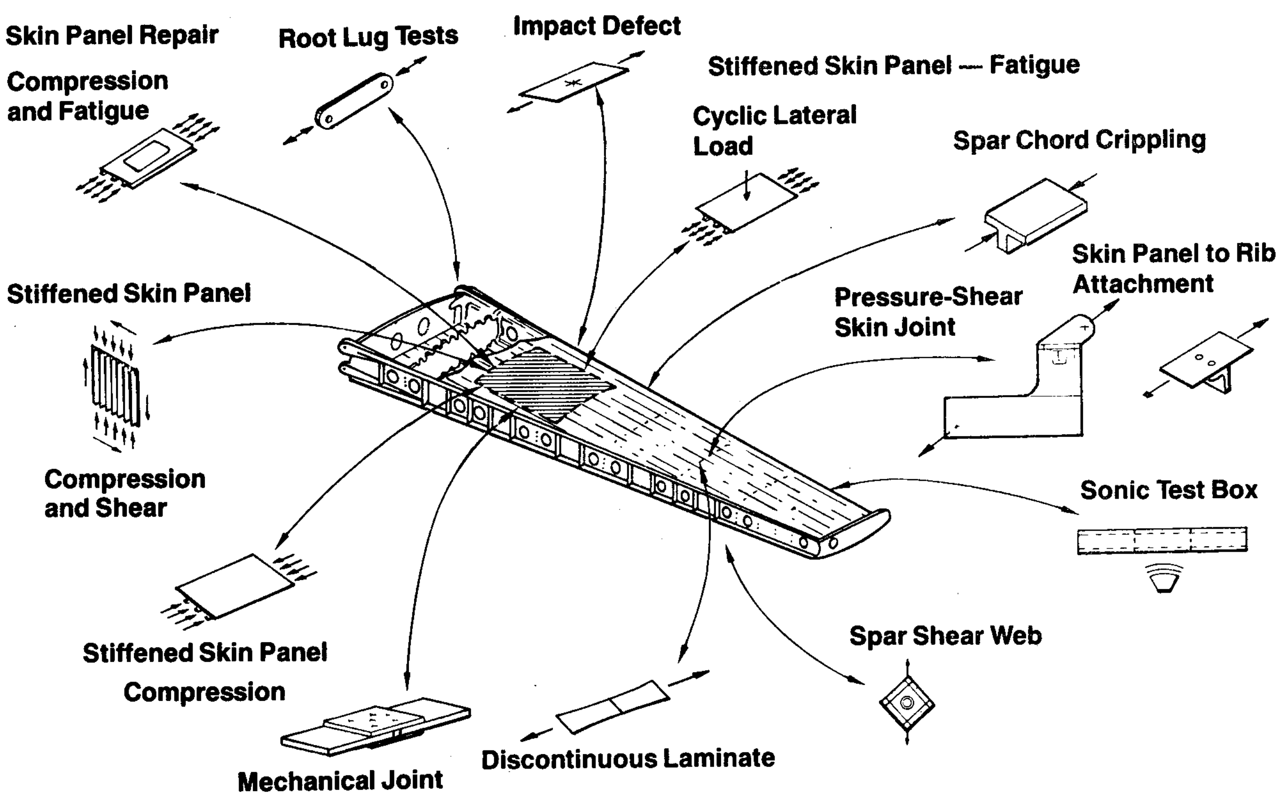
The Boeing approach to show compliance with the Federal regulations was to certify by structural analysis with supporting test evidence. Some highlights of the test program will be discussed. Additional work, including maintenance and planning, for production deliveries will be reviewed.

- Test Program
  - Ancillary Tests
  - Stub Box
  - Full-Scale Box—
    Damage Tolerance
  - Environmental Test Panel
- Maintenance Planning
- Certification
- Delivery

BOEING

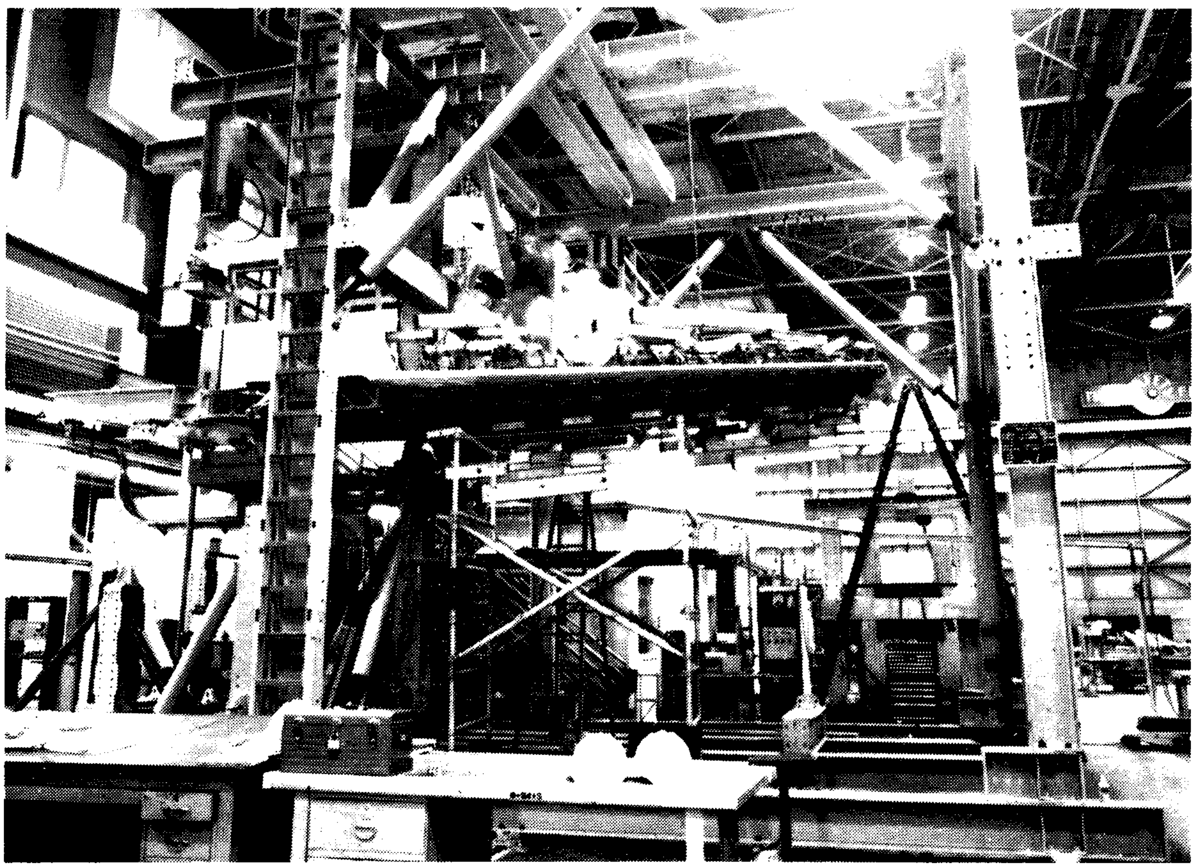
Ancillary Test Plan

The coupon, structural element, and subcomponent tests that were accomplished in the ancillary test program are listed here.



Full-Scale Ground Test Setup

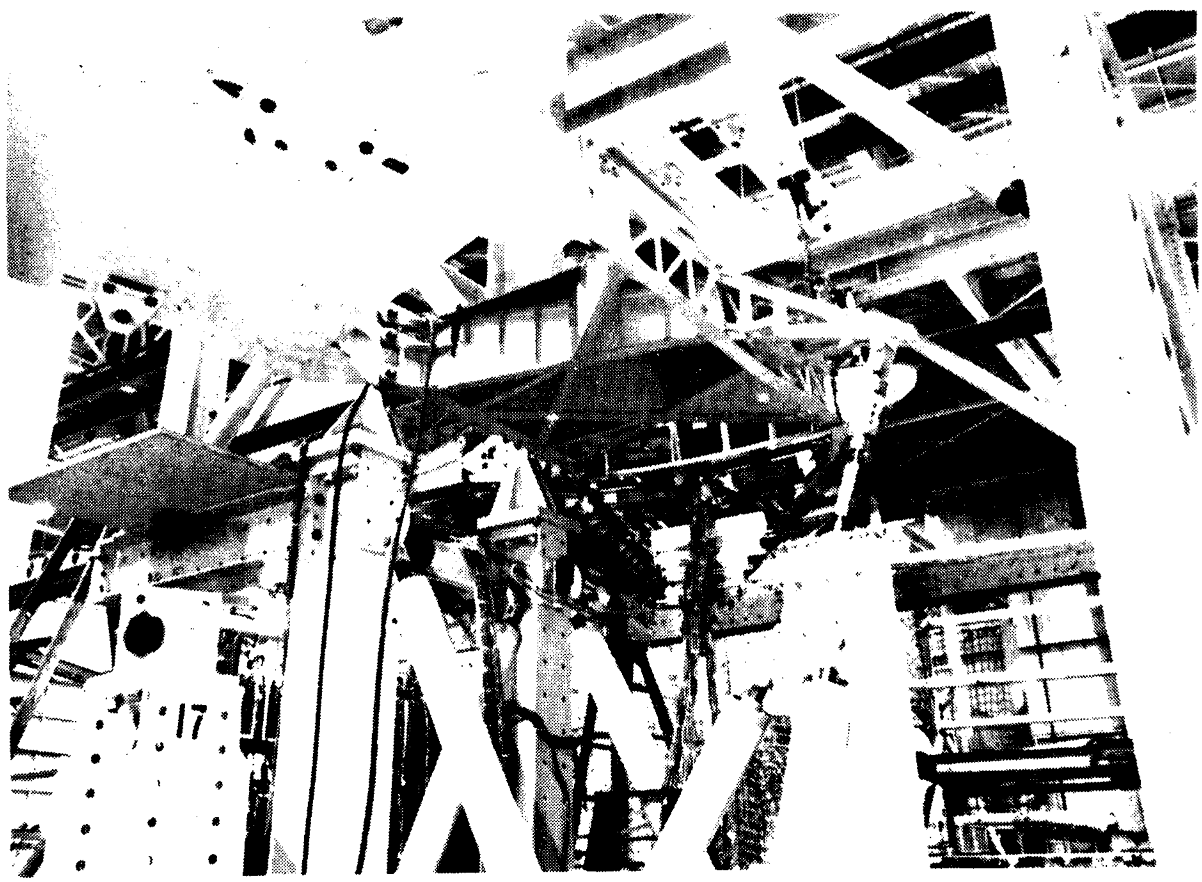
This photo shows the full-scale test specimen mounted in the support jigs. Loads were applied by a system of pads to simulate spanwise and streamwise load distribution.



BOEING

Full-Scale Test Specimen Mounting Structure

This photo shows the stabilizer's center section interface. Attachment of the stabilizer is made with five bolts: three at the rear spar and two at the front spar.



BOEING

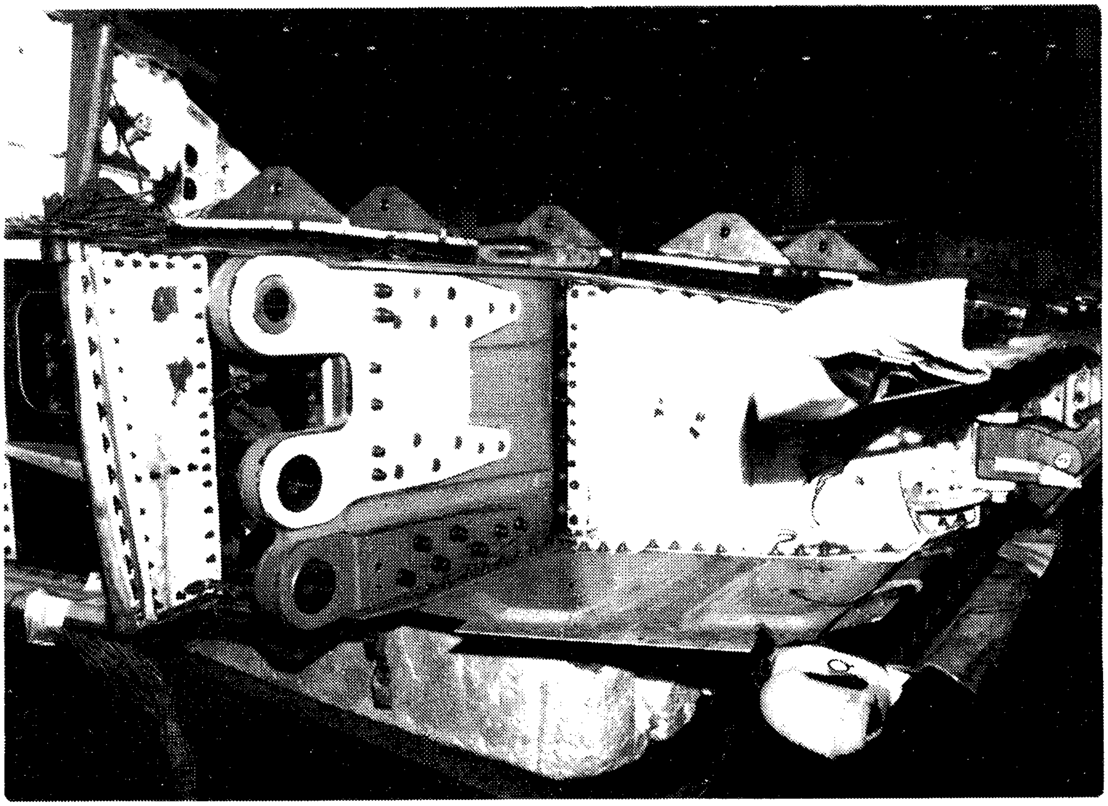
Test Configuration
Failure

- Removed Pin in Upper Lug Rear Spar
  - Simulated Center Section Failure
- Applied Load Case 4010 (Down Bending)
  - 67% DUL Required
- Failure Occurred at 61% DUL

BOEING

Full-Scale Test Specimen

This photo shows the test article with the repair installed.

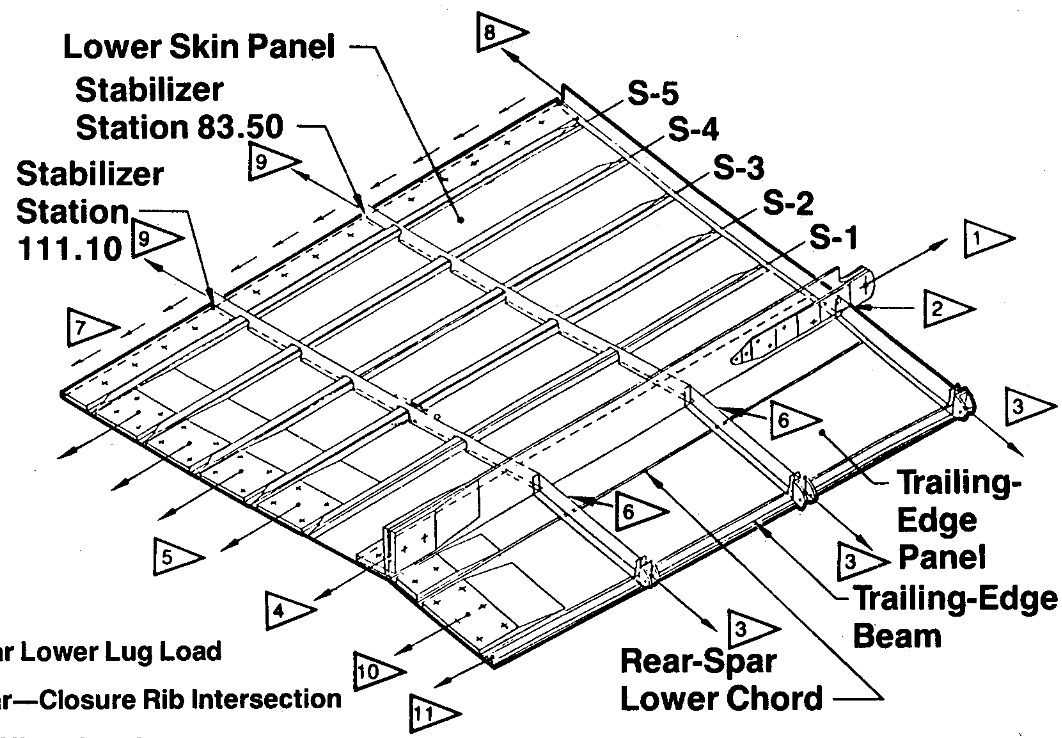


BOEING

Environmental Test Panel

Stabilizer Rear-Spar Chord and Skin Panel

The environmental test panel was representative of critical structure at the rear-spar inboard trailing edge. The panel was subjected to biaxial loading. Testing was conducted at ambient temperature and humidity, cold temperature and ambient humidity, and at elevated temperatures and wet environment. All tests were successful.



1 Rear-Spar Lower Lug Load
2 Rear Spar—Closure Rib Intersection
3 Elevator Hinge Loads
4 Spar Chord Load
5 Skin Panel Load
6 Trailing-Edge Rib Spar Intersection
7 Skin Panel Shear Load
8 Closure Rib Chord Reaction
9 Rib Cord Reaction
10 Trailing-Edge Panel Load
11 Trailing-Edge Beam Load
12 As Received After Manufacturing
13 Exposed to 140°F and 80 to 85% Relative Humidity Until Weight Stabilizes as Determined by a Moisture Rider

Test Condition	Environmental Condition		Load
	Temperature	Humidity	
1.	Ambient (70°F)	Ambient 12	67% DUL
2.	-75°F	Ambient	67% DUL
3.	180°F	Wet 13	100% DUL

BOEING

Structural Inspection
737 Composite Horizontal Stabilizer

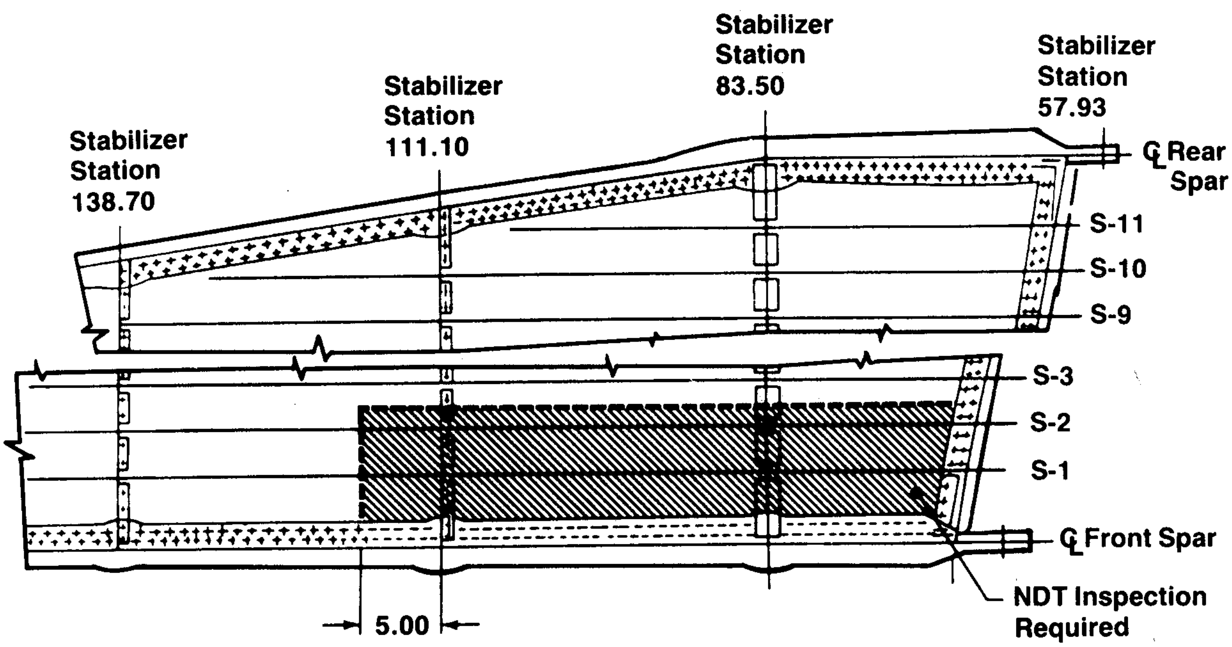
Essential to any production hardware program is early recognition of the need to develop a maintenance plan. The plan for the 737 horizontal stabilizer includes early structural inspection of the first two airplanes to reach approximately 7000 hours of service.

- **Inspect First Two Units at 7000 hr**
- **Inspect at Normal Structural Inspection Intervals ($\sim$ 14,000 hr) per D6-46036**
- **Inspection Includes NDE of Inboard Skins at Rear Spar**

BOEING

NDT Inspection Requirements
Upper and Lower Skin Panel

An inspection procedure that uses pulse-echo ultrasonic equipment was developed for the graphite stabilizer. The specific area requiring this inspection is shown.



BOEING

Design and Certification Requirements

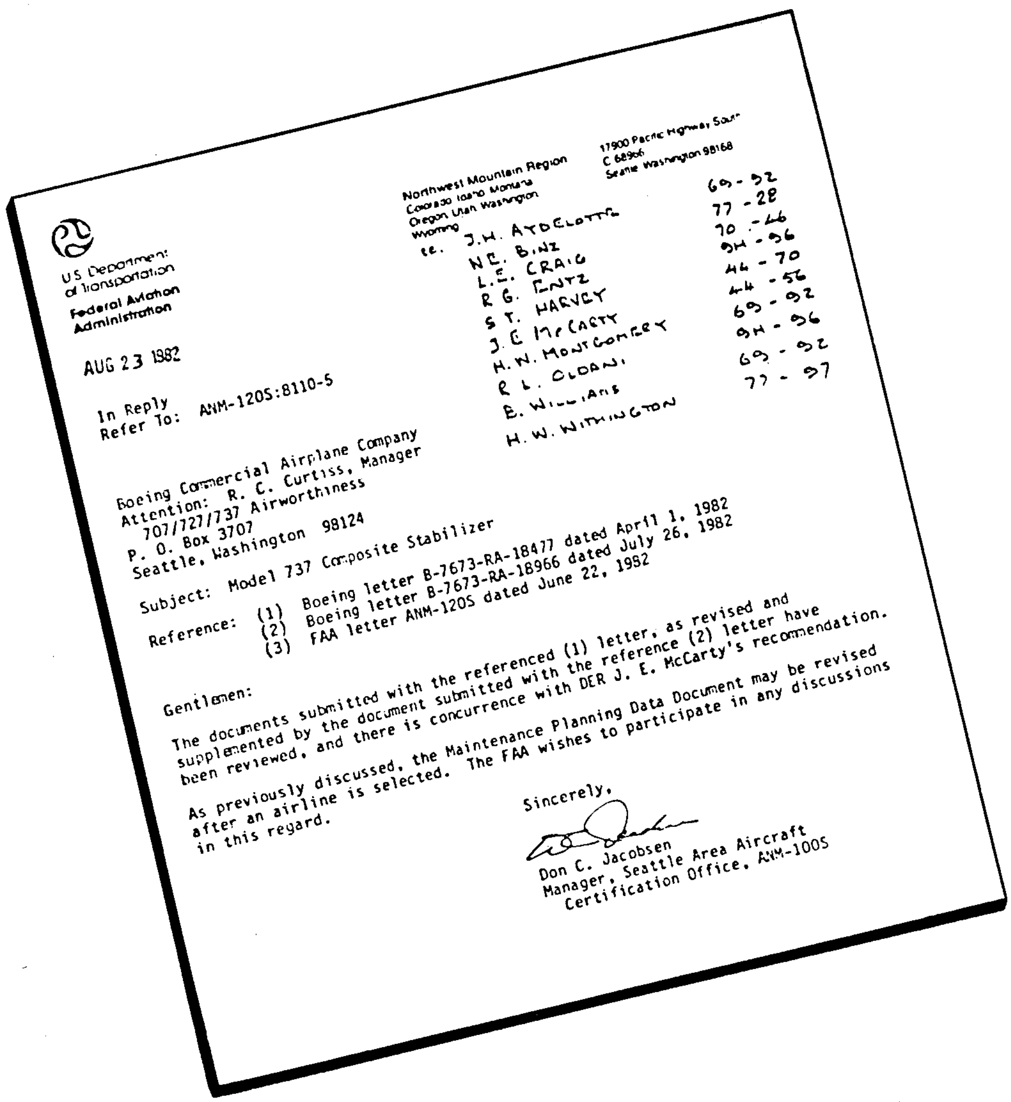
Certification requirements for commercial transport aircraft are defined in FAR Part 25. FAA Advisory Circular 20-107, issued in July 1978, sets forth recommended means of compliance with the provisions of the regulations. These two documents apply directly to the 737 horizontal stabilizer program.

- **Detail Boeing Design Requirements**
- **FAA Certification Requirements**
  - **FAR Part 25**
- **FAA Advisory Circular**
  - **AC No. 20-107**

BOEING

Certification Letter

Certification of the model 737 graphite-epoxy composite stabilizer was received in August 1982.



U.S. Department of Transportation
Federal Aviation Administration

Northwest Mountain Region
Colorado Idaho Montana
Oregon Utah Washington
Wyoming

17900 Pacific Highway South
C 68966
Seattle Washington 98168

AUG 23 1982

In Reply
Refer To: ANM-120S:8110-5

cc. J.H. AYDELOTTE 69-92
N.E. BINZ 77-28
L.E. CRAIG 70-46
R.G. FNTZ 94-96
S.T. HARVEY 44-70
J.E. McCARTY 44-56
H.W. MONTGOMERY 69-92
R.L. OLDANI 94-96
E. WILLIAMS 69-92
H.W. WITHINGTON 77-97

Boeing Commercial Airplane Company
Attention: R. C. Curtiss, Manager
707/727/737 Airworthiness
P. O. Box 3707
Seattle, Washington 98124

Subject: Model 737 Composite Stabilizer

Reference: (1) Boeing letter B-7673-RA-18477 dated April 1, 1982
(2) Boeing letter B-7673-RA-18966 dated July 26, 1982
(3) FAA letter ANM-120S dated June 22, 1982

Gentlemen:

The documents submitted with the referenced (1) letter, as revised and supplemented by the document submitted with the reference (2) letter have been reviewed, and there is concurrence with DER J. E. McCarty's recommendation.

As previously discussed, the Maintenance Planning Data Document may be revised after an airline is selected. The FAA wishes to participate in any discussions in this regard.

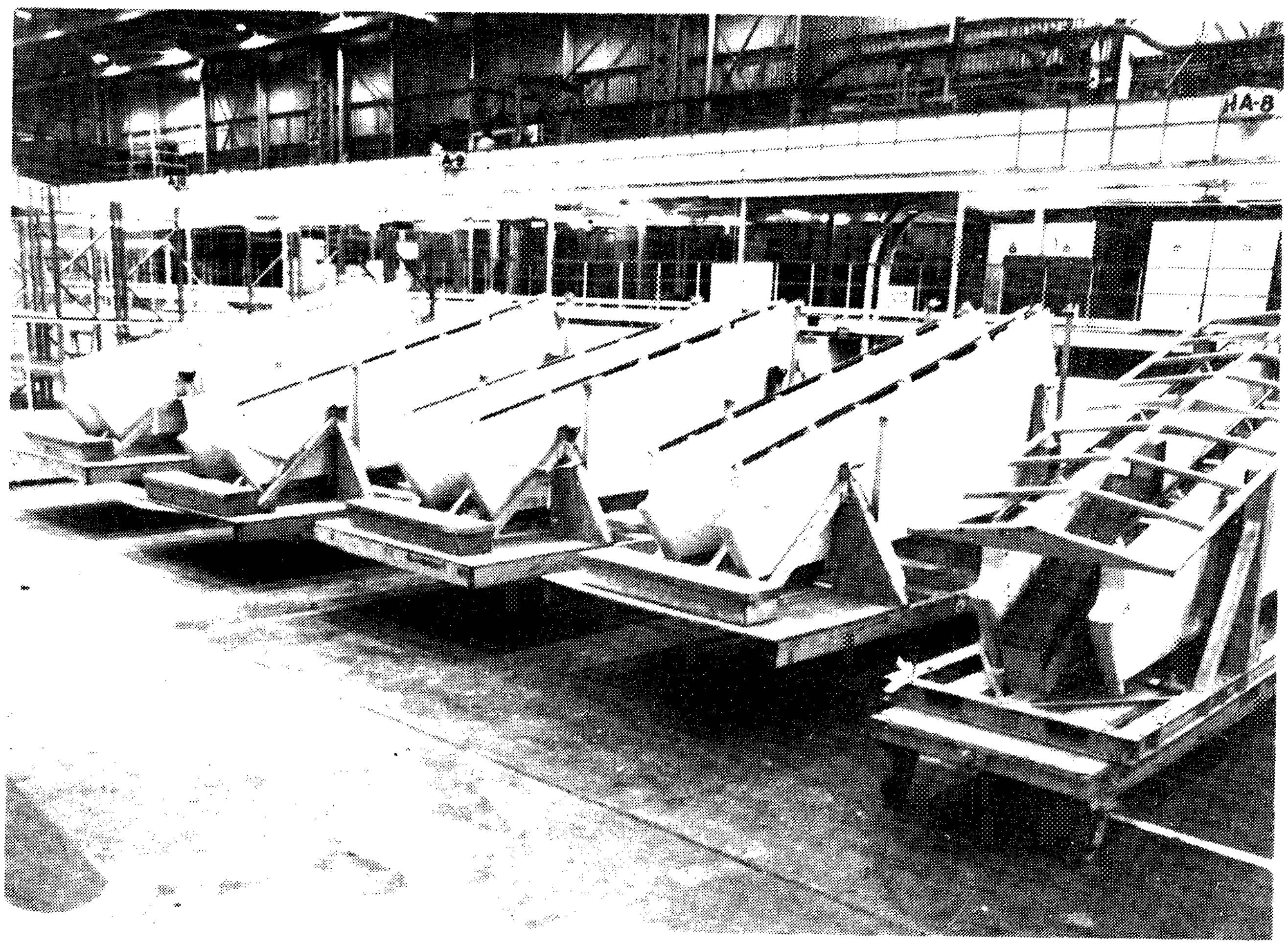
Sincerely,

Don C. Jacobsen
Manager, Seattle Area Aircraft
Certification Office, ANM-100S

BOEING

Graphite-Epoxy Stabilizers
Production Shipsets

Five production shipsets of graphite-epoxy stabilizers were fabricated.



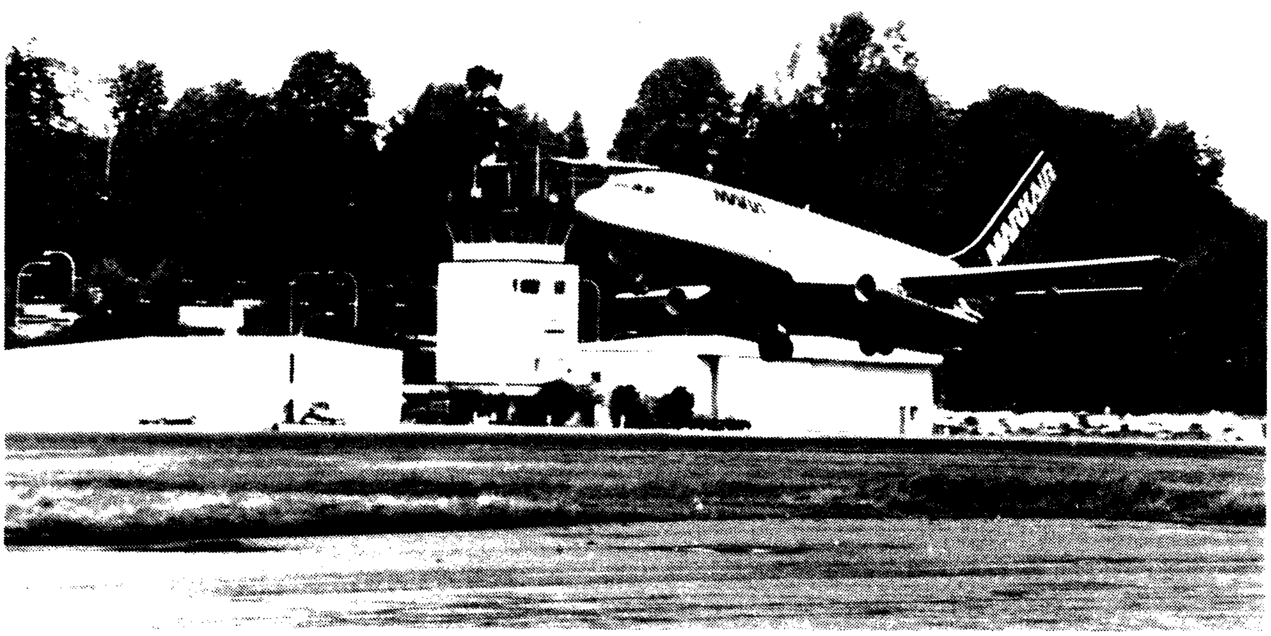
737 Graphite-Epoxy Stabilizer
First Installation

The first installation of the 737 graphite-epoxy composite stabilizer is shown.

BOEING

737 Graphite-Epoxy Stabilizer
In Commercial Service

All five shipsets of composite stabilizers were in commercial service by August 1984.



BOEING

737 Graphite-Epoxy Horizontal Stabilizer

Five shipsets of flight hardware designed, produced, certified, and introduced into flight service.

BOEING

Summary

- Extensive Advanced Composite Usage Committed to Production — Secondary Structure
- 737 Horizontal Stabilizers Entering Service
- Future Designs: Significantly Increased Advanced Composite Usage

BOEING

Composite Wing Panel Durability and Damage Tolerance Technology Development

Presented at ACEE Composite Structures Technology Conference

Robert D. Wilson
Boeing Commercial Airplane Company
August 1984

BOEING

Composite Wing Panel Durability and Damage Tolerance Technology Development

Three areas of recent NASA-sponsored composite wing panel durability and damage tolerance technology development at Boeing are presented for discussion. The development goals of structural efficiency and cost reduction are essential to the incorporation of advanced composites in wing primary structures. A 1983 status of panel design and subsequent toughened materials evaluation will be discussed.

- Wing Panel Technology Development Goals
- 1983 Baseline Panel Design
- Toughened Materials Evaluation

BOEING

Wing Panel Technology Development Goals

Weight reduction has been a primary goal in the use of composites. The associated benefits of increased durability (fatigue life) and less corrosion potential are valuable characteristics of composite structure. The requirements of damage tolerance and the need for cost-efficient manufacture of composite structures have been key drivers in design considerations.

Structural Efficiency

- Weight
- Durability
- Damage Tolerance

Cost Reduction

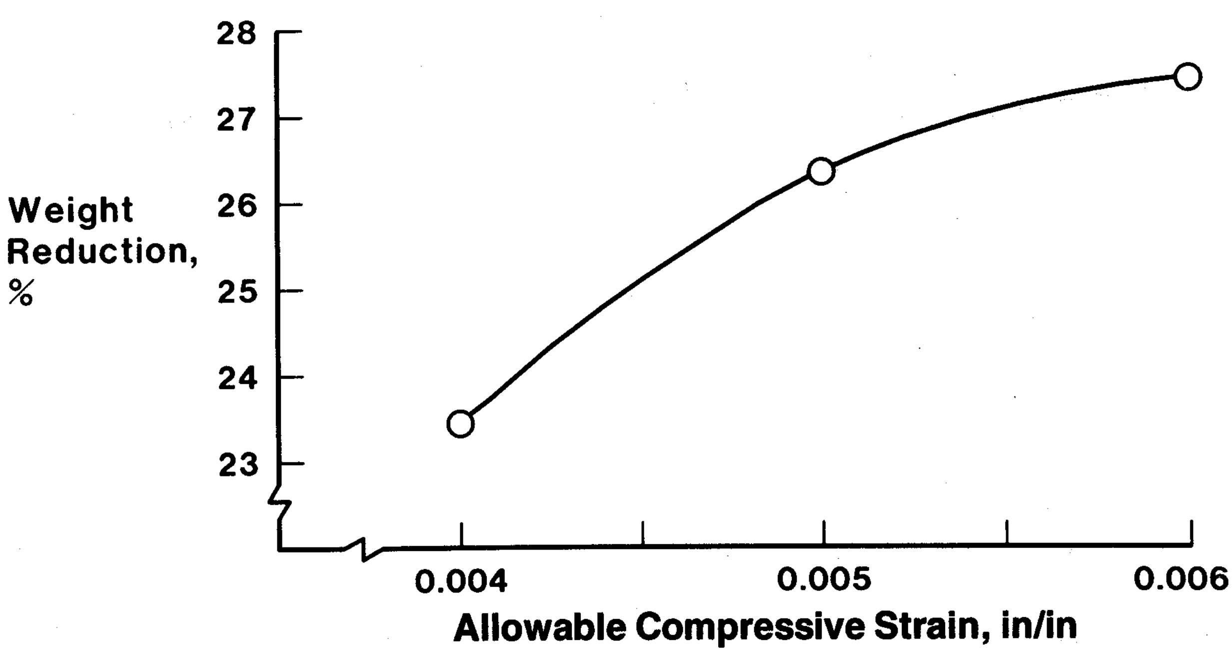
- Manufacture
  - Automation
  - Part Count Reduction
- Service
  - Corrosion Minimized
  - Fatigue Problems Minimized

BOEING

Wing Panel Technology Development Goals

Weight

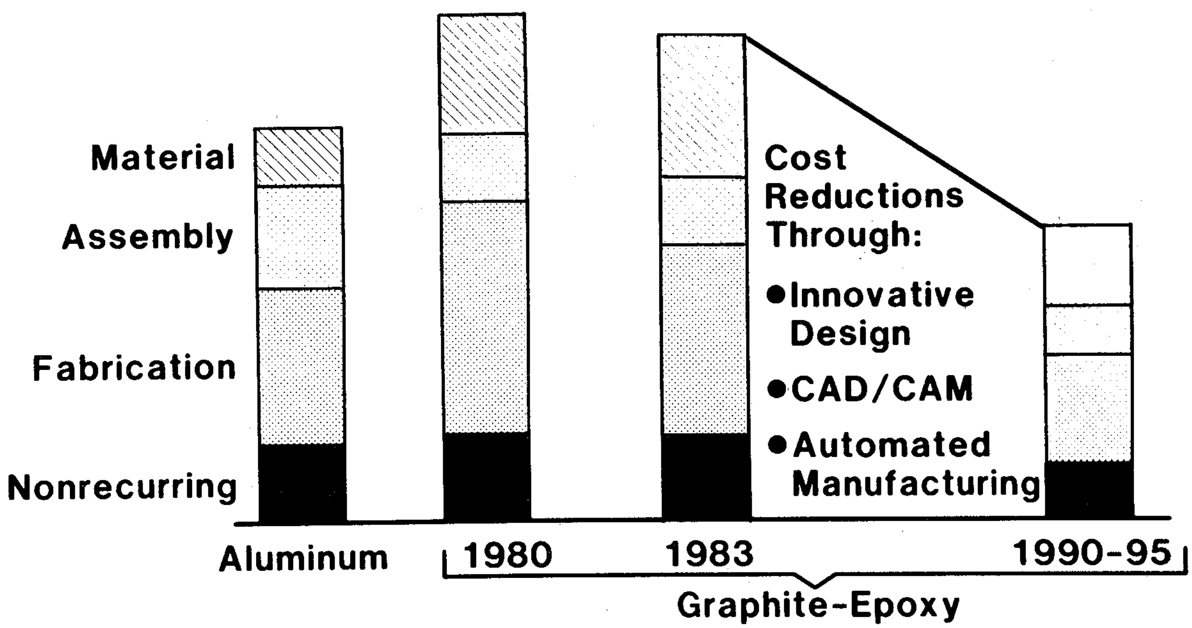
Apparent wingbox weight reduction, compared with current technology aluminum, is presented as a function of design allowable strains. Typically, a strain of 0.0060 in/in is achievable for tension panel design and is a constant in the curve shown.



BOEING

Typical Wing Structure
Comparative Cost

Relative costs of current aluminum versus graphite-epoxy wing production are shown. Improvements in materials, design configurations, and manufacturing methods have continued to bring costs down. Projected costs for the 1990-95 time period are very encouraging for composite wing structure.



BOEING

Wing Panel Technology Development Goals

Durability/Damage Tolerance

Composites' sensitivity to damage has required careful consideration of design details. Minimizing delamination tendencies to prevent damage initiation is a design goal. Apparent cyclic load characteristics of composites have led to a possible no-damage growth design criteria. For maximum weight reduction, high strain designs must be achieved. Design details must resist damage. Containment of possible impact damage at the event and arrestment of damage propagation, if it occurs, must be inherent in the design.

Durability

- Minimize Damage Initiation and Growth
- High Cyclic Life

Damage Tolerance

- High Strain Design
- Damage Resistance
- Damage Containment and Arrestment

BOEING

ACEE NASA

1983 Baseline Panel Design

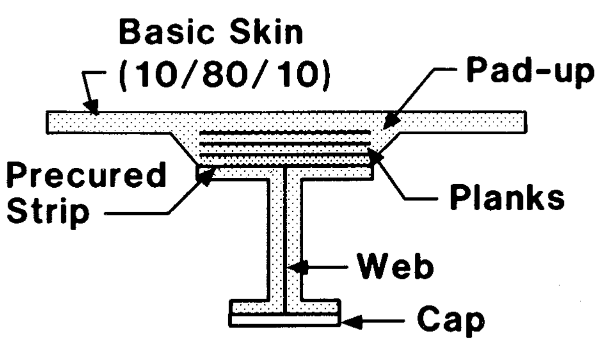
The 1983 baseline panel design was derived from NASA-sponsored work at Boeing and previous Boeing IR&D. A discussion of the structural design features, materials, durability and damage tolerance characteristics, and design strain capabilities is presented in the following charts.

- Design Features
- Materials
- Durability
- Damage Tolerance
- Strain Capabilities

BOEING

1983 Baseline Panel Design
Design Features

The NASA-sponsored development program considered a baseline airplane from which design loads and stiffness requirements were established. The design panel end load is 30 kips/inch. The wingbox torsional stiffness requirements is 1200 kips/inch (skin shear modulus x skin thickness). For damage tolerance considerations, the basic skin extensional modulus is kept low (E_{sk} = 5 Msi). Local 0-degree planks combine with the high modulus stiffener to form the primary load carrying area. The stiffener is formed by back-to-back channels cobonded with the skin through a precured strip interface.



Skin/Stiffener

- 30 kips/in
- Gt = 1200 kips/in
- E_{skin} = 5 msi
- E_{stiff} = 12.5 msi

1983 Baseline Panel Design
Material Selection

The 1983 baseline panel material was AS6/2220-3. This material combined a toughened resin system with a high strain fiber. The resin system had demonstrated good interlaminar toughness as exhibited by compression strength after impact events. It also demonstrated good hot/wet compression strength properties suited to commercial transport aircraft design temperatures. The fiber had demonstrated a strain capability of 1.7 percent while maintaining a current modulus of 36 Msi.

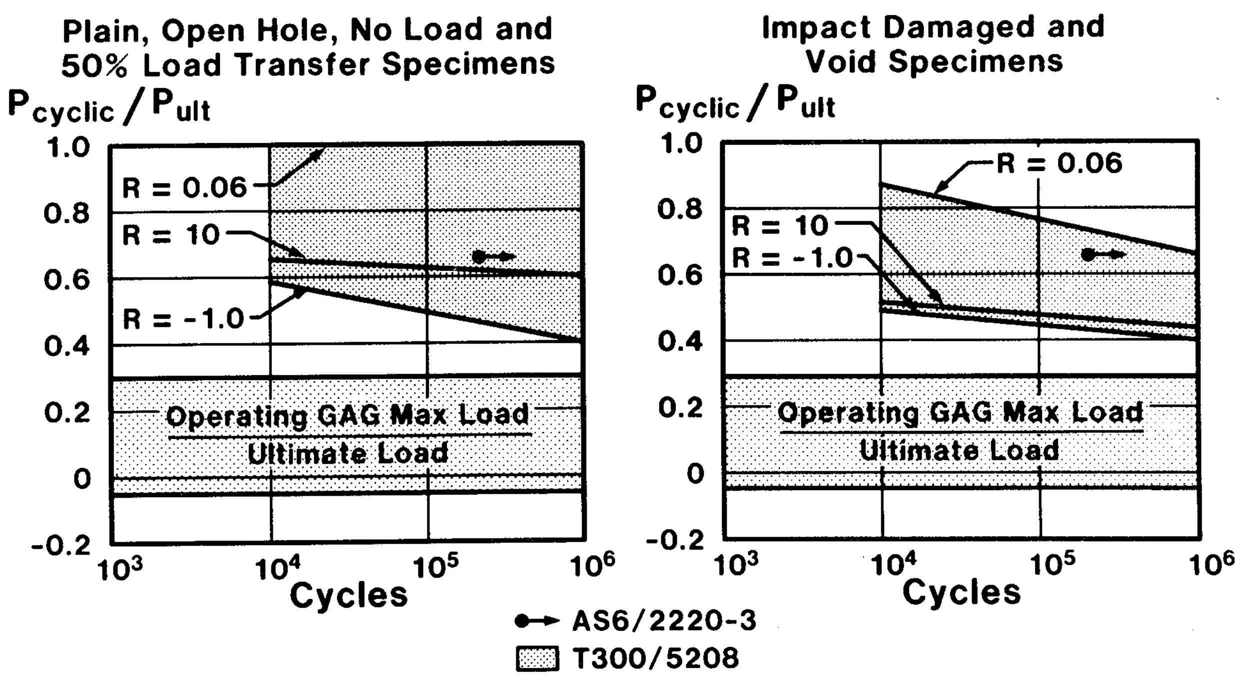
- Hercules AS6/2220-3
- Resin Characteristics
  - Interlaminar Toughness
  - Hot/Wet Compression
- Fiber Characteristics
  - Increased Fiber Strain
  - Retention of Current Modulus

BOEING

ACEE NASA

1983 Baseline Panel Design
Durability

Composites have demonstrated high cyclic load capability as shown in the accompanying typical curves for various coupon configurations and stress ratios. These data are for T300/5208 and are typical for current high strain materials. The typical cyclic load spectra for commercial transport wings, as a percentage of the design ultimate load, are well below the apparent cyclic load ratios that demonstrate a long cyclic life in composites. An AS6/2220-3 coupon test is referenced for comparison to typical coupon data.



BOEING

1983 Baseline Panel Design
Damage Tolerance

The 1983 baseline design features were reviewed previously in this presentation. In the event of impact damage or a "cut" skin between stiffeners, the load to be redistributed is minimized due to the small percentage (10 percent) of 0-degree fibers in the basic skin. Local concentrations of 0-degree fiber are placed in the skin " padup" at the stiffener. This padup area reduces the effects of local impact delamination and acts as damage tolerance material to arrest damage and redistribute load. All major elements of the stiffener and stiffener-to-skin interface are closely matched for Poisson's ratio to minimize strain incompatibility.

- "Soft" Skin
- Skin Pad-up
  - Imbedded Uniaxial Material
  - Reduce Impact Delamination
  - Arrest Damage
- Optimized Transition Between Adjacent Elements

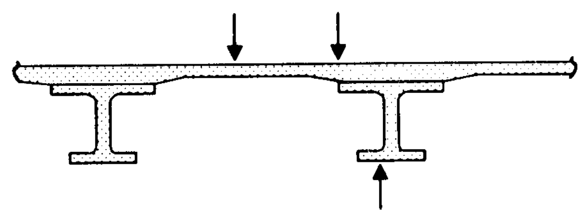
BOEING

1983 Baseline Panel Design
Damage Tolerance

Wing skins are exposed to many types of impact damage ranging from tools and runway debris to ground handling and hail. Critical locations for impact are shown in the sketch. The panel strength is most affected by the presence of damage in these areas. Load requirements vary depending on the severity and detectability of the damage.

Damage Exposure
- Manufacture
  - Tool Drop
  - Handling
- Service
  - Tool Drop
  - Debris
  - Hail
  - Ground Equipment

Damage Extent
- Barely Visible
- Easily Visible
- Severe Damage

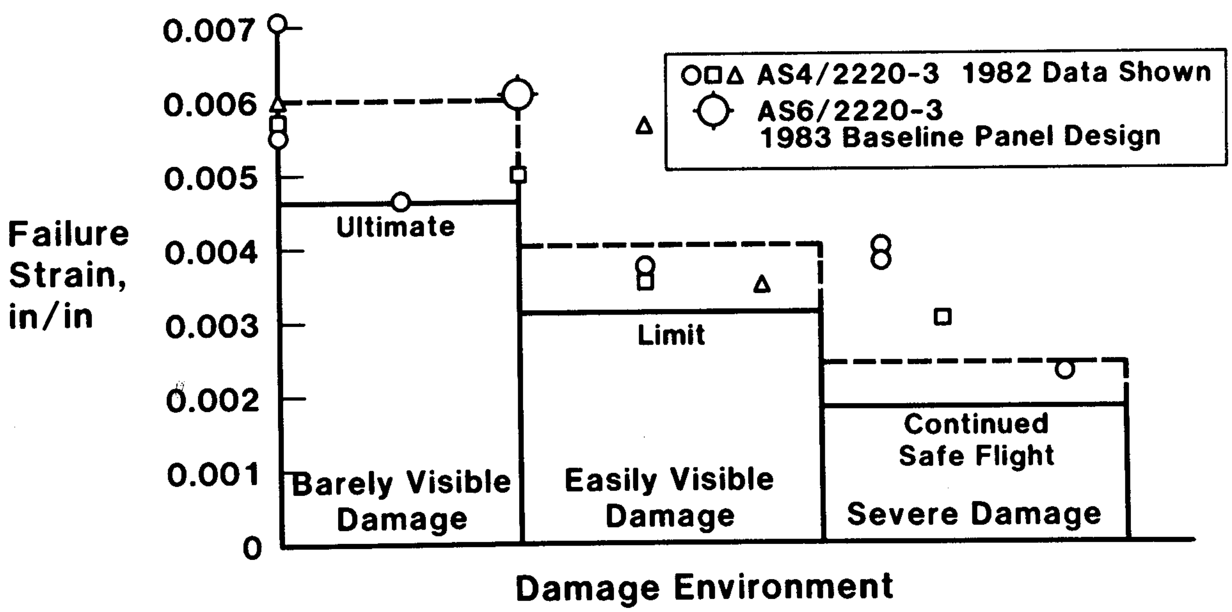


Critical Damage Locations

BOEING

1983 Baseline Panel Design
Strain Capabilities

The design ultimate strain goal is 0.0060 inch/inch. Limit load would be 2/3 ultimate (0.0040 inch/inch) and continued safe flight is approximately 40 percent of ultimate. Previous 1982 designs achieved approximately 75 percent of these strain goals. The 1983 baseline panel design has demonstrated, at the small test panel level, approximately 100 percent of the design ultimate strain goal.



BOEING

ACEE NASA

Toughened Materials Evaluation

As part of further design improvement, toughened materials were evaluated for use in large panel validation tests. NASA standard test coupon configurations and test procedures were used to screen materials. In addition, material coupon samples were damaged and load cycled to evaluate damage growth. Through the thickness, stitching was evaluated to measure improvements in impact damage containment and effects on damage growth.

- Candidate Materials
- NASA Standard Tests for Toughened Materials - Static Loads
- Damage Growth Tests - Cyclic loads
- Enhanced Design - Stitching

BOEING

Toughened Materials Evaluation

Candidate Materials

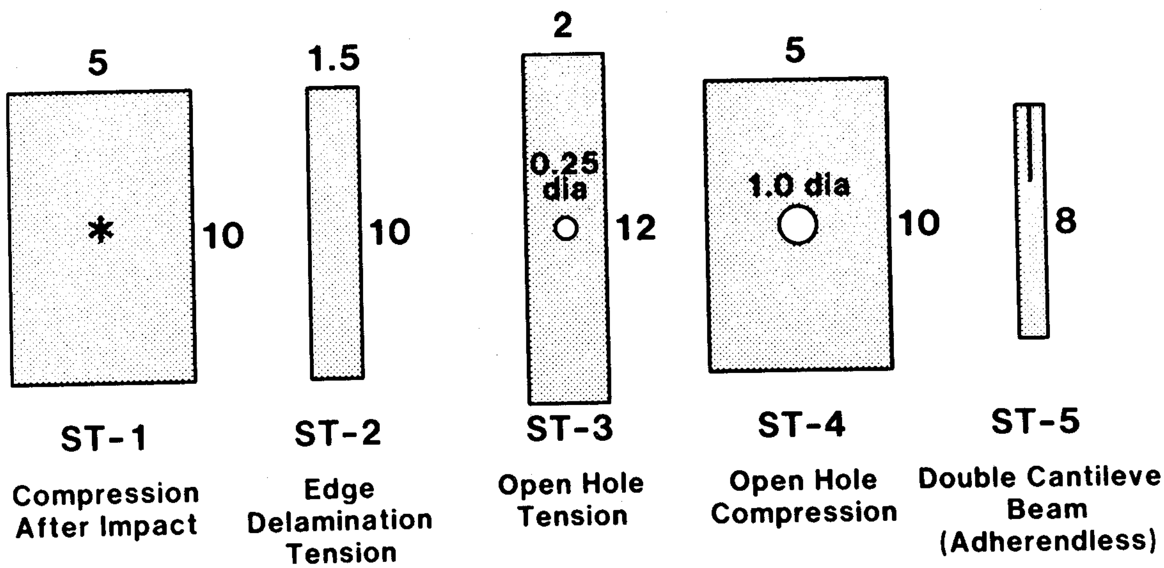
The 1983 baseline toughened material was AS6/2220-3. This selection was based on the previous performance of AS4/2220-3 and the increased strain capability of the AS6 fiber. Newer toughened materials for evaluation included thermoset and thermoplastic.

- 1983 Baseline Toughened Material
  - AS6/2220-3
- New Toughened Materials
  - AS6/5245C
  - AS4/5245C
  - AS4/Peek

BOEING

Toughened Materials Evaluation
NASA Standard Tests for Toughened Resin Composites NASA RP 1092, May 1982

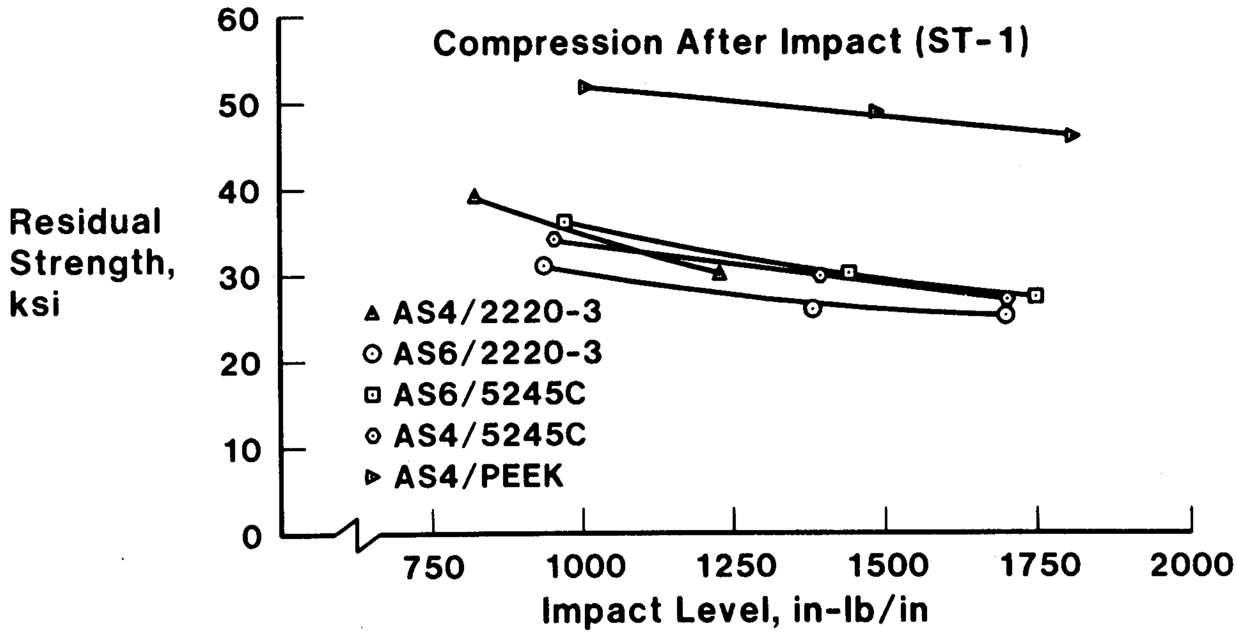
The NASA Standard Tests For Toughened Resin Composites (NASA RP 1092, May 1982) was used to evaluate the newer toughened materials for use on large validation test panels. These standard tests were mutually agreed upon among NASA, Boeing, Douglas, and Lockheed for use in the wing key technology contracts.

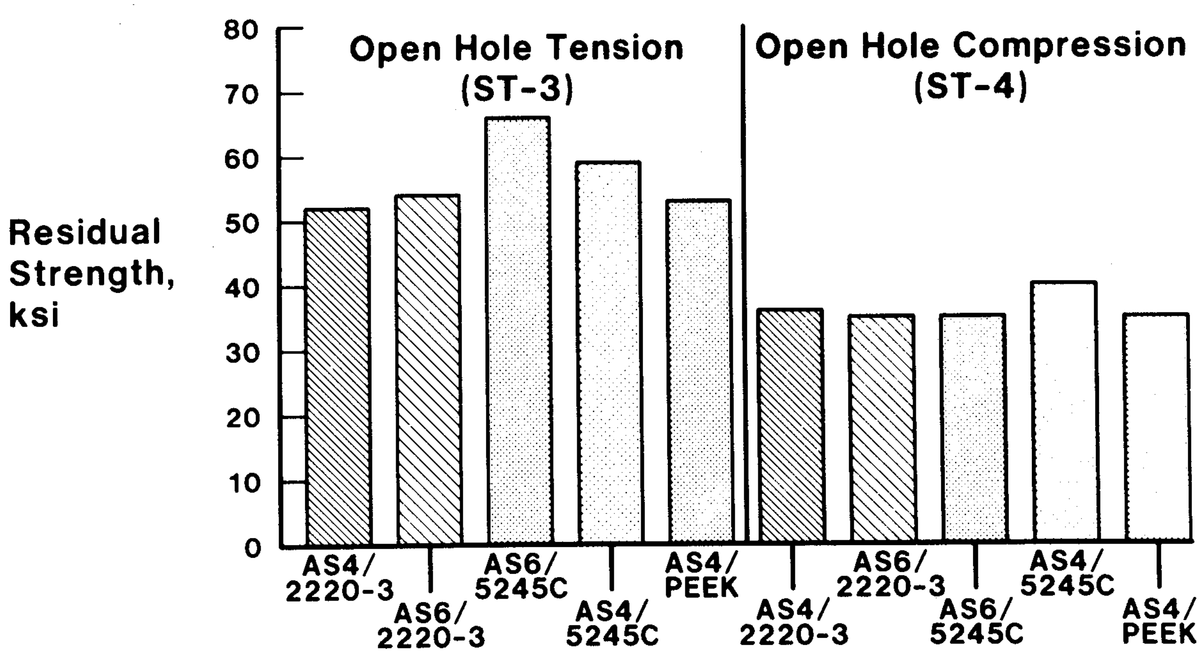


BOEING

Toughened Materials Evaluation
NASA Standard Test Results

Three NASA standard specimen test results are shown for the materials evaluated.





BOEING

ACEE NASA Damage Growth Coupon Tests

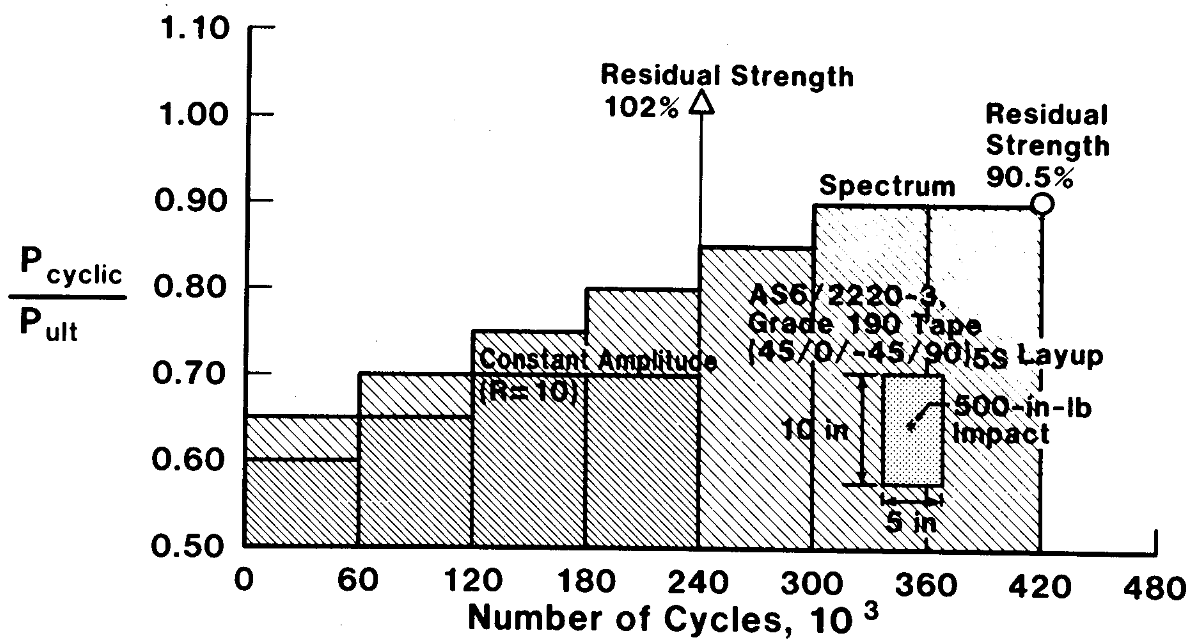
The AS6/2220-3 baseline material was tested for damage growth of five types of initial damage as shown. Two fatigue environments and three different strain levels were utilized. A compression-compression constant amplitude and compression-dominated wing spectrum loading at strain levels above 60 percent of static ultimate strength were used. Periodic inspections monitored damage growth.

- Damage Growth Evaluation of Baseline Material at Coupon Level
- Five Types of Known Initial Damage
  - Delaminations From 280-in-lb Impact Energy
  - Delaminations From 500-in-lb Impact Energy
  - Open Hole With Delamination Damage
  - Impact Delaminations in Stitched Enhancement
  - Multidelamination Simulation of Impact Damage
- Two Fatigue Environments
  - Constant Amplitude, R=10.0
  - Upper Wing Surface Spectrum
- Three Strain Levels
- Periodic NDI to Monitor Damage Growth

BOEING

Damage Growth Coupon Tests

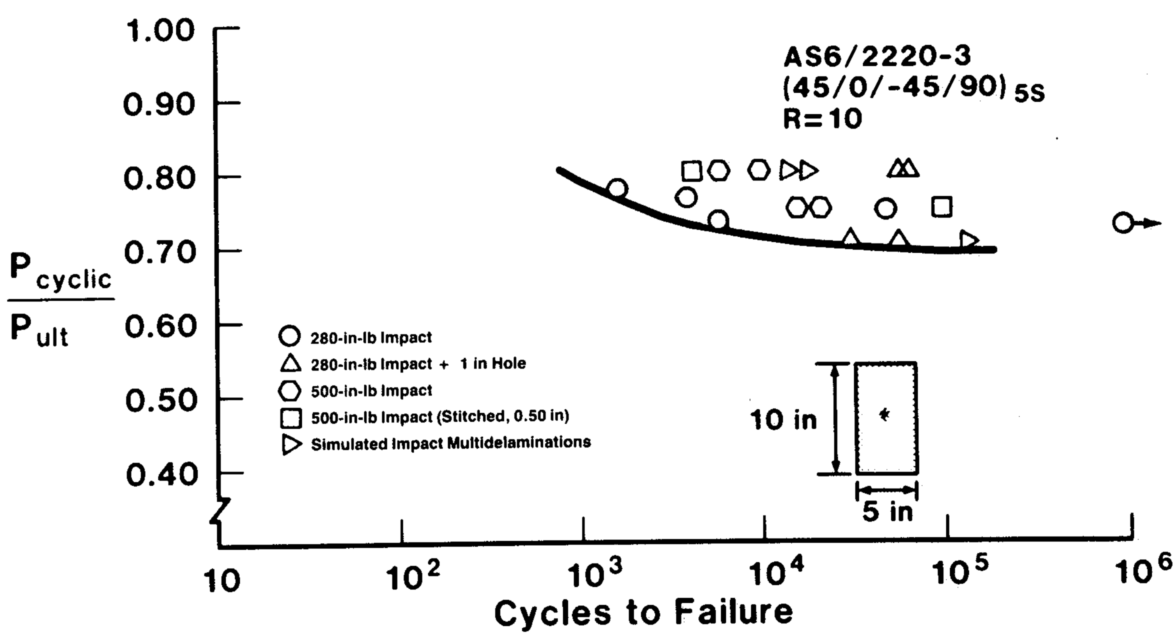
The data shown are two examples from the testing discussed on the previous page. One lifetime of service is approximately 60,000 cycles. One specimen was cyclic loaded at R = 10 in steps up to 70 percent of ultimate and four lifetimes. The residual strength at the time was 102 percent of ultimate. The other specimen was spectrum tested in steps to seven lifetimes. Residual strength was approximately 90 percent of ultimate.



BOEING

Damage Growth Coupon Tests

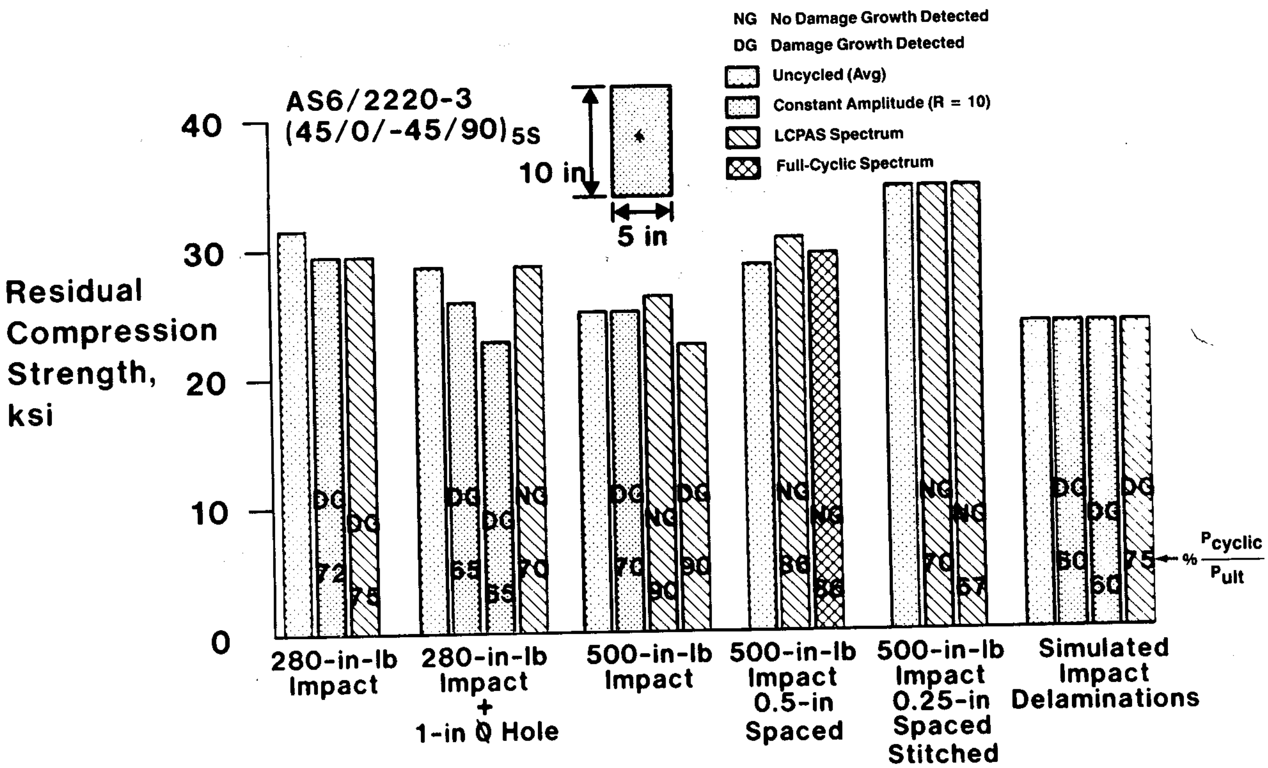
Various types of damaged specimens, as outlined on previous pages, were cyclic tested to failure at load levels of 70-percent static ultimate strength and above. These data demonstrate that cyclic loading is not a basic concern with these kinds of damaged composite specimens. As shown earlier, the typical commercial load spectra varies up to approximately 30 percent of design ultimate, which is well below test values discussed above.



BOEING

Damage Growth Coupon Tests

Specimens with damages, as shown, were cyclic tested for several lifetimes and then residual strength tested to static failure. These data show, in general, that static strengths after this cyclic testing are very close to original uncycled strengths.



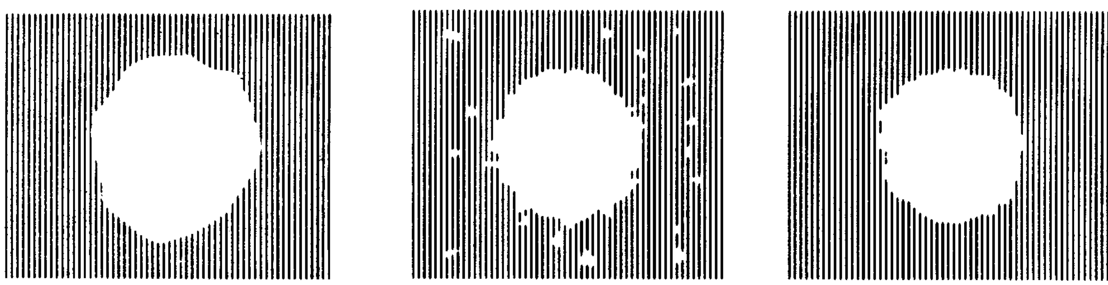
BOEING

Enhanced Design
Stitching

The effects of Kevlar stitching on the strength of compression after impact were evaluated using the NASA standard specimen. Stitching was placed in the axially loaded direction. Specimens were tested at room temperature. A row spacing of 0.25 inch demonstrated an apparent strength improvement of 41 percent.

TTU C-Scans of Impacted Specimens

Unstitched	Kevlar Stitched (4 Stitches/in) Row Spacing = 0.5 in	Kevlar Stitched (4 Stitches/in) Row Spacing = 0.25 in

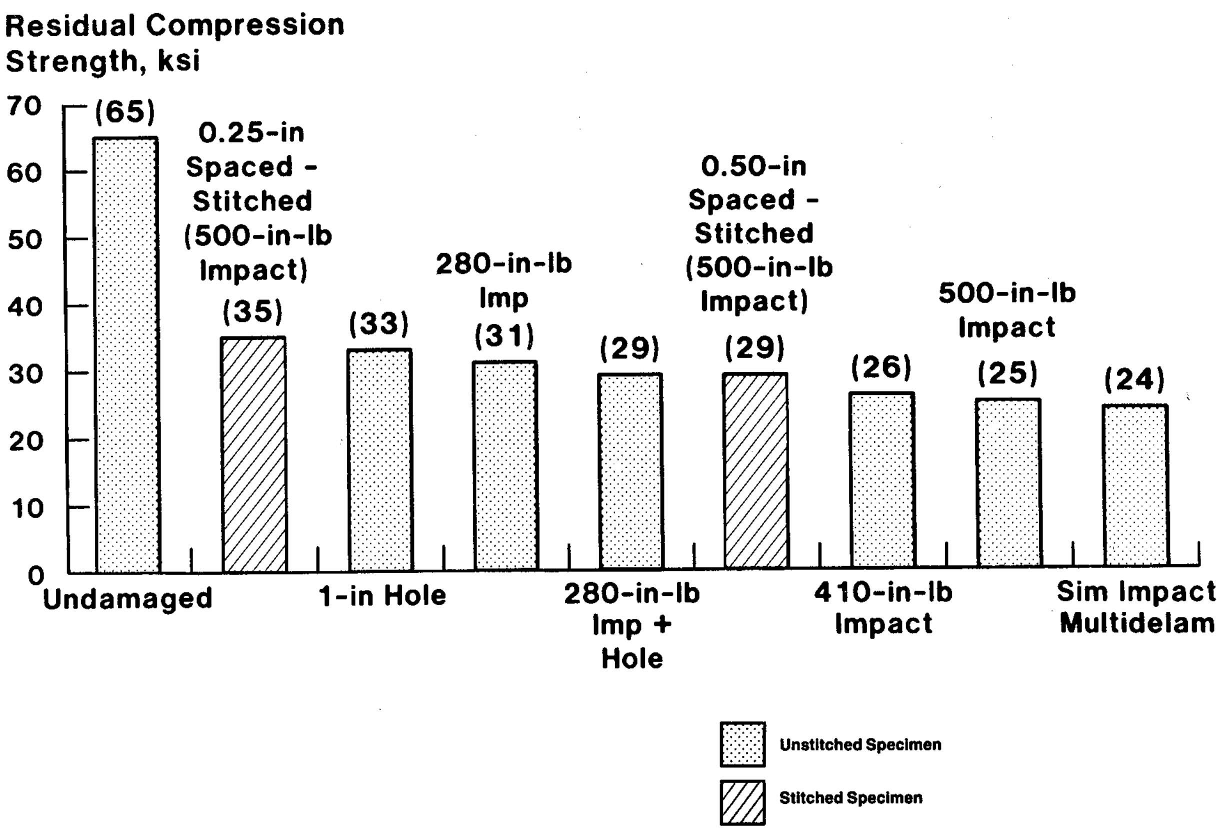


Unstitched	Row Spacing = 0.5 in	Row Spacing = 0.25 in
500 in-lb	500 in-lb	500 in-lb
25 ksi	28.8 ksi	35.4 ksi
4070 μin/in	4280 μin/in	5440 μin/in
———	+15.2%	+41.6%

BOEING

Residual Compression Strength Comparison

Impacted stitched and unstitched specimens' strength are compared with various specimen configurations. The material is the baseline AS6/2220-3. Tests were conducted at room temperature. The contribution of the stitching to the residual compression strength of the 500-in-lb impacted specimens is clearly evident in the comparison shown.



BOEING

Composite Wing Panel Durability and Damage Tolerance Technology Development Summary

Significant progress in composite wing panel durability and damage tolerance technology development has been made in recent years. Toughened materials, design improvements, and modern manufacturing methods have combined to produce weight- and cost-competitive hardware. The cyclic load life and resistance to service corrosion have been greatly enhanced, and the structures meet damage tolerance requirements.

- Technology Development Goals Being Reached
  - Weight
  - Cost
  - Durability
  - Damage Tolerance
- Current Composite Designs Competitive With Conventional Metal Designs
- New Toughened Materials and Design Enhancements Offer Further Gains

Design Development of Heavily Loaded Wing Panels

Presented at ACEE Composite Structures Technology Conference

Peter J. Smith
Boeing Commercial Airplane Company
August 1984

BOEING

ACEE NASA

Panel Development
Objectives

The principal objective of the Large Composite Primary Aircraft Structure (LCPAS) program was to improve the then current composite wing panel design capability. The initial phase of the program focused on the critical upper surface and demonstrated improved design load capability. The goal of Phase II of the LCPAS program was to continue design improvements and achieve a 30 percent weight reduction from aluminum on compression-loaded panels. This goal required the structure to be capable of ultimate design loads at 0.006-inch/inch strain or 50-ksi stress with critical damage.

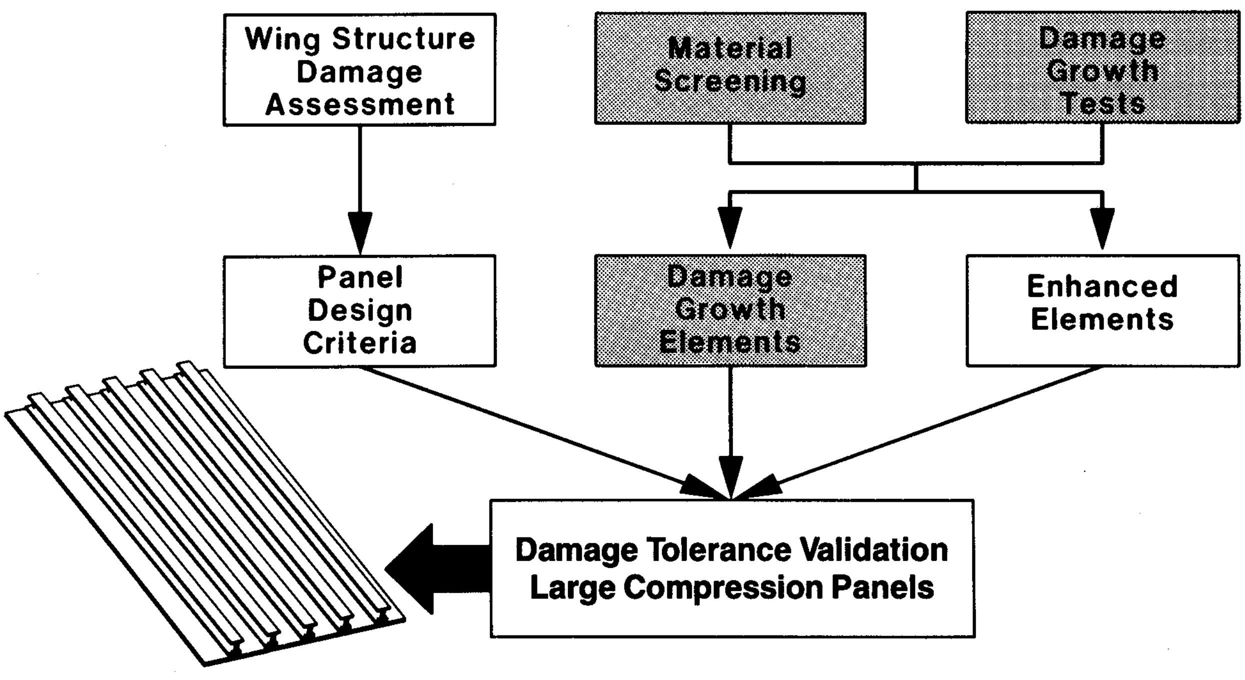
A Phase II mandate required an assessment of the wing structure damage environment so the final program objective could be achieved. This objective was to verify the final damage tolerance panel design through a series of large five-stiffener panel tests.

- Improve Compression Panel Design Capability
- Assess Wing Structure Damage Environment
- Large Panel Verification of the Damage Tolerance Design

BOEING

Damage Tolerance Panel
Development Program

The damage tolerance panel development program was structured to allow a systematic progression toward the final large panel verification tests. This presentation will concentrate on the wing structure damage assessment study, the enhanced element tests, and the damage tolerance validation of the large compression panels. The material screening tests, damage growth tests, and damage growth element tests are outlined in the discussion of Composite Wing Panel Durability and Damage Tolerance Technology Development by Mr. R. D. Wilson.



BOEING

Design Goals

The design goals of the LCPAS program were to meet the requirements of a 1990's fuel-efficient, 200-passenger airplane with a high-aspect-ratio wing. The area of the wing selected for study on this program was the upper wing panel at the nacelle where the design end loads approach 30 kips/inch and the shear stiffness requirement was Gt = 1200 kips/inch. The design end loads were to be met with an average stress of 50-ksi and 0.006- inch/inch P/AE strain. The design loads were to be achieved without the skins buckling.

- 30 kips/in Compression End Load
- Torsional Shear: Gt = 1200 kips/in
- Ultimate Compression: 50 ksi and 0.006 in/in
- Skins Nonbuckling at Ultimate Load

ACEE NASA Lessons From 1983 Technology

The lessons from Phase I of the LCPAS program, the material screening tests, and damage growth element tests of Phase II are shown. The soft skins were demonstrated to be damage tolerant, and the I-stiffener allows structural efficiency with torsional stability and simplicity of fabrication. The 8-inch stiffener spacing provides an efficient compression panel geometry. The embedded 0-degree planks allow balanced loads in the skin and stiffeners and also add a damage containment feature by limiting delaminations. The Phase I testing and the material screening tests of Phase II indicated the need for a toughened material.

- Soft Skins: 10/80/10
- I-Stiffener Configuration
- 8-in Stiffener Spacing
- Embedded 0° Planks
- Balanced Loads in Skin and Stiffeners
- Toughened Material Needed

BOEING

ACEE NASA Wing Damage Assessment Study

One of the main objectives of Phase II of the LCPAS program was to perform an in-depth assessment of the wing structure damage environment. This required an investigation into the damage threats, manufacturing and quality control processes, service maintenance and inspection practices, and service histories of commercial airplane wings. This investigation was to focus on important aspects of quality control and damage detection and to provide timely conclusions for the design of the final damage tolerance panels.

- Investigate Damage Sources, Types, Locations, Severity, and Possibilities
- Review Manufacturing Processes and Quality Control Practices
- Review Service Maintenance and Inspection Practices and Service Histories of Commercial Airplane Wings
- Focus on Important Aspects of Quality Control, Damage Detection, and Significant Structural Design Criteria

BOEING

Wing Damage Threats

The wing structure damage assessment study investigated damage threats in both the manufacturing and service environment. Damage threats typically found in the manufacturing environment were impacts from dropped tools, foreign body inclusions, warpages due to curing strains, and voids/porosity delaminations due to material processing problems. The service damage types included impacts from dropped tools and vehicle contacts, gouges and scratches from tool, vehicle and gantry contacts, and natural hazards such as hail and lightning strikes.

Manufacturing Damage

- Impacts
- Foreign Body Inclusions
- Warpage
- Voids/Porosity
- Delaminations

Service Damage

- Impacts
- Gouges/Scratches
- Hail
- Lightning Strikes

BOEING

Service Maintenance and Inspection Methods

An investigation into typical service maintenance and inspection methods was performed on Boeing airplanes currently in service with the airlines. Typical service maintenance intervals, time spent on the wing skins, and inspection methods are shown. The investigation showed that little time is actually spent on the wing skins, particularly the upper surface skin, until the C or D checks, when methods other than visual inspection may be used.

- Typical Service Maintenance Intervals

	Service Interval, No. of Flights	Time Spent on Wing Skin, hr	Likely Inspection Methods
Transit	Each	0.1	Visual
Preflight	Daily	0.4	Visual
A Check	80	0.4	Visual
B Check	370	0.4	Visual, Ultrasonics Eddy-Current, X-ray
C Check	1300	4.0	Visual, Ultrasonics Eddy-Current, X-ray
Structural Check (or D Check)	13 000	140.0	Visual, Ultrasonics Eddy-Current, X-ray

BOEING

Wing Damage Assessment
Conclusions

The results of the wing structure damage assessment study have indicated that impact damage is the most critical damage threat. This kind of damage can cause delaminations in the brittle composite laminated structure. The composite design must be designed with damage containment features that will minimize the delaminations caused by impact; i.e., the structure may contain extensive delaminations with little visible indication. Therefore, adequate quality control during manufacturing and in-service inspection techniques are needed to maintain safe structure.

- Impact Damage Most Critical
- Designs Must Contain Damage Containment Features
- Focus on Manufacturing Quality Control
- Establish Service Inspection Techniques

BOEING

Wing Panel Design Criteria

Design criteria established by the FAA and Boeing require maintenance of a level of safety equivalent to that of the aluminum structure of current large transport airplanes. This means that the composite structure must carry ultimate loads with manufacturing defects and barely visible damage due to accidental impact caused in either manufacturing or service. Also, if in-service damage should occur, the remaining structure must be capable of carrying limit loads until the damage is detected by planned maintenance. This kind of damage is defined as easily visible damage. The final requirement is that if severe in-flight damage occurs that is obvious to the crew, then the remaining structure must be capable of carrying continued safe flight loads.

- Ultimate Load With Barely Visible Damage
- Limit Load With Easily Visible Damage
- Safe Flight Load With Obvious Partial Failure-Severe Damage

BOEING

Enhanced Panel Design

The enhanced panel design was developed utilizing the experience gained from the Phase I panel tests, the Phase II damage growth coupon and baseline element tests, and state-of-the-art damage containment concepts. The soft skin has continued to demonstrate good damage tolerance, and the damage growth coupon tests demonstrated improved damage containment and residual compression strength from grid stitching of laminates.

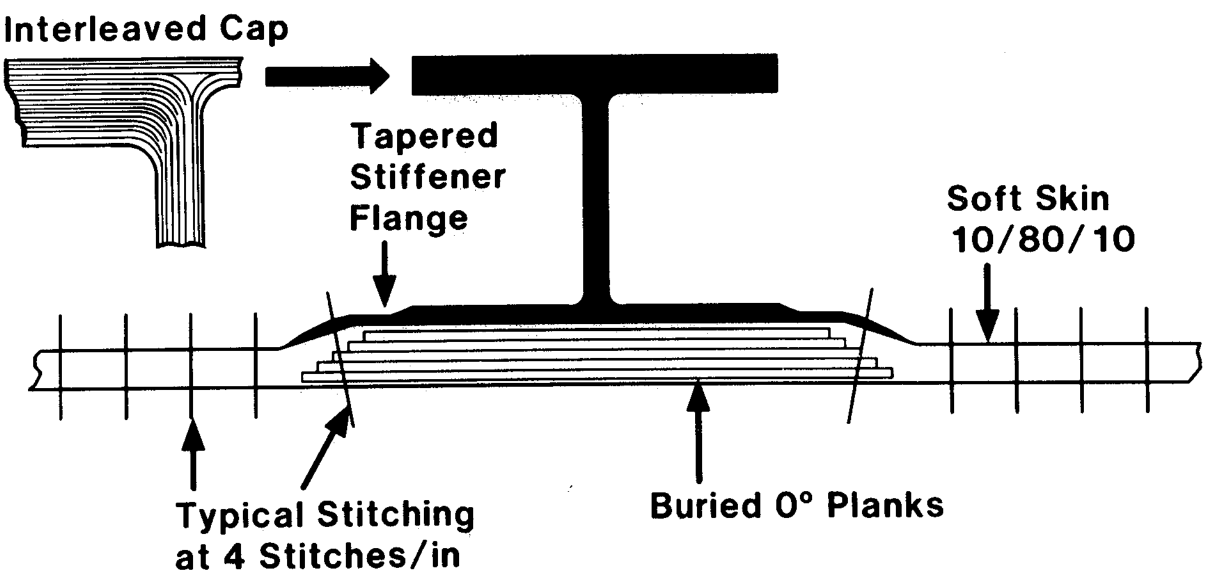
The wing structure damage assessment study highlighted the criticality of the stiffener cap, and the interleaving of the 0-degree plies with 45-degree plies was an effort to increase the damage tolerance of the stiffener cap. The interleaved stiffener cap allows for a slimmer stiffener web and more 0-degree planks buried in the skin. The stitching of the stiffeners to the skin is an effort to enhance this critical interface. The baseline element material of AS6/2220-3 was retained in order to provide a direct design comparison.

- Retain Soft Skin: 10/80/10
- Increase 0° Planks
- I-Stiffeners With Interleaved Cap and Tapered Skin Flanges
- Grid Stitching of Skins
- Stitch Skin/Stiffener Interfaces

BOEING

Enhanced Design
Damage Containment Features

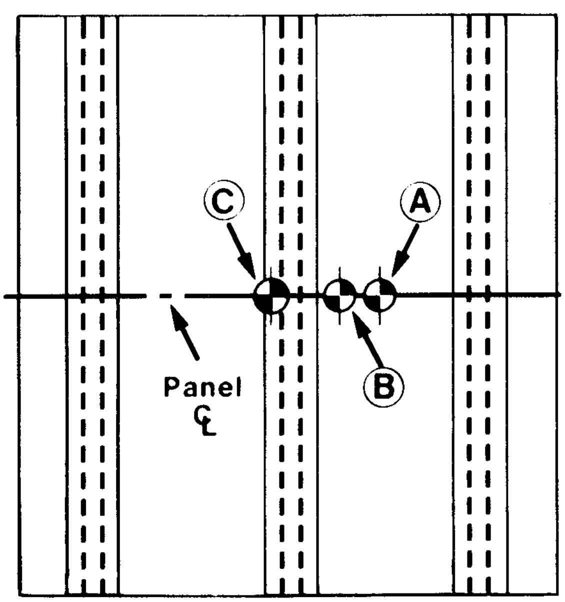
The enhanced design damage tolerance features are shown. The stiffeners feature tapered skin flanges that wrap down over the skin ramp. This allowed the stitching of the skin/stiffener interface while the stiffener tools were in place. The tapering of the stiffener flanges also enables the flanges to conform to the heavy skin ramp in the event that out-of-plane deformation takes place.



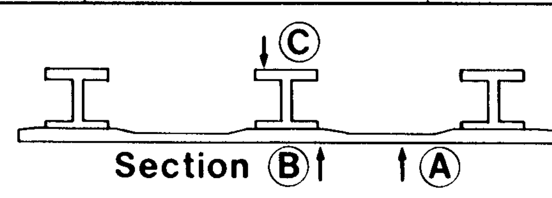
BOEING

Enhanced Element
Test Program

The enhanced element test program is shown. The program consists of testing three 3-stiffener panels in static compression to determine the ultimate load damage tolerance of the enhanced design. The panels were 25 inches long by 21 inches wide. The 1983 baseline panel tests had shown that durability was not as critical as damage tolerance, and the Phase I panel tests had demonstrated that the ultimate load case with barely visible damage was the most critical design case. The 1000-inch-pound impact damage to the skin is the Boeing standard of barely visible service damage, and the 400-inch-pound impact damage to the stiffener cap represented damage caused during manufacturing.



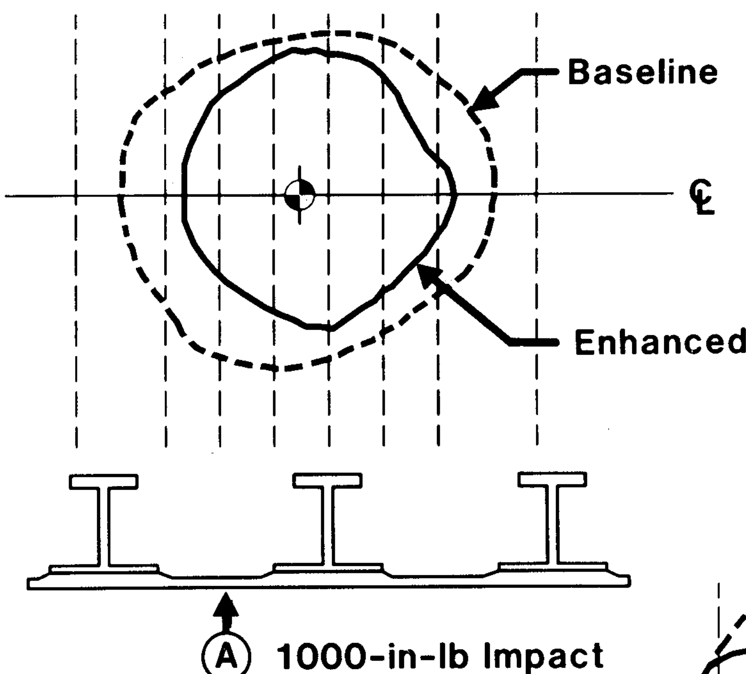
Panel No.	Damage Type and Site	Load Case
A	1,000 - in - lb Impact on Skin Barely Visible	Ultimate
B	1,000 - in - lb Impact on Skin Ramp Barely Visible	Ultimate
C	400 - in - lb Impact on Stiffener Barely Visible	Ultimate

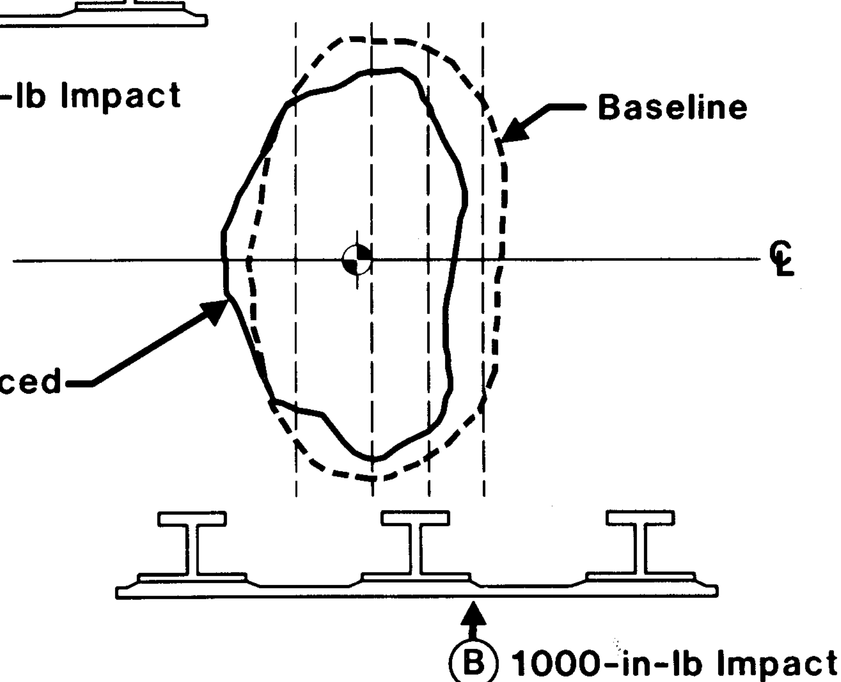


BOEING

Pulse Echo Indications of Impact Delaminations
Baseline vs Enhanced Designs

The results of the pulse echo inspections of the delamination damage caused by the 1000-inch-pound impact on the skins of panels A and B are shown. The pulse echo indications of the same level of impact to the 1983 baseline panels also are shown. It is clearly demonstrated that in each case, the stitching of the enhanced panels reduces the area of delaminations resulting from the impact.

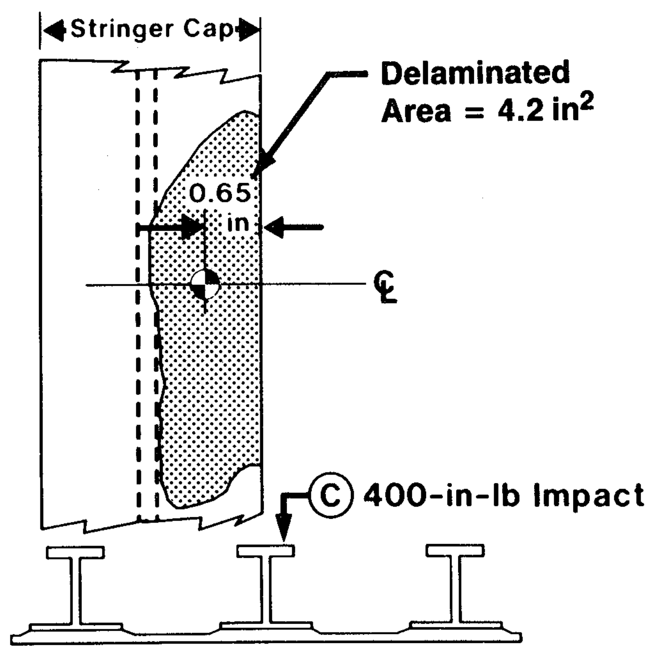




BOEING

Enhanced Panel
Pulse Echo Indication of Delamination Damage on Stiffener Cap

The result of the pulse echo inspection of the impacted stiffener cap of panel C is shown. The 400-inch-pound impact energy imparted to the stiffener cap caused a considerable area of delaminations with just a barely visible dent in the surface of the cap.

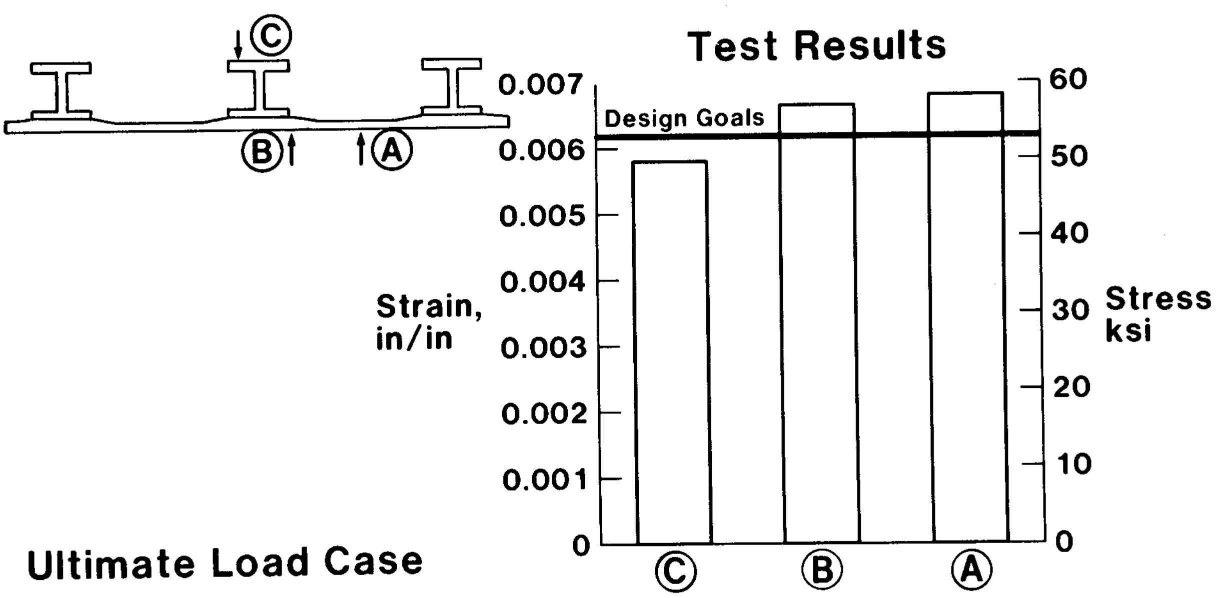


BOEING

ACEE NASA

Enhanced Element
Test Results

The results of the enhanced element test program are shown. The design goal of 30-kips/inch ultimate load required that the panels sustain an average P/AE strain of 0.00615 inch/inch at an average gross area stress of 52.0 ksi. Panels A and B, with the 1000-inch-pound impact damage to the skin, carried an average of 32.7-kips/inch load. This represents an average P/AE strain of 0.0067 inch/inch and an average gross area stress of 56.7 ksi. Panel C, with the 400-inch-pound impact damage to the stiffener cap, carried 28.5 kips/inch at an average strain of 0.00585 inch/inch and 49.5-ksi average stress.

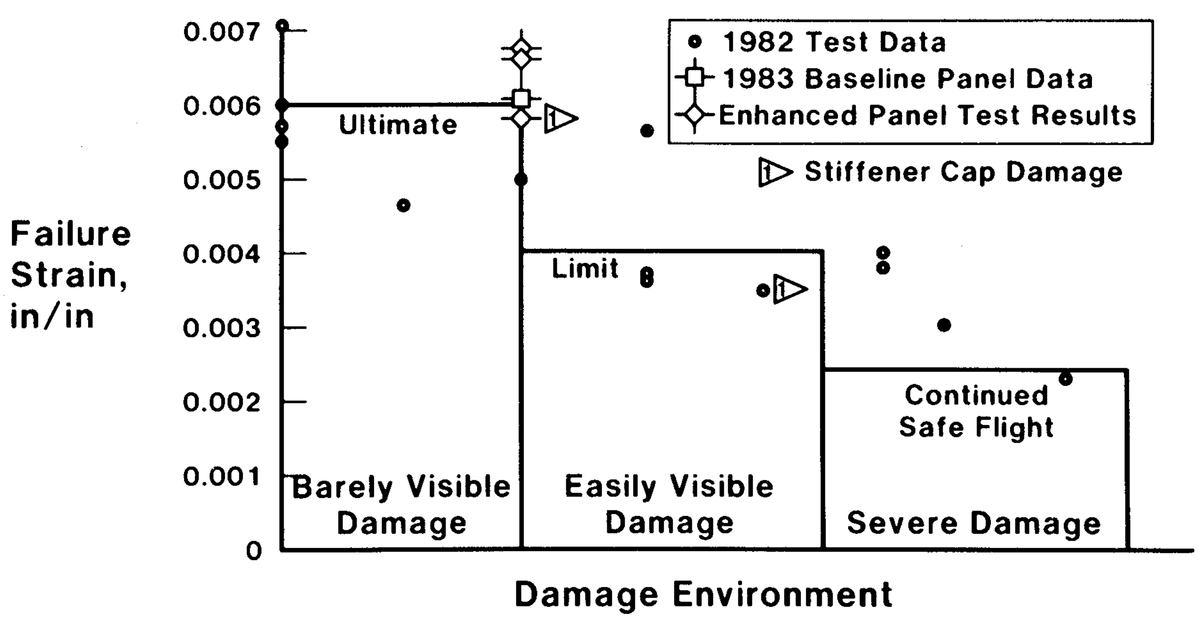


Ultimate Load Case
– Barely Visible Damage

BOEING

Enhanced Panel Tests
Demonstrated Capability

The demonstrated capability of the enhanced panel design is presented and compared with the previous test panel data generated during the LCPAS program. The basic 0.006-inch/inch P/AE strain goal for the barely visible damage load case was easily demonstrated for the skin damaged panels. This compared favorably with the previous test data. The stiffener cap damaged enhanced panel did not quite make the design goal but compared well with the 1982 test.



Enhanced Element Tests
Conclusions

The enhanced element test program proved the damage tolerance of the design for in-service or manufacturing inflicted skin damage. The enhanced design demonstrated a 12-percent increase in ultimate strength over the best of the previous designs for this skin damage condition. The manufacturing inflicted stiffener cap damaged enhanced panel came within 2.5 percent of the basic 0.006 inch/inch design goal. The test program indicated that stiffener cap damage is critical and the grid stitching of the skins contained damage due to impact.

- Demonstrated 12% Increase in Ultimate Strength Over 1983 Baseline
- Within 2.5% of 0.006-in/in Design Goal
- Stiffener Cap Damage Critical
- Skin Stitching Contains Damage

BOEING

ACEE NASA Large Panel Damage Tolerance Validation

The large panel damage tolerance program was designed to validate the final panel design. Five-stiffener panels were tested in static compression. The panels contained damage tolerance features derived from the results of the previous test data generated during the LCPAS program. Both skin and stiffener damage were evaluated, and the ultimate, limit, and safe flight load capabilities of the design were assessed.

- Validate Wing Panel Design Using Five-Stiffener Compression Panels
- Derive Damage Tolerance Features From Results of
  - LCPAS Phase I
  - Material Screening Tests
  - Element Tests
  - Wing Damage Assessment Study
- Evaluate Both Skin and Stiffener Damage
- Assess Ultimate, Limit, and Safe Flight Load Capabilities

BOEING

Large Panel Configuration and Design Features

The large compression panels were 37 inches wide with five stiffeners and 60 inches long to simulate two rib bays. Aluminum ribs were attached to the stiffener caps with C-clamps, and the ends of the panels had doublers and were potted in order to provide stable load introduction.

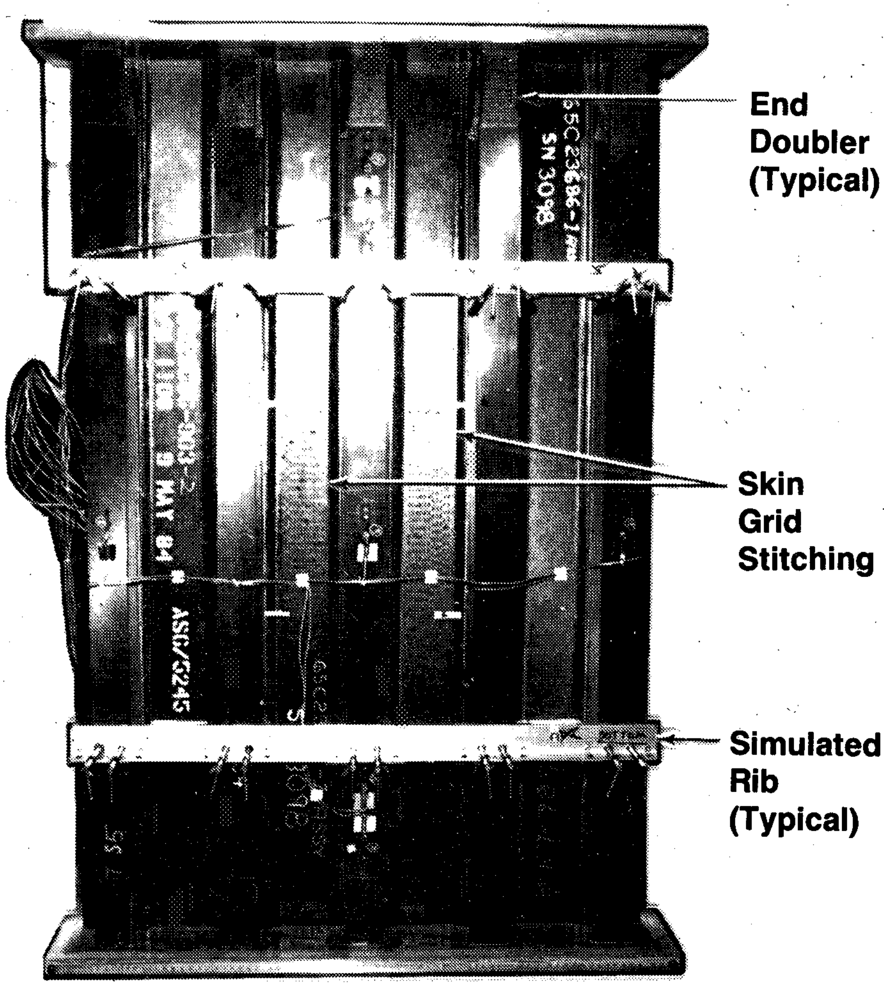
The panels retained the enhanced element damage tolerance features with the added feature that the rows of grid stitches in the skins were doubled. This added stitching had demonstrated increased damage containment during the damage growth test program. Three of the panels retained the skin/stiffener flange stitching of the enhanced design, and two panels were fabricated without this feature. One other change from the enhanced panel design was the selection of AS6/5245C material. This material had demonstrated an 11-percent increase in compression-after-impact strength over the AS6/2220-3 baseline material during the material screening tests.

- Five Stiffeners Wide and Two Rib Bays Long to Assess Euler Column
- Simulated Ribs
- Enhanced Panel Design Features Retained
  - Soft Skin
  - 0° Planks
  - Stitching
  - Interleaved Stiffener Caps
- Employ Toughened Material
  - AS6/5245C

BOEING

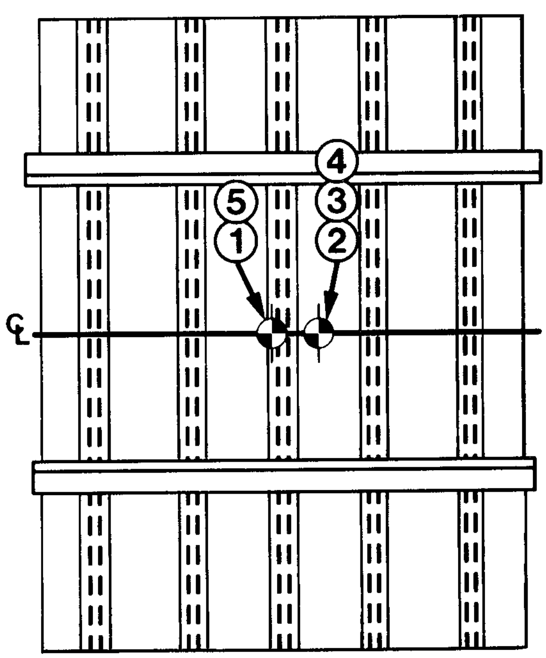
Large Five-Stiffener Test Panel

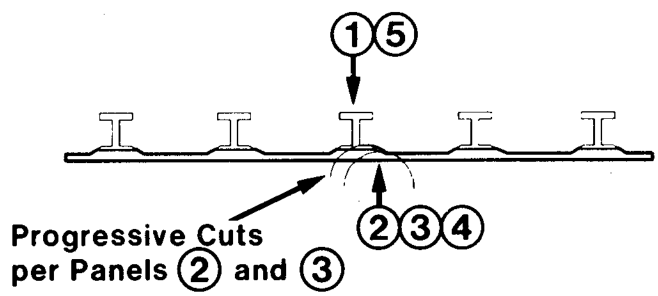
The five-stiffener panel test configuration is presented. The end-potting, doublers, and simulated ribs are shown together with installed test instrumentation. The instrumentation consisted of axial strain gages to record panel strains, EDI deflectometers to record out-of-plane deflections and acoustic emission transducers to monitor damage growth. The skin side of the panels was painted with moiré fringe material in order to provide a record of any skin deflections and buckles.



Large Panel Damage Tolerance
Validation Test Program

The large panel damage tolerance validation test program is shown. The five panels were to be tested in static compression to evaluate the ultimate, limit, and safe flight load capability of the chosen panel design. Panels 1, 4, and 5 were to demonstrate the ultimate load case with the barely visible impact damage. Panels 2 and 3 were to demonstrate limit load with the 2000-inch-pound easily visible damage and then be inflicted with progressive severe damage to evaluate the continued safe flight load capability.





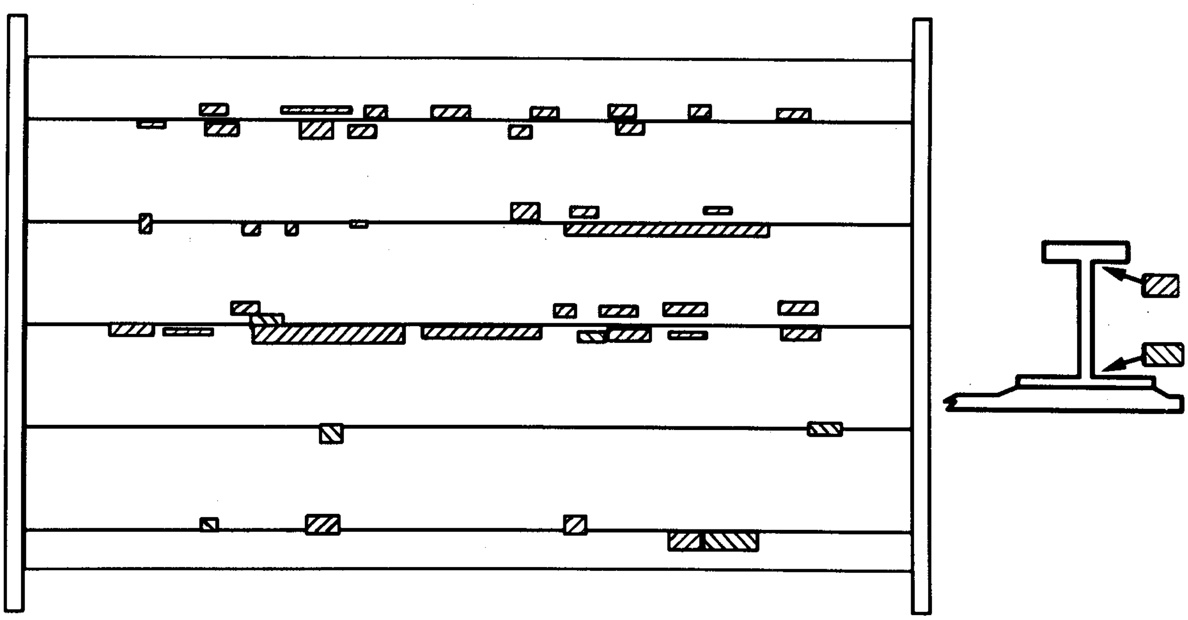
Panel No.	Damage	Load Case
1	400-in-lb Impact on Stiffener	Ultimate
2 3	2000-in-lb Impact + Progressive Cuts in Skin	Limit + Safe Flight
4	1000-in-lb Impact on Skin	Ultimate
5	200-in-lb Impact on Stiffener	Ultimate

BOEING

Large Five-Stiffener Panel Tests
Pulse Echo Indications of Anomalies in Panel No. 1

During the fabrication of the five 5-stiffener panels, a number of material processing problems arose. The AS6/5245C material was difficult to use because of lack of tack and boardiness. These material problems made stiffener ply lay-down on the tools extremely difficult. The plies were compacted after each lay-down as standard practice, but the complete compaction of the stiffeners was in doubt.

The pulse echo inspection results of the completed no. 1 panel are shown. Panels 2, 3, 4, and 5 had similar pulse echo indications throughout their lengths, but the no. 1 panel had the most.



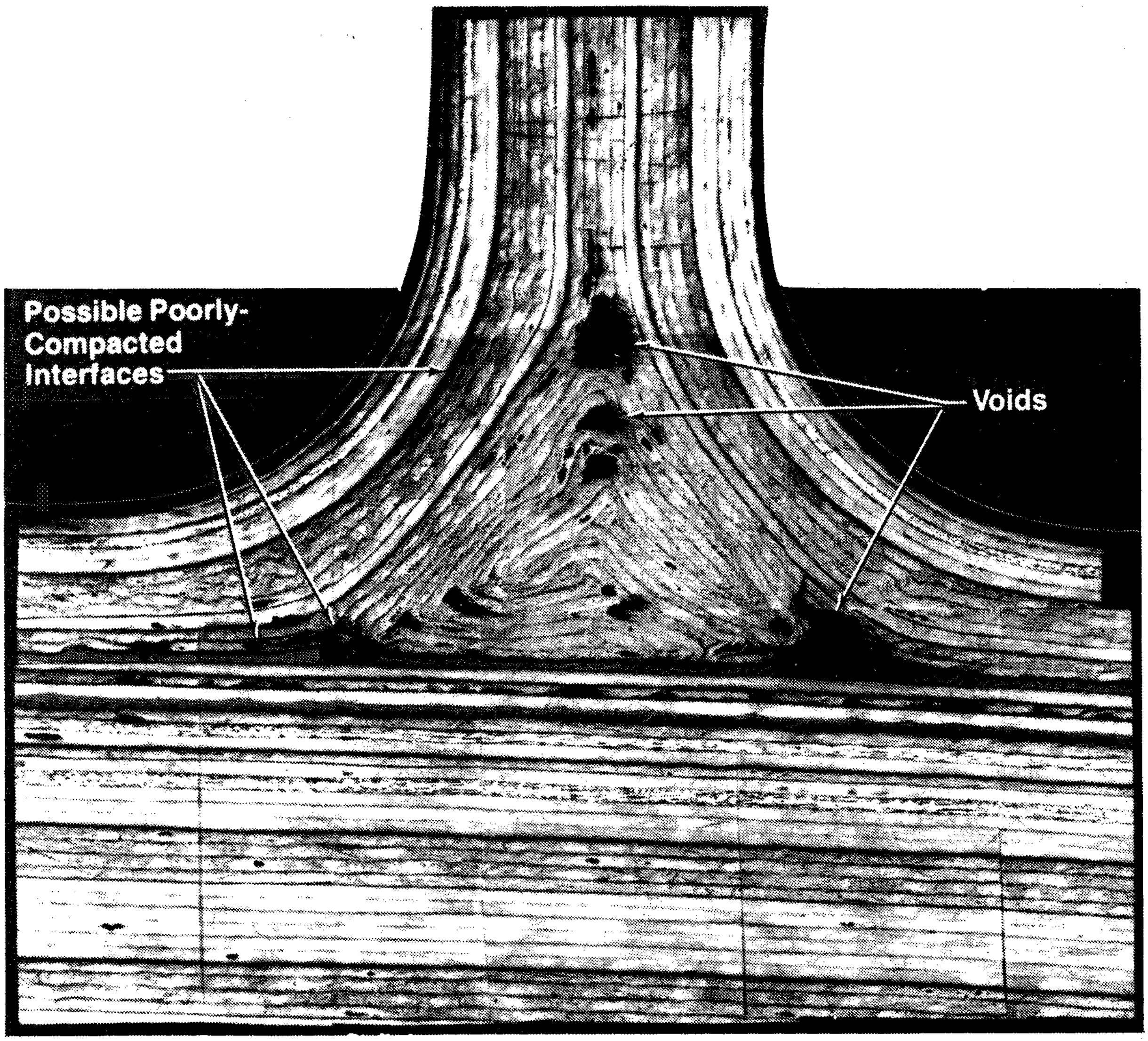
BOEING

ACEE NASA

Large Five-Stiffener Panel Tests

Photomicrograph of Sectioned Stiffener and Skin Area

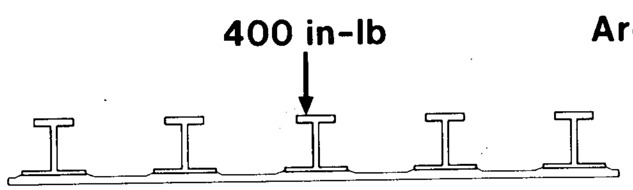
The end trim from the no. 1 panel was sectioned through a number of the pulse echo indicated areas, and photomicrographs were taken. A photomicrograph of one of these sections is shown. A number of voids can be seen clearly in the stiffener radius and stiffener/skin interface. A number of suspect stiffener ply compaction areas also can be noted.

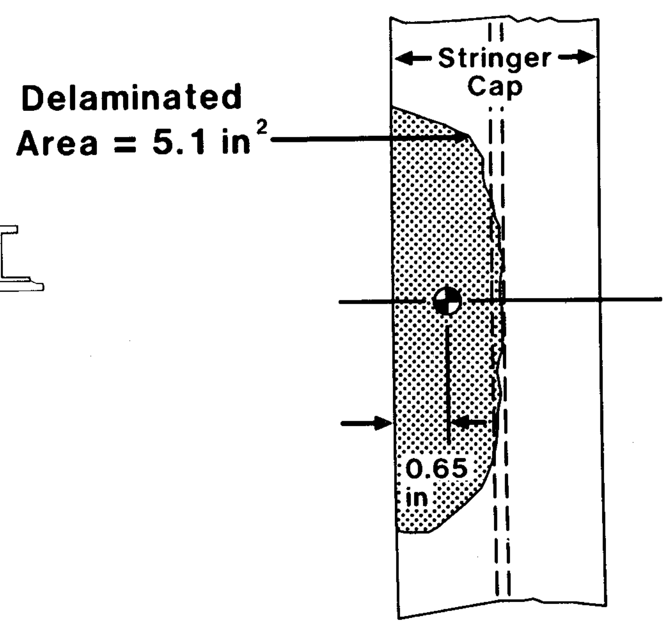


BOEING

Large Panel Tests - Panel No. 1
Impact Damage on Stiffener

The results of the static compression load test on the no. 1 panel is shown. The 400-inch-pound impact energy on the stiffener cap produced a 5.1-square inch delaminated cap area, and the panel failed at 19.5 kips/inch, which is 97.5 percent of limit load. This end load represents an average P/AE strain of 0.004 inch/inch and a gross area stress of 33.8 ksi. The panel was initially warped with the panel bowing concave to the skin, but upon failure of the central impacted stiffener at 17-kips/inch end load, the panel immediately deflected in the opposite direction, and the three center stiffeners separated from the skin over a wide area. The panel continued to carry load until overall panel failure occurred.





Test Results

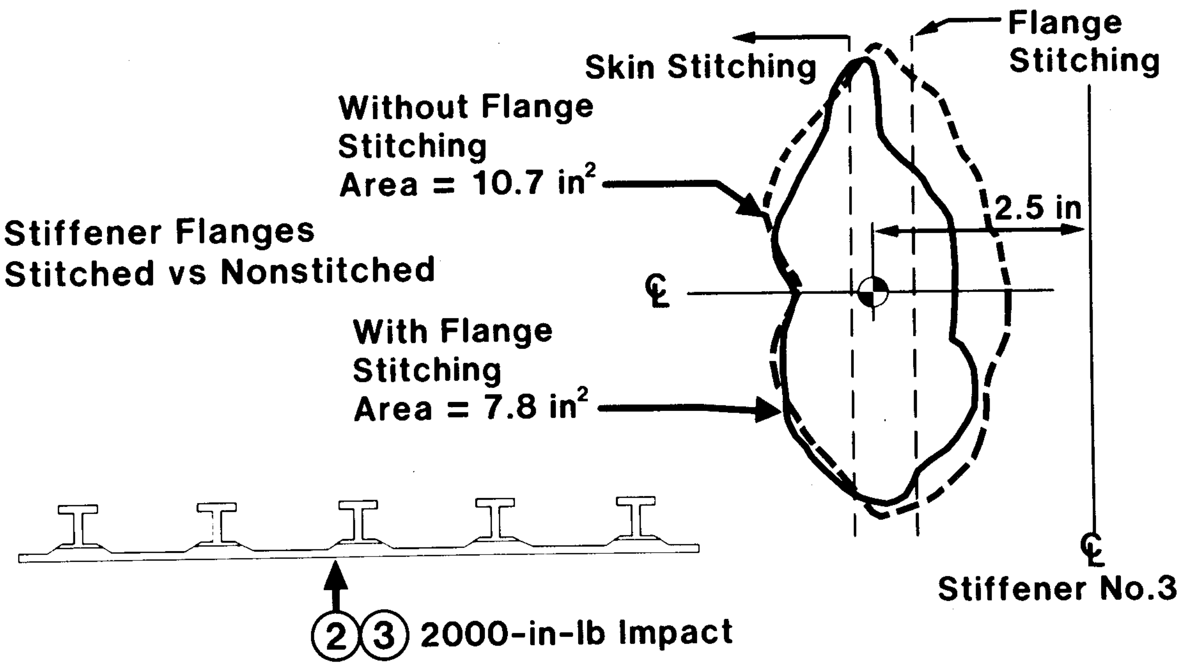
33.8 ksi – Stress
0.004 in/in – Strain

BOEING

Pulse Echo Indication of Skin Delaminations

Easily Visible Damage

The results of the pulse echo inspections of the impact sites of panels 2 and 3 are presented. The no. 2 panel, which featured the skin/stiffener flange stitching, seemed to have less delaminated area around the impact than did the non-flange-stitched no. 3 panel.

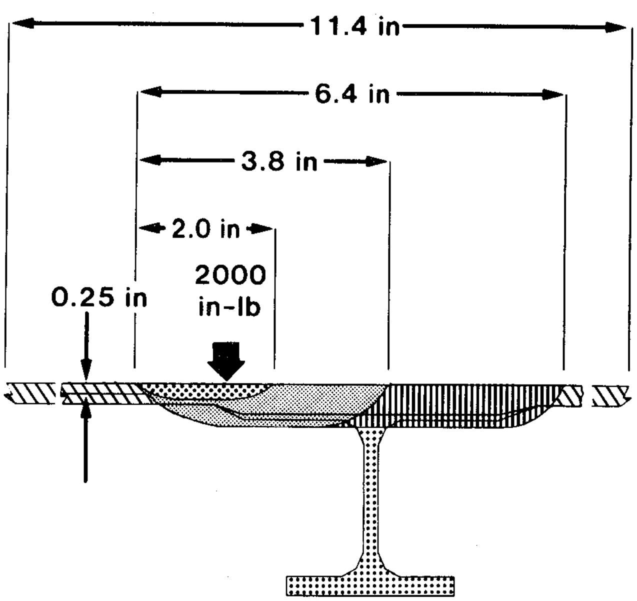


BOEING

C-2

Large Panel Test Results
Limit and Safe Flight Load Cases

The results are shown of the tests of no. 2 and 3 panels evaluating limit and safe flight load capability. In each case, the limit load capability of 20 kips/inch was demonstrated without failure. The panels were then progressively cut through the skin into the stiffener and tested to the continued safe flight load of 12.0 kips/inch each time the cut was enlarged. The final damage of the stiffener and skin totally cut through for a length of 11.7 inches yielded an end load of 11.77 kips/inch before panel failure. This massive damage load was 97.5 percent of safe flight load and represents a P/AE strain of 0.0024 inch/inch and a gross area stress of 20.3 ksi.



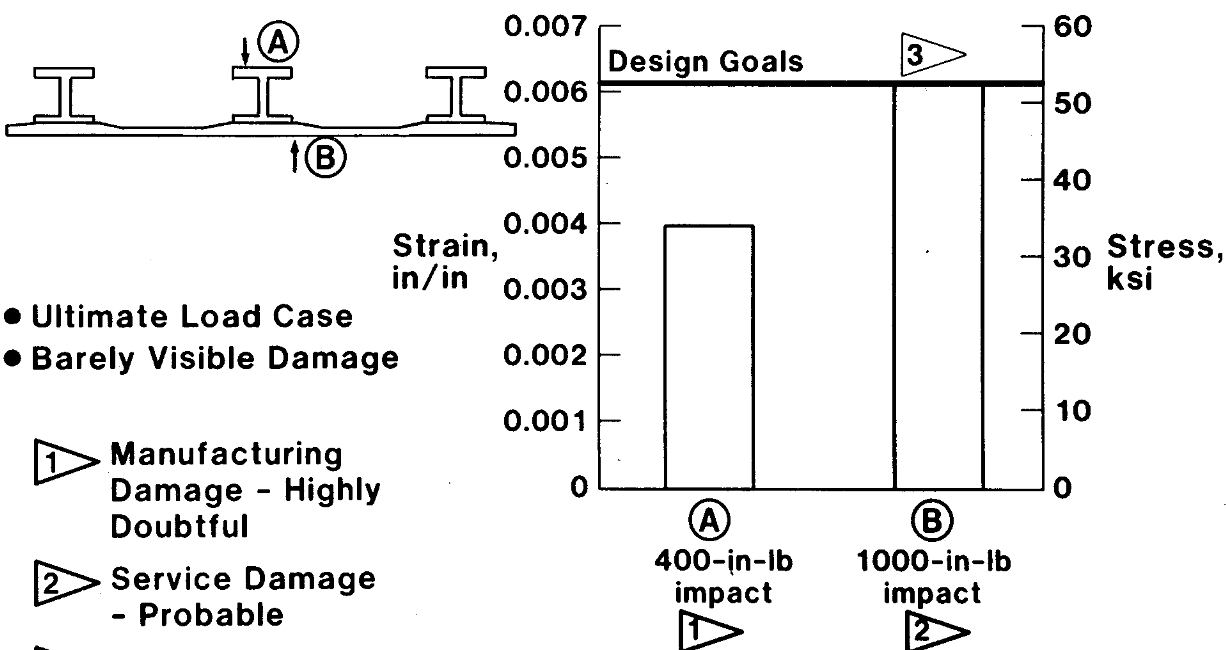
Load Case	Skin/Stiffener Interface: Stitched	Skin/Stiffener Interface: Non-Stitched	Load kips/in	Strain in/in	Comments
1 Limit Load	✓		20.0	0.0041	1
		✓			
2 Safe Flight	✓		12.0	0.0025	1
		✓			
3 Safe Flight	✓		12.0	0.0025	1
		✓			
4 Safe Flight Plus	✓		14.6	0.0030	
		✓	12.0	0.0025	1
5 Safe Flight		✓	12.0	0.0025	1
6 Safe Flight		✓	11.77	0.0024	

1 No Failure

BOEING

Large Panel Tests

The results of the ultimate load tests with barely visible damage are presented. The design goal of 30 kips/inch ultimate load was demonstrated on the panel with the in-service 1000-inch-pound skin damage without panel failure. The panel will be impacted with a second skin damage of 1000 inch-pound in an adjacent skin bay and tested to failure at a later date. The panel with the 400-inch-pound manufacturing damage on the stiffener cap was able to sustain limit load only.



- Ultimate Load Case
- Barely Visible Damage

1 Manufacturing Damage - Highly Doubtful

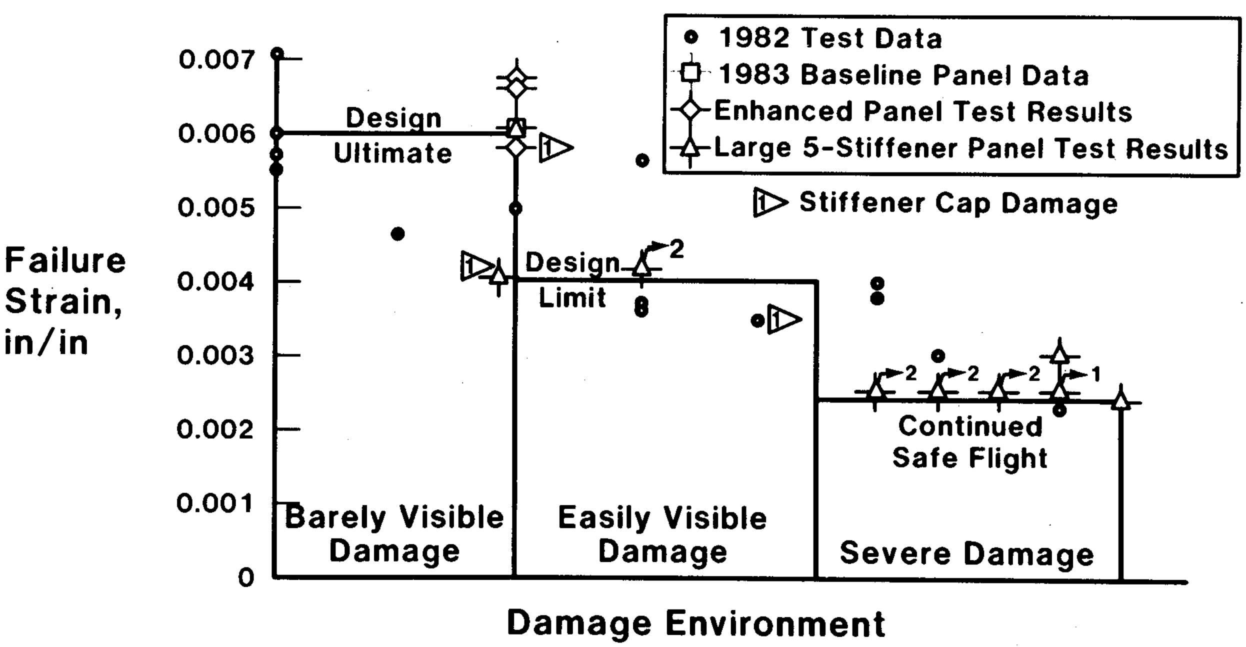
2 Service Damage - Probable

3 No Failure

BOEING

Large Panel Verification Tests - Demonstrated Capability

The results of the large panel damage tolerance validation tests are presented. The large five-stiffener panel test results compare favorably in all cases with the previous program test data. The final panel design has been verified as damage tolerant and has demonstrated ultimate, limit, and continued safe load capability with service damages.



BOEING

ACEE NASA

Large Panel Damage Tolerance Validation Tests
Conclusions

The large panel damage tolerance validation test program has successfully demonstrated design load capability with service damage. In all of the design load cases (ultimate, limit, and safe flight) the large panel results compared favorably with previous program test data.

The program has highlighted the need to address barely visible stiffener cap damage. This kind of damage is only possible in the manufacturing environment, and quality control procedures will need to be evaluated to prevent this damage.

- Design Load Capability Demonstrated With Service Damage
- Stiffener Cap Damage Must Be Addressed
- Through-Thickness Skin Enhancement Contains Damage

BOEING

Design Development of Heavily Loaded Wing Panels
Summary

The LCPAS program has shown systematic progress toward the overall program design goals. These goals were to improve composite wing panel design capability and demonstrate a 30-percent weight reduction from an aluminum wing design. This has now been achieved even on the critical compression-loaded upper surface panels. The ultimate design goal of 0.006 inch/inch for both tension and compression has been reached, and the damage tolerance of the final selected design has been verified. During the LCPAS program, the wing structure damage environment was investigated, and focus was placed on critical damage threats, service inspection, and quality control requirements.

- Improved Compression Panel Design Capability
- Assessed Wing Structure Damage Environment
- Verified the Damage Tolerance of the Selected Design

BOEING

ACEE NASA

Pressure Containment and Damage Tolerance in Fuselage Structure

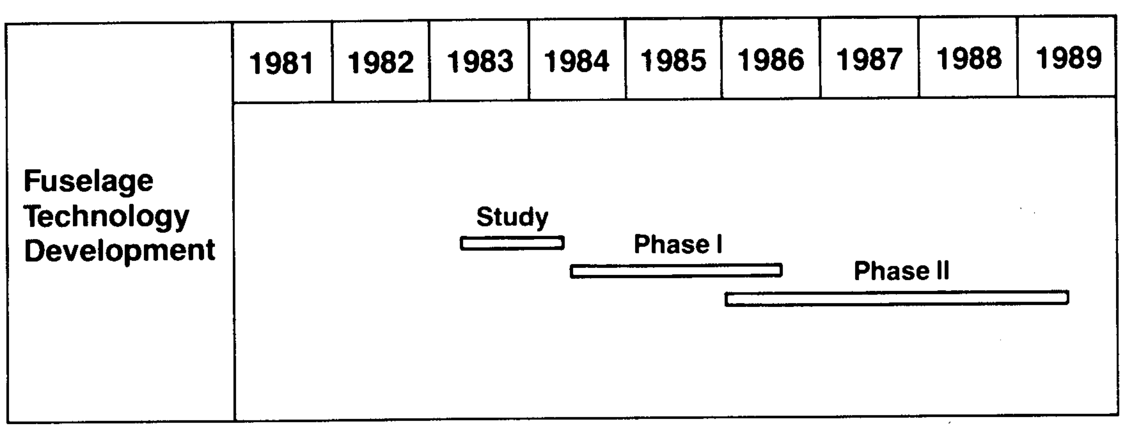
ACEE Composite Structures Technology Conference
Seattle, Washington
August 16, 1984

Presented by
R. W. Johnson
Boeing Commercial Airplane Company

BOEING

NASA Composite Fuselage Programs at Boeing

The NASA composite fuselage programs at Boeing are shown. The study program (NASA contract NAS1-17417) started in May 1983 and was completed in June 1984. The Phase I Critical Technology Program (NASA contract NAS1-17740) started in May 1984 and is scheduled to be completed in July 1986. The Phase II Technology Demonstration Program (NASA contract NAS1-17740) is scheduled to start in December 1985 and be completed in May 1989. These programs are sponsored by the National Aeronautics and Space Administration, Langley Research Center (NASA-LRC). Herman L. Bohon is the NASA-LRC ACEE COMPOSITES project manager and Jon S. Pyle is the NASA-LRC technical manager.



BOEING

Figure 2

Advanced Composites Fuselage Development

The major tasks of the fuselage development programs are shown. In the study program, the major technology issues that need to be addressed for composite fuselage structure were identified. Six design concepts were developed and evaluated. Several program options to develop the required technology were identified and a preferred option was recommended. The schedule and resource requirements for the recommended option were identified. In the Phase I program, efforts were concentrated on the critical technology issue of damage containment. Two design concepts, stringer stiffened laminate and honeycomb, were selected for further study. Developmental test parts will be designed, fabricated, and tested. Demonstration panels will be designed based upon the developmental test program results. The demonstration panels will be fabricated and tested and the results will be evaluated. Based upon these results, one design will be selected. In Phase II, additional test programs including frame bending, combined shear and compression loading on curved body panels, and combined loading on body panel splices will be conducted. Large panels containing window belt and keel beam details and updated damage tolerance details will be tested and evaluated. The results of the critical technology programs at the Douglas Aircraft Company and Lockheed Company will be incorporated into the Phase II program.

Study Program (12 Months)

- Technology Issues
- Design Concept Evaluation
- Program Options
- Program Schedule and Resource Requirements

Phase I (26 Months)

Damage Tolerance/Pressure Containment

- Two Designs (Laminate, Sandwich)
- Development Tests
- Demonstration Tests
- Design Selection

Phase II (41 Months)

Technology Readiness Demonstration

- One Design
- Large Panels, Frames
- Crown, Window Belt, Keel

Figure 3

Advanced Composites Fuselage Development

The program design goals are as shown. It is anticipated that a 10% cost reduction will be achieved due to a 20% reduction in part count and a reduction in assembly costs. Assembly cost reductions will be achieved by fabricating larger panel segments and reducing the number of mechanical splice joints. A 30% shell weight reduction will be achieved by optimizing all structural details for strength, stiffness, and damage tolerance requirements.

Design Goals

- 10% Cost Reduction
- 30% Weight Reduction Monocoque Shell
- 20% Part Count Reduction

BOEING

Figure 4

Fuselage Component Test Plan

Development tests will evaluate tension fracture strength, tension and compression damage containment, compression crippling strength, postbuckled panel compression strength, postbuckled panel shear strength, pressure pillowing effects, frame bending, fastened joints, window cutouts, and combined load effects. The fuselage component test plan is summarized as shown in Figure 5.

The test program planned for Phase I is defined as follows:

- Flat fracture panels, laminate and honeycomb
- Curved pressure loaded fracture panels
- Stringer crippling elements
- Skin-stringer compression buckling panels
- Honeycomb compression buckling panels
- Flat stringer stiffened shear panels
- Compression damage tolerant panels, laminate and honeycomb
- Stringer-frame intersection pressure loaded details
- Curved combined load damage tolerance panels, laminate and honeycomb

The test program planned for Phase II is defined as follows:

- Frame bending tests
- Curved panels under combined load of tension, shear, and compression
- Combined load tests on panel splices
- Combined load tests on window panel details
- Compression test of keel beam concentrated load redistribution area
- Verification tests of critical damage tolerance designed panels

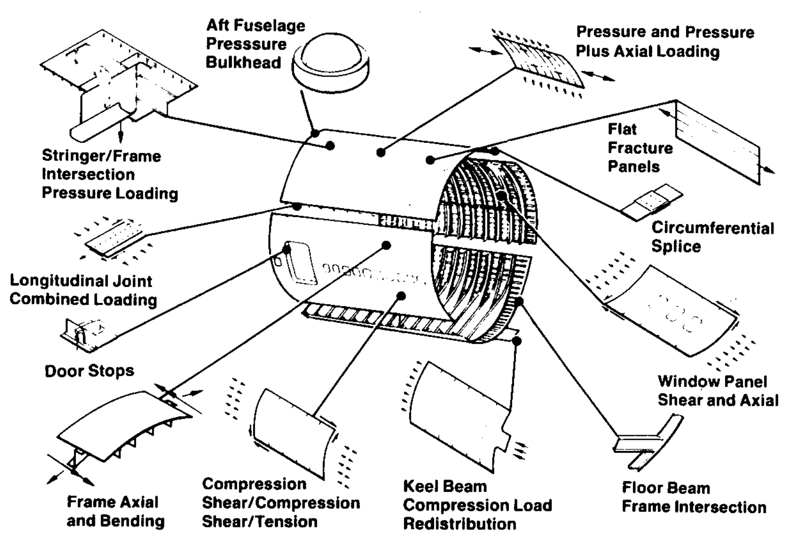


Figure 5

BOEING

Typical Commercial Transport

The distribution of the total aircraft structural weight between the fuselage, wing, empennage, landing gear, and wing control surfaces is shown. Since the fuselage and wing contain about the same percentage of the total aircraft weight, the potential for weight reduction for both wing and fuselage structure is the same. Thus the development of technology directed towards the application of composite materials to wing and fuselage structure should receive the same emphasis.

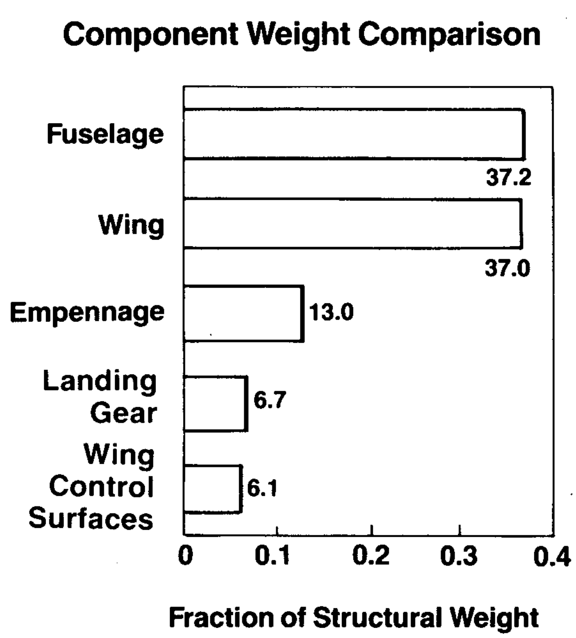


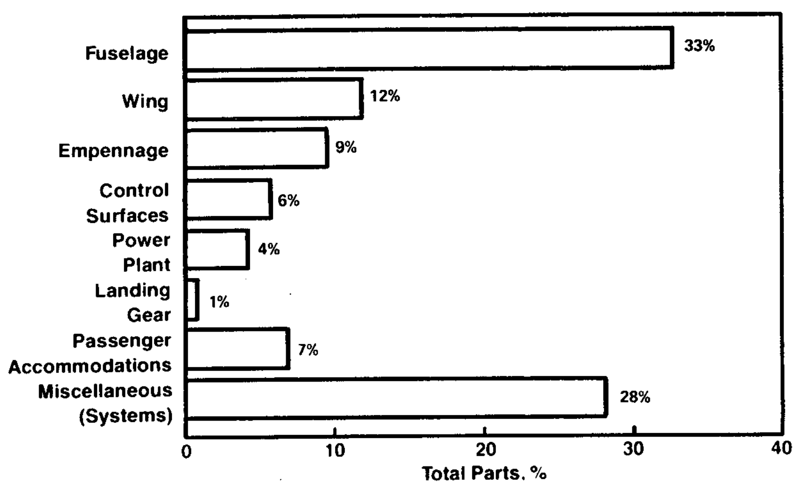
Figure 6

BOEING

Typical Commercial Transport

The distribution of total parts in a typical commercial transport is shown. The potential for cost reduction in the fuselage, by reducing parts, is greater than for the wing due to the significantly higher part count as shown.

Total Airplane Distribution of Parts



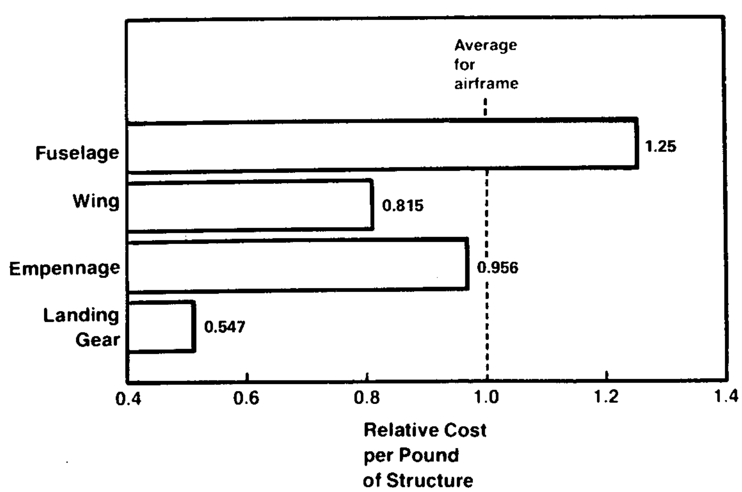
BOEING

Figure 7

ACEE NASA Typical Commercial Transport

The relative cost per pound of structure for the major components of a typical commercial transport is shown. The fuselage structure, at 1.25 times the average for the airframe, due to the higher part count, presents the greatest potential for cost reduction.

Component Cost Comparison



BOEING

Figure 8

Baseline Study Section

The study section selected as the baseline for design development and for aluminum to composite cost-weight comparisons is the 757 aft fuselage. This section is representative of state-of-the-art standard body, aluminum fuselage design. The existing set of internal loads for the 757 were used for sizing the composite components. In order to maintain consistency with the current 757, the composite concepts retained the same internal and external configuration as the 757 airplane including frame spacing and inner (IML) and outer mold lines (OML). Floor beams, doors, door cutout reinforcement, keel beams, and bulkheads were not included in the composite designs. These components were included when the study section results were later extrapolated to a complete fuselage.

Commercial Transport

- Loads
- Configuration
- Weight Comparison

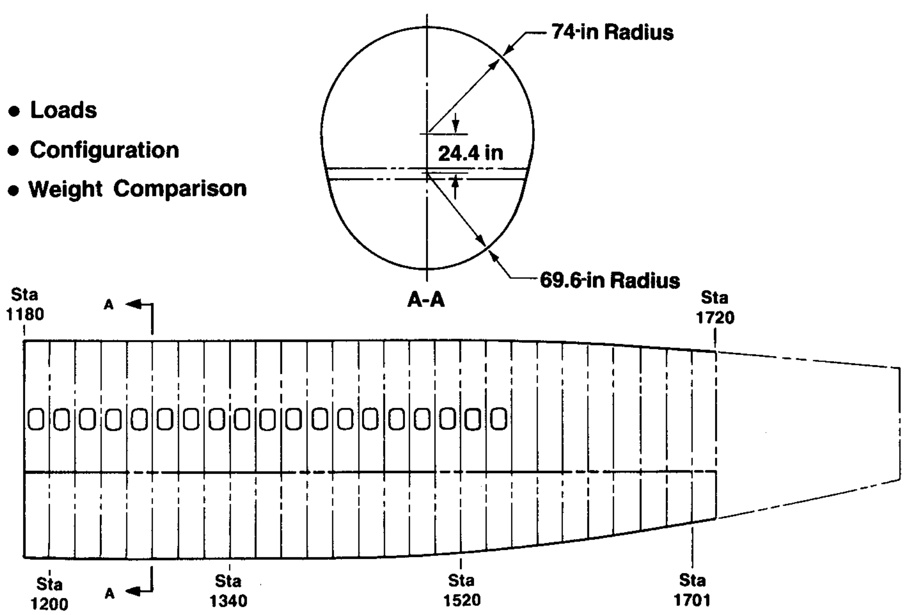


Figure 9

BOEING

Design Loads

The envelope of maximum design loads along the fuselage study section is shown. In the crown, the maximum design tensile loads result from bending and internal pressure. The compression design loads in the crown come from bending with no internal pressure. The crown compression loads are considered since this type of loading is critical for general shell stability. In the keel, the maximum compression design loads result from bending with no internal pressure. The side panel design shear loads come from body bending.

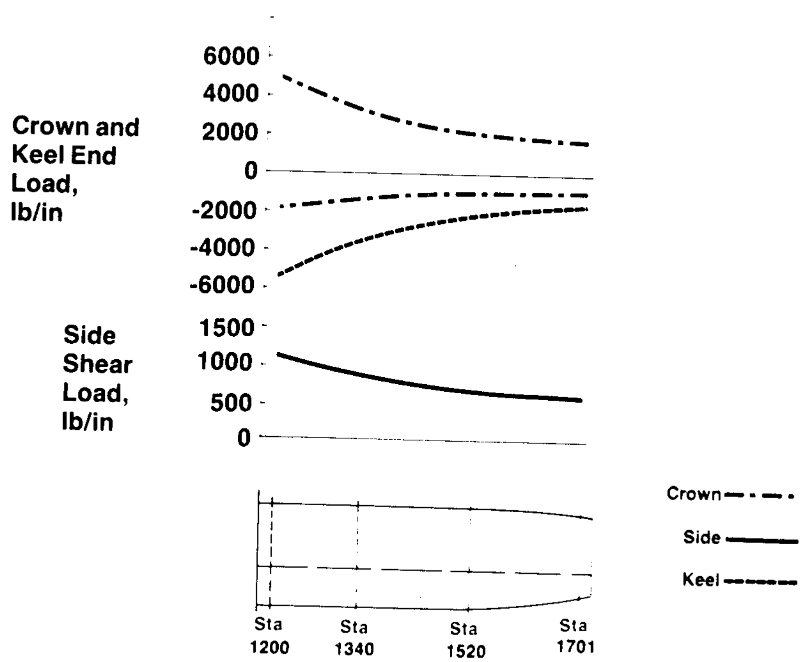


Figure 10

Criteria and Design Constraints

The criteria and design constraints used to develop the composite designs are shown. The design loads and damage tolerance criteria that Boeing typically uses for the design of commercial transport aircraft in conjunction with the F.A.R. requirements are applied to the composite design. The ultimate design strains of 0.006 in/in tension, 0.005 in/in compression, and 0.010 in/in shear are considered as program goals. These design strain values have been validated by the NASA-funded LCPAS studies conducted by Boeing for heavily loaded wing panels. The skin panels between stiffeners in the stiffened laminate panel designs will be allowed to buckle at 30% design ultimate load (DUL). This buckling level has been selected to provide buckle-resistant fuselage panels at the 1g cruise condition. This minimizes fatigue cycling of the buckled structure and provides minimum aerodynamic drag. The design constraint of balanced and symmetric plies has been imposed to minimize warping and residual stresses in the laminates.

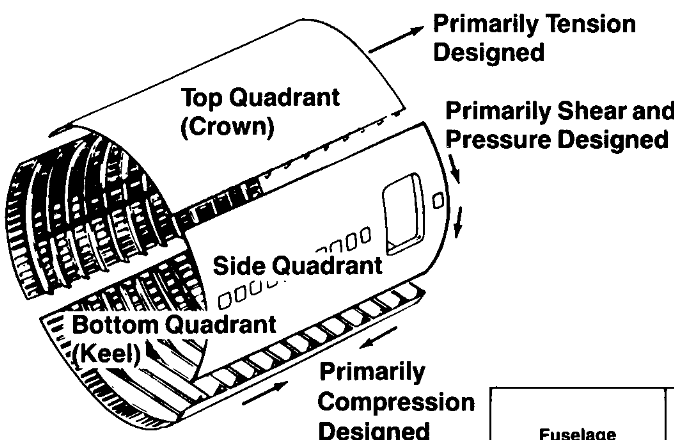
- **Design Loads and Damage Tolerance per Boeing and F.A.R. Requirements**
- **Design Ultimate Strains**
  - **Tension 0.006 in/in**
  - **Compression 0.005 in/in**
  - **Shear 0.010 in/in**
- **Stiffened Laminate Postbuckled Skin Design**
- **Balanced and Symmetric Ply Stacking**

BOEING

Figure 11

Typical Fuselage Construction Major Design Parameters

The major design parameters for the top, side, and bottom quadrant areas of a typical fuselage section are shown. The relative panel load magnitudes show that the top quadrant is designed by tension, the side quadrant is designed by shear, and the bottom quadrant is designed by compression. As previously mentioned (Figure 10), the design of the top quadrant must also consider compression loads since this type of loading is critical for general shell stability.



Fuselage	Quadrant	Relative Panel Load Magnitude		
		Tension	Compression	Shear
	Top	High	Low	Nominal
	Side	Nominal	Nominal	High
	Bottom	Low	High	Nominal

BOEING

Figure 12

Typical Weight Distribution
Commercial Transport Fuselage

The weight breakdown of the major structural categories of a typical commercial transport fuselage is shown. As can be seen, the shell structure, which includes the skin, stiffeners, and frames, contains by far the greatest percentage of the fuselage weight. Therefore, detail design efforts were concentrated on the shell structure to predict an accurate weight reduction for the most significant contributor to the total fuselage weight.

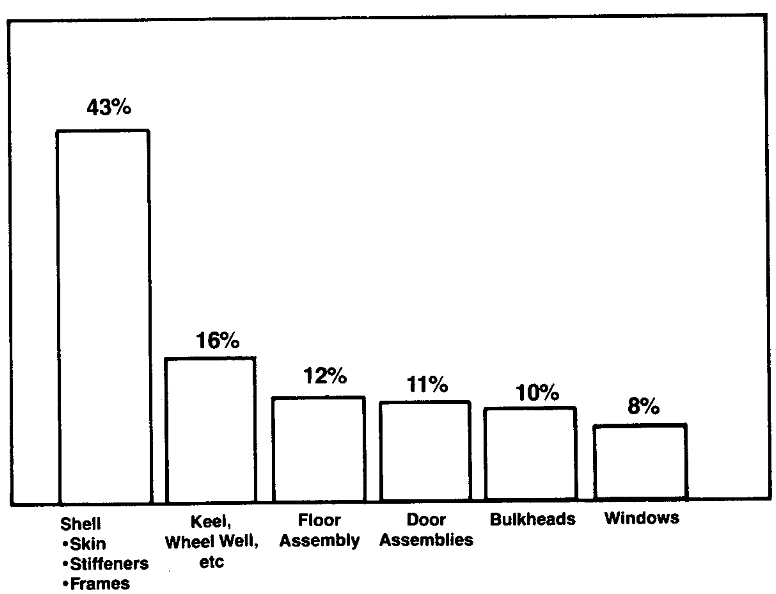


Figure 13

Shell Design Concepts

Four basic fuselage design configurations, as noted, were studied. The first design concept, full-depth honeycomb skin panels without frames, was not carried into the detailed design phase. The development of this concept would require a complete evaluation of the floor and interior support structure and was considered to be outside the scope of the present program. Detail designs for the other three configurations were developed and details are presented in the following figures. From the three basic configurations, six designs were developed.

- Concept 1 — Honeycomb sandwich with frames
- Concept 2 — I-section stringer stiffened laminate
- Concept 3 — Foam filled hat section stringer stiffened laminate with cobonded frame shear ties
- Concept 4 — Foam filled hat section stringer stiffened laminate with mechanically attached frame shear ties
- Concept 5 — I-section stringer stiffened honeycomb skin
- Concept 6 — Foam filled hat section stringer stiffened honeycomb skin

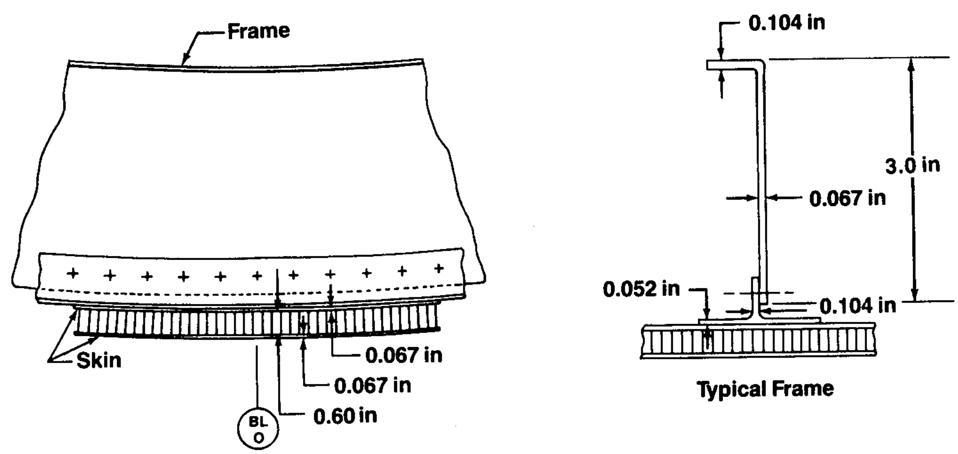
- **Full-Depth Honeycomb — Without Frames**
- **Full-Depth Honeycomb — With Frames**
- **Stringer-Stiffened Laminate Skins**
- **Stringer-Stiffened Honeycomb Skins**

BOEING

Figure 14

Honeycomb Sandwich Skin
Concept 1

The full-depth honeycomb sandwich design with frames, Concept 1, is shown. This design contains J-frame sections at 20-inch spacing. Dimensions for a typical section are shown. The inner and outer skin laminates are laid up on the core with an adhesive layer and the assembly is then cocured. The frame T is cobonded to the panel and the J-frame is mechanically attached.
The core shear modulus and thickness and the face sheet thickness for all the panels are selected based on a design requirement for no compression, shear, or combined compression and shear buckling below ultimate load.



BOEING

Figure 15

ACEE NASA

Honeycomb Design
(Concept 1)

Detailed dimensions for the crown and keel panels for Concept 1 are shown. The body station references are defined in Figure 9. The face gages were selected based on the design loads (Figure 10) and the design strains and requirements for balanced and symmetric ply stacks as shown in Figure 11. The laminate definition is the standard nomenclature; number of 0-deg plies, number of 45-deg plies and number of 90-deg plies. The thickness per ply was 0.0074 inch. The number of face sheet plies is based on load and strain requirements and minimum gage considerations for damage were not considered. The core thicknesses are based on the buckling requirements as stated in Figure 15. The honeycomb core used in the design is fiberglass and weighs 4.0 pounds per cubic foot.

Station	Crown			Keel		
	Core Depth, in	Face Sheet t, in	Laminate	Core Depth, in	Face Sheet t, in	Laminate
1200	0.20	0.052	3/2/2	0.60	0.067	3/4/2
1340	0.20	0.037	2/2/1	0.50	0.052	3/2/2
1520	0.15	0.037	2/2/1	0.35	0.037	2/2/1
1701	0.15	0.037	2/2/1	0.30	0.037	2/2/1

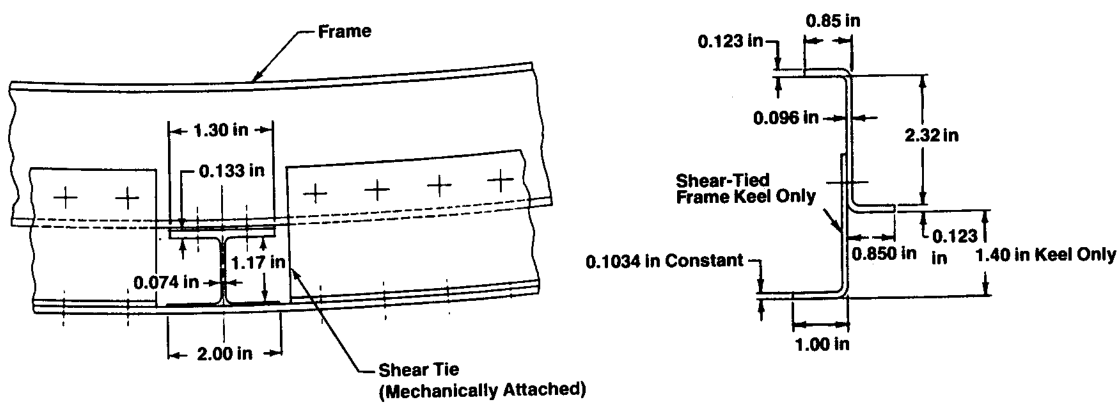
BOEING

Figure 16

Laminate Skin With Discrete Stringers

Typical I-Section Stringer and Frame — Concept 2

Typical details for the I-section stringer stiffened laminate design, Concept 2, are shown. The stringers, located at 10-inch spacing in the crown and 8-inch spacing in the keel, were cocured to the skin laminate. The body frame is a Z-section that is mechanically attached to the skin panel. The frames are located at 20-inch spacing. In the upper crown area, the frame is only attached to the stringer with fasteners through the I-section cap. In the lower crown area and keel area, the body frame is shear tied to the skin panel with mechanically attached angle sections. The I-section stringer is designed as a stable column element between the frames for the critical compression designed areas. In the shear critical areas, the stringers are designed with sufficient stiffness to restrict local buckles to a single bay. The thickness of the stiffener flange at the skin interface is selected to prevent out-of- plane peeling forces from delaminating the stiffeners.



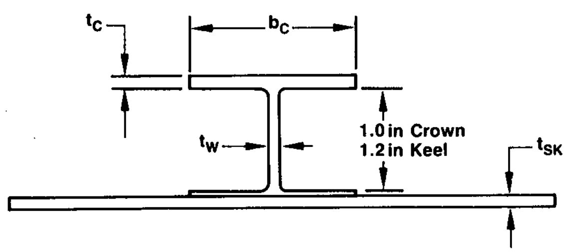
Keel Region

BOEING

Figure 17

I-Section Stringer — Laminate Skin Design
(Concept 2)

Detailed dimensions for the crown and keel panels for Concept 2 are shown. The body station references are defined in Figure 9. The face gages and stringer sections were selected based on the design loads (Figure 10) and the design strains and requirements shown in Figure 11. The laminate definition is the standard nomenclature as defined in Figure 16. The thickness per ply was also 0.0074 inch. The stringer section was sized as a stable Euler column over the 20-inch frame spacing for the compression critical design areas. Since the skin panels are allowed to buckle, only the effective skin width, as determined from classic panel analysis, contributes to the Euler column section. The skin thickness between stiffeners was selected based on the 30% DUL buckling criteria as defined in Figure 11. The skin panel was assumed to be simply supported on all four sides with a panel width defined between the edges of the stringer skin flange and a panel length defined by the frame spacing. The skin panel was analyzed using a Boeing developed buckling program, LEOTHA.



Body Location	Station	Skin Laminate	t_{SK} (in)	b_c (in)	t_C (in)	t_W (in)
Crown	1200	3/8/2	0.096	0.80	0.081	0.059
Crown	1340	2/8/2	0.089	0.80	0.081	0.059
Crown	1520	4/4/2	0.074	0.60	0.059	0.059
Crown	1701	4/4/2	0.074	0.50	0.059	0.059
Keel	1200	6/8/2	0.118	1.30	0.133	0.074
Keel	1340	3/8/2	0.096	1.20	0.118	0.074
Keel	1520	5/4/2	0.081	0.74	0.074	0.074
Keel	1701	2/4/2	0.059	0.50	0.059	0.074

Figure 18

BOEING

Laminate Skin With Discrete Stringers

Typical Hat Section Stringer and Frames — Concepts 3 and 4

Typical details for the foam filled hat section designs, Concepts 3 and 4, are shown. Concept 3 utilized cobonded frame shear attach tee sections and Concept 4 utilized a full-depth frame channel that was mechanically attached at the skin line. The frame channel would be cut for the stiffener to pass through and a reinforcing angle was mechanically attached to provide a continuous frame chord. Both frame attach concepts applied only in the lower crown and keel area. The crown region frame attach clip, as shown, was common for both concepts. The frames were located at 20-inch spacing.

The foam filled hat stiffeners, located at 10-inch spacing in the crown and 8-inch spacing in the keel, were cocured to the skin panel. Hat sections were considered as stiffener elements since they are a more structurally efficient element than the I-section. The hat section provides a greater width of stable skin and the closed section is more torsionally stable. The foam filled concept for the hat sections was selected for evaluation since it was initially considered to provide a manufacturing cost advantage. The hat stiffener elements were sized in the same manner as discussed for the I-section stiffeners, Figure 17 and 18. Detailed skin and stringer section dimensions are not presented for Concepts 3 and 4 since they were not recommended for further study because of difficulty in nondestructively inspecting the foam filled stiffener elements.

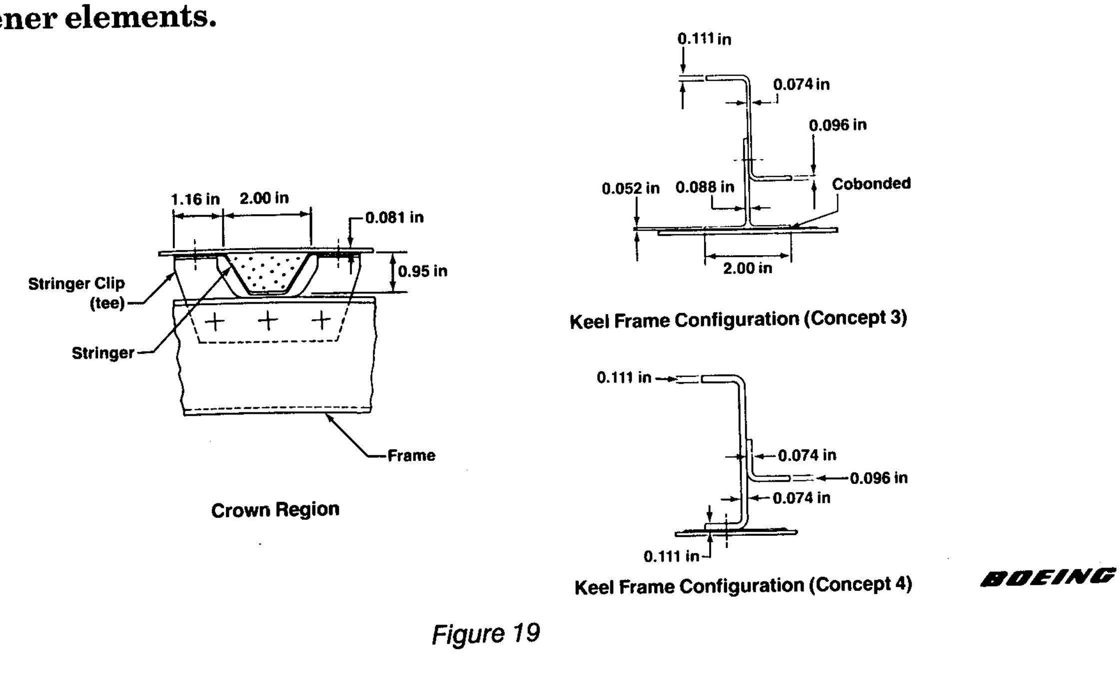
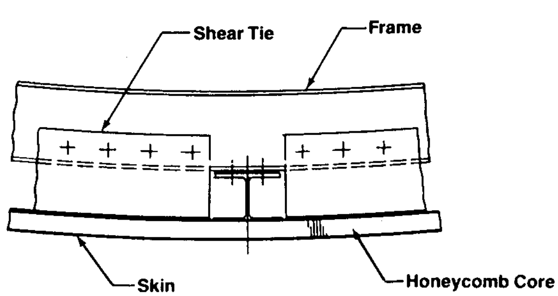


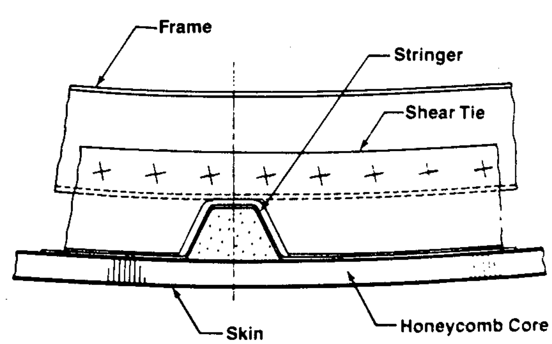
Figure 19

Honeycomb Sandwich Skin
With Discrete Stringers — Concepts 5 and 6

Typical details for the stringer stiffened honeycomb panel designs, Concepts 5 and 6, are presented. Concept 5, I-section stiffened honeycomb panel, utilized similar stringer and frame elements as shown for Concept 2 (Figure 17). For Concept 5, the frame shear ties would be cobonded to the honeycomb panel. Concept 6, foam filled hat section stiffened honeycomb panel, utilized similar stringer, frame, and shear tie elements as shown for Concept 3 (Figure 19). These designs were considered since the stable stringer and honeycomb panel elements would be more structurally efficient than the stiffened laminate skin concepts. Detailed skin and stringer section dimensions are not presented for Concepts 5 and 6, since they were not recommended for further study (see Figure 23).



I-Section Stringers (Concept 5)



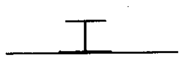
Hat Section Stringers (Concept 6)

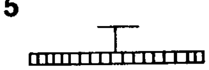
BOEING

Figure 20

Fuselage Design Concepts
Shell Weight Reduction

The six design concepts were developed for the study section (Figure 9) and a detailed weight reduction analysis was performed. The resulting weight reductions for the six design concepts are shown. This analysis pointed out that the more structurally efficient hat section produced the greatest weight reduction. The full-depth honeycomb design, Concept 1, did not produce a weight reduction better than Concept 2 due to the balanced, symmetric ply constraints. The higher structural efficiency of Concepts 5 and 6 as compared to Concepts 2 and 3 did not show a large weight benefit, also due to the balanced, symmetric ply constraints.

1	 (28%)	• Full-Depth Honeycomb Core
2	 (28%)	• Laminate Skin • Cocured I-Stringers
3	 (32%)	• Laminate Skin • Cocured Foam Filled Hat Section Stringers • Bonded Frames

4	 (29%)	• Laminate Skin • Cocured Foam Filled Hat Section Stringers • Frames Mechanically Attached
5	 (29%)	• Honeycomb Core • Cocured I-Stringers
6	 (30%)	• Honeycomb Core • Cocured Foam Filled Hat Section Stringers

20-inch Frame Spacing

BOEING

Figure 21

Cost Comparisons
Labor Requirements for Shell Fabrication

A detailed cost analysis, to fabricate a constant section of fuselage, was developed for each design concept. The results indicate that the foam filled hat section designs, Concepts 3 and 4, which were initially considered to be more cost effective than the I-section design, Concept 2, are not significantly less. The Concept 1 honeycomb design is shown to be the most cost effective design due to the structural simplicity and minimum number of parts. Concepts 5 and 6 are shown to be the most costly due to the requirement of having to provide for stringer and frame shear tie tooling on the honeycomb panels.

Description Concept	Basic Factory Labor Normalized Hours [1]
Concept 1 Honeycomb-Sandwich Skin No Stringers	1000
Concept 2 Laminate Skin I-Section Stringers	1050
Concepts 3,4 Laminate Skin Hat Section Stringers	1040
Concept 5 Honeycomb Sandwich Skin I-Section Stringers	1280
Concept 6 Honeycomb Sandwich Skin Hat Section Stringers	1400

[1] Labor Hours Based on Fabrication of Constant Section With Body Frames at 20-in Spacing

BOEING

Figure 22

Design Recommendation

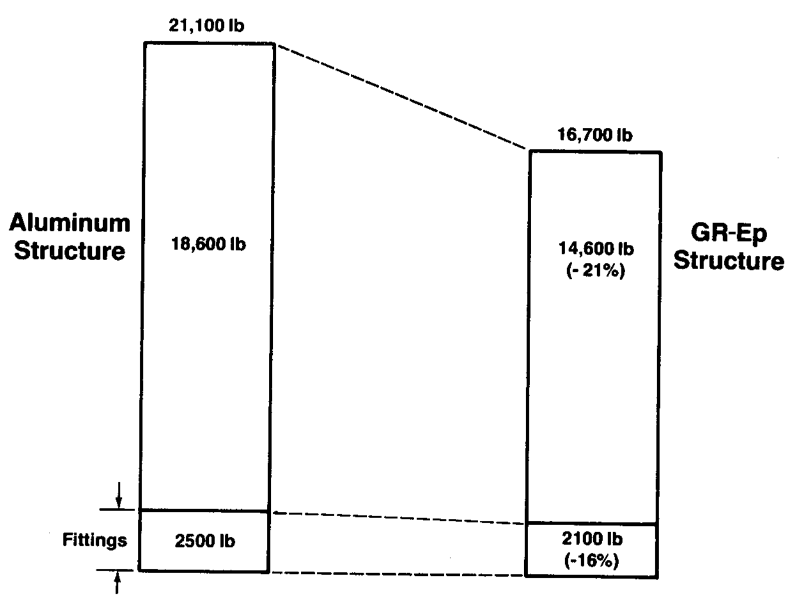
The designs recommended for further study are the full-depth honeycomb sandwich skin design (Concept 1) and the I-section stringer stiffened laminate design (Concept 2). The design recommendation was based on weight, cost and inspectability. The honeycomb design, Concept 1, was selected primarily based on cost, since the cost analysis indicated this design concept to be the most cost effective. The foam filled hat section designs, Concepts 3 and 4, were not seelcted even though the cost and weight analysis indicated these designs were better than the I-section stiffener designs. An in-depth analysis of inspection requirements during fabrication and in service indicated that the foam filled hat designs were not suitable for commercial transport fuselage structure. The primary concern was water absorption by the foam material and the associated difficulties of having to inspect the stiffeners to ensure that no water was present. In addition, the closed hat section would make repairs significantly more difficult as compared to the I-section stiffener.

- Concepts Recommended for Further Study:
  - Full Depth Honeycomb Sandwich Skin (Concept 1)
    - Frames, No Stringers
  - Laminate Skin With I-Section Stringers (Concept 2)
- Selection Basis
  - Weight
  - Cost
  - Inspectability

Figure 23

Commercial Fuselage Weight Reduction

The weight reduction analysis that was performed for the Concept 2 design was applied to a commercial transport fuselage. A detailed weight breakdown for a fuselage was produced and 18,600 lb of aluminum fuselage structure was identified as participating structure for conversion to composite structure. The remaining nonparticipating structure included window glass, floor seat tracks, and those structural components that were already fabricated from composite materials. In addition, 2500 lb of existing fittings were identified as having potential of being reduced by designing different load paths in the local area. The 18,600 lb of aluminum structure was broken down into skin, stringer, and frame elements. The weight reduction, obtained in the study section for each of these three basic structural elements, was then applied to the skin, stringer, and frame elements of the total fuselage. The resulting analysis produced a 21% reduction for the participating structure and a 16% weight reduction for the fittings for a total reduction of 4400 lb.



BOEING

Figure 24

Technology Issues

Technology issues were identified in the areas of structures and materials. The issues selected were those that would have the greatest influence on a fuselage design from a weight, cost, and size of the existing technology base viewpoint. The structures items were identified based primarily on the lack of a technology base for primary fuselage structure. The material design strain level was identified as a technology issue due to the lack of a data base for damage tolerance of composite fuselage structures. The flammability and fire protection issue was selected due to the lack of a data base in this area.

Structures

- Damage Containment
- Postbuckling
- Impact Dynamics
- Bolted Joints
- Cutouts
- Repair

Materials

- Design Strain Levels and Impact Damage
- Flammability and Fire Protection

BOEING

Figure 25

Technology Issues

Technology issues were identified in the areas of systems and manufacturing. The systems issues are those that are known to be influenced by a composite shell. Acoustic transmission of external noise is a technology issue since noise attenuation is influenced by mass. The electromagnetic effects and lightning protection of a composite fuselage are known to be significant technology areas where data bases will have to be developed. In the areas of manufacturing, fabrication, assembly, and quality assurance, technologies will have to be significantly upgraded to support production of cost effective composite fuselage structure.

Systems	Manufacturing
• Acoustic Transmission	• Fabrication
• Electromagnetic Effects	• Assembly
• Lightning Protection	• Quality Assurance

BOEING

Figure 26

Key Issue — Damage Containment

The key issue to be addressed in the current NASA-Boeing fuselage program is damage containment. In the design of a commercial transport, uncontained engine parts, uncontained high energy rotating machinery, and foreign object damage must be considered. These damage threats have sufficient energy to penetrate the composite shell and completely sever stringers and/or frames. Since the fuselage can be pressurized at the time the incident occurs, pressure as well as flight loads must be considered and the damage must be contained. For this design requirement, the present concern is the lack of a technology data base and the lack of verified analysis techniques.

- Damage Threat
  - Uncontained Engine Failure
  - Uncontained High Energy Rotating Machinery
  - Foreign Object Damage
- Damage Must Be Contained

BOEING

Figure 27

ACEE NASA Two Damage Containment Designs

Damage containment can be accomplished by applying two design approaches defined as no growth or growth and arrest. For the no growth approach, there is a damage size and strain level relationship where, if the damage size is less than the critical size at that strain level, then the damage will not grow. This design approach usually produces a heavier weight design since the design strains must be lowered by adding material. The growth and arrest design approach allows the damage to propagate to the boundaries of the panel where a damage arrest design feature stops the damage from further propagation. This design approach has been shown to be more weight efficient. The basic technology concern is that a limited data base exists for both design approaches.

- No Growth
  - Size vs Strain
  - Weight Penalty
- Growth and Arrest
  - Tear Straps
  - Weight Efficient
- Limited Data Base for Either Concept

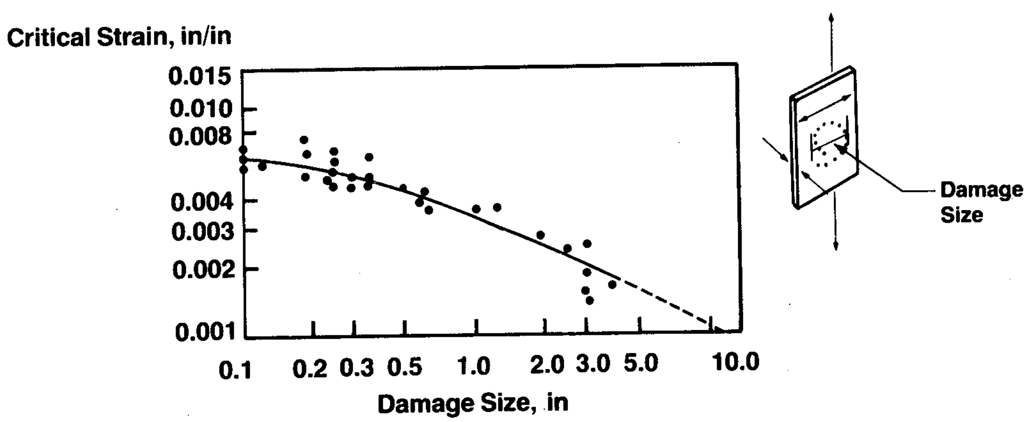
BOEING

Figure 28

Damage Size and Critical Strain

Fiber-Dominated Laminate

The critical damage size and strain relationship for commonly used graphite fibers is shown. The curve has been produced from industry data and shows that the largest damage that has been tested to date is approximately 3.5 inches. The design requirement is that a commercial transport fuselage must be damage tolerant to penetrations in the pressure shell up to approximately 12 inches. The primary concern is the shape of the damage size versus critical strain curve when it is extrapolated out to 12 inches.



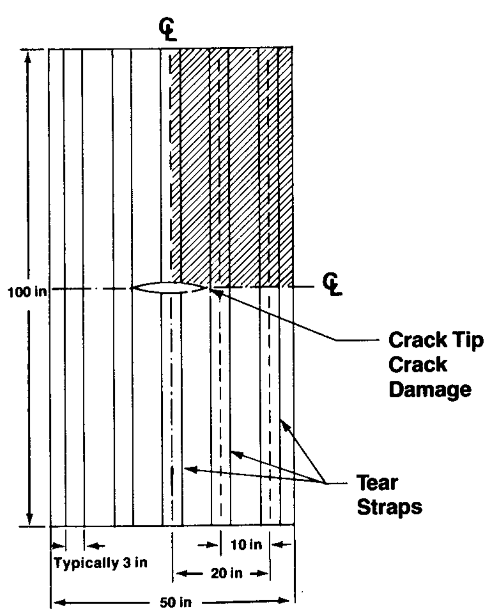
BOEING

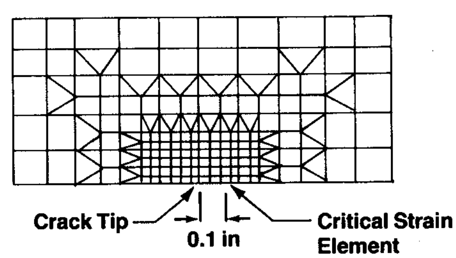
Figure 29

Fracture Panel Analysis

Tear Strap Design

To address the second approach to damage containment, an analysis procedure must be developed that will predict the requirements needed to arrest the damage. A finite element analysis procedure is being developed that will provide the information necessary to design tear straps for damage arrestment. The analysis model and the fine grid paneling around the crack tip is shown in the figure. The preliminary analysis is based on a critical fiber strain of 0.015 in/in located at the edge of an intense energy region, 0.10 inch in front of the damage. This critical strain and intense energy region approach has been verified by limited Boeing in-house testing.





Finite Element Model

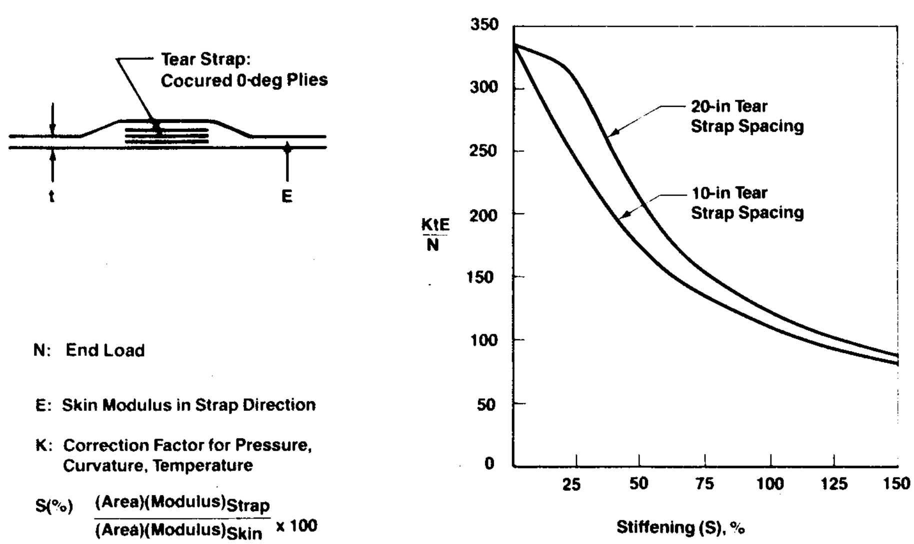
- Critical Fiber Strain 0.015 in/in Located in Front of Crack Tip 0.10 in

BOEING

Figure 30

Tear Strap Design

The analysis procedure discussed in Figure 30 has been performed and the resulting design curve for 10- and 20-inch spacing as a function of percent stiffening is shown. The design curve defines a percent stiffening required for a known end load with a defined extensional modulus and thickness of laminate. A correction factor for environment and curvature will have to be experimentally determined for each design case. The design curve is based on tear straps that are integrally cocured in the laminate as shown.

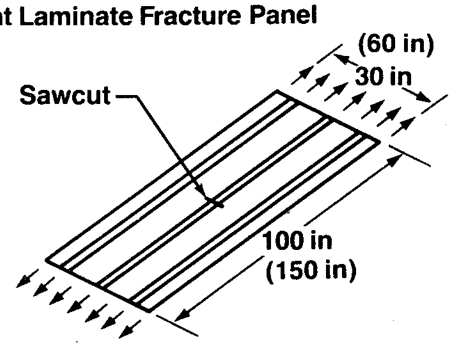
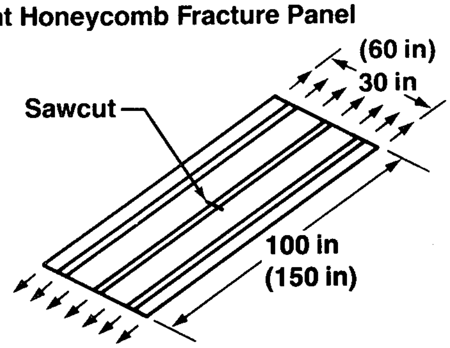


BOEING

Figure 31

Flat Fracture Panels

In the present NASA-Boeing fuselage program, flat fracture panels will be tested and evaluated to verify the tear strap analysis procedure outlined in Figures 30 and 31. A total of 10 laminate panels and two honeycomb panels will be tested. Tear strap spacing of both 10 and 20 inches will be evaluated. Low temperature tests, at -65°F, will be performed to determine the effect of temperature on the critical fiber strain.

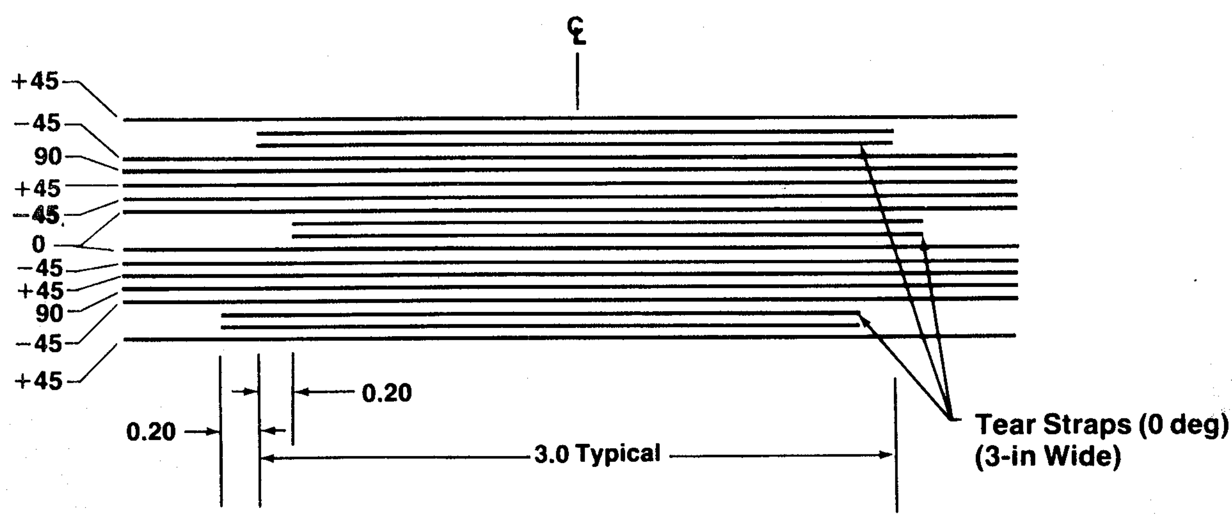
Test No.	Description	Purpose
1A	Flat Laminate Fracture Panel 	• Determine In-Plane Fracture Strains at Which: • Flaw Growth Is Initiated • Flaw Growth Is Arrested • Establish Analytical Data Base for Determining Pressure Containment Capability of Composite Fuselage Structure
1B	Flat Honeycomb Fracture Panel 	

BOEING

Figure 32

Tear Strap Detail

A typical 3-inch wide tear strap detail is shown. The tear strap contains six plies of 0-degree tape material. The plies are offset 0.20 inch to produce a taper along each edge of the tear strap when the panel is cured.

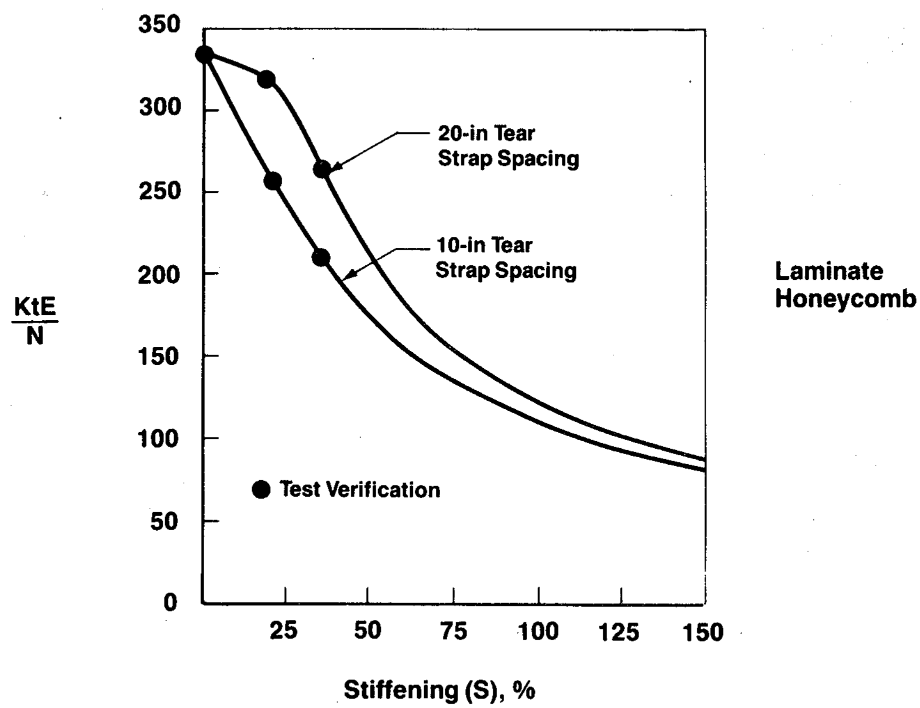


BOEING

Figure 33

Flat Fracture Panel Tests

The test program will produce the analysis verification points as shown. Two laminates with different extensional properties will be tested without tear straps to provide the data to extend the curve, shown in Figure 29, to 12 inches. As shown, two different percent stiffening ratios and different values of KtE/N will be tested. Laminate and honeycomb panels will be tested.

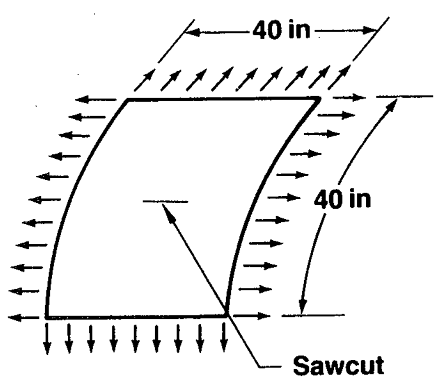


BOEING

Figure 34

Curved Laminate Fracture Panels

A curved laminate test program will be performed to determine out-of-plane pressure effects on the critical fiber strain. These tests will provide an empirical correction factor to be applied to the flat fracture panel test results to account for curvature and environmental effects. Biaxial load cases will also be evaluated. A total of eight panel tests will be performed.

Test No.	Description	Purpose
2	Curved Laminate Fracture Panel 	• Determine Out-of-Plane Effects on Fracture Strains at Which Flaw Growth Is Initiated: • Curvature • Pressure • Applying Damage While Panel Is Loaded • Evaluate Effects of Varying Panel Edge Supports (Axial or Biaxial) • Establish Analytical Data Base for Determining Pressure Containment Capability of Composite Fuselage Structure

BOEING

Figure 35

Summary

The planned program will address the critical technology issue of damage containment by conducting a test program that is directed toward building a data base necessary for the design of damage tolerant fuselage structure. Tear strap designs are required for minimum weight and stiffened laminate and honeycomb designs will be tested and evaluated. A toughened resin, high strain fiber material system, 2220-3/AS6, has been selected for the stringer stiffened laminate designs. The damage tolerance analysis verification tests have been defined and the first panel tests will be performed in the first quarter of 1985.

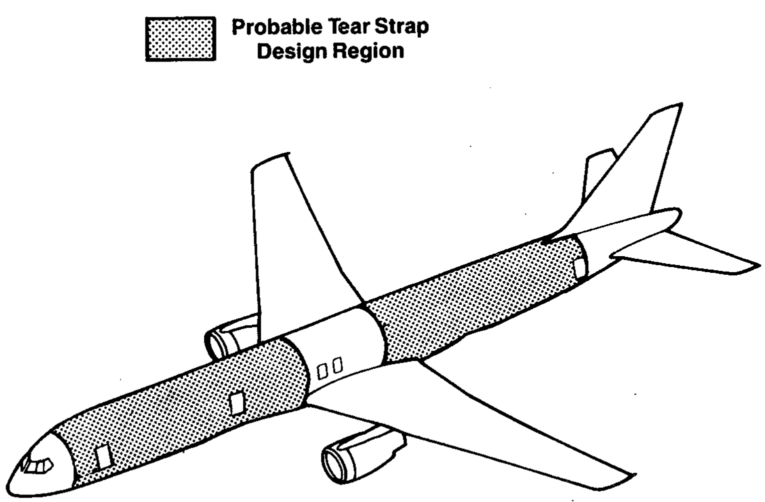
- Planned Program Addresses Critical Issue of Damage Containment
- Tear Strap Design Required for Minimum Weight
- Stiffened Laminate and Honeycomb Designs Will Be Tested
- Toughened Resin/High Strain Fiber System Selected
- Damage Tolerance Analysis Verification Tests Defined
- First Tests First Quarter 1985

BOEING

Figure 36

Damage Containment

As an example, the probable areas of a commercial transport that will require a tear strap design is shown. This example points out that damage tolerance requirements will influence a major portion of the fuselage.



BOEING

Figure 37

Selected Materials

As mentioned in Figure 36, 2220-3/AS6 tape material has been selected for the skins and stiffeners of the stringer stiffened laminate design. The 2220-3 resin material was selected since it has demonstrated improved toughness for impact damaged structure loaded in compression. The AS6 fiber was selected since it has demonstrated a high strain to failure value of 0.015 in/in. The high strain to failure performance of the AS6 fiber will strongly influence the pressure damage containment design. The material for fabricating the frame and shear ties will be a combination of 2220-3/AS4 tape and fabric. The AS4 fiber was chosen over the AS6 fiber since the higher strain to failure capability of the AS6 fiber is not required. At the present time, the honeycomb skin laminate material has not been selected.

- 2220-3/AS-6, Tape
  Skin-Stringer Laminate Design
- 2220-3/AS-4, Tape and Fabric
  Body Frames, Shear Ties
- Honeycomb Material
  Selection Not Finalized

BOEING

Figure 38

ACEE NASA

Selected Materials

As mentioned in Figure 36, 2220-3/AS6 tape material has been selected for the skins and stiffeners of the stringer stiffened laminate design. The 2220-3 resin material was selected since it has demonstrated improved toughness for impact damaged structure loaded in compression. The AS6 fiber was selected since it has demonstrated a high strain to failure value of 0.015 in/in. The high strain to failure performance of the AS6 fiber will strongly influence the pressure damage containment design. The material for fabricating the frame and shear ties will be a combination of 2220-3/AS4 tape and fabric. The AS4 fiber was chosen over the AS6 fiber since the higher strain to failure capability of the AS6 fiber is not required. At the present time, the honeycomb skin laminate material has not been selected.

- **2220-3/AS-6, Tape**
  Skin-Stringer Laminate Design
- **2220-3/AS-4, Tape and Fabric**
  Body Frames, Shear Ties
- **Honeycomb Material**
  Selection Not Finalized

BOEING

Figure 38

1. Report No.	2. Government Accession No.	3. Recipient's Catalog No.
NASA CR-172358		

4. Title and Subtitle	5. Report Date August 1984
ACEE Composite Structures Technology Papers by Boeing Commercial Airplane Company	6. Performing Organization Code
7. Author(s) John T. Quinlivan, Robert D. Wilson, Peter J. Smith, Ronald W. Johnson	8. Performing Organization Report No.
	10. Work Unit No.
9. Performing Organization Name and Address Boeing Commercial Airplane Company P.O. Box 3707 Seattle, Washington 98124	11. Contract or Grant No. NAS1-15025 NAS1-16863 NAS1-17740
	13. Type of Report and Period Covered Contractor Report
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, DC 20546	14. Sponsoring Agency Code 534-06-13

15. Supplementary Notes

Langley technical monitors: Andrew J. Chapman
Marvin B. Dow
Jon S. Pyle

16. Abstract

The NASA Aircraft Energy Efficiency (ACEE) Composite Primary Aircraft Structures Program has made significant progress in the development of technology for advanced composites in commercial aircraft. Under NASA sponsorship, commercial airframe manufacturers have demonstrated technology readiness and cost-effectiveness of advanced composites for secondary and medium primary components and have initiated a concerted program to develop the data base required for efficient application of safety-of-flight wing and fuselage structure. The third special oral review of the ACEE Composites Programs was held in Seattle, Washington, on August 13-16, 1984. The conference included comprehensive reviews of all composites technology development programs by ACEE composites contractors. In addition, special sessions included review of selected NASA-sponsored research and several important Department of Defense programs in composite materials and structures. The papers were compiled in five documents. Papers prepared by personnel from Boeing Commercial Airplane Company, Douglas Aircraft Company, and Lockheed-California Company are contained in NASA CR-172358, CR-172359, and CR-172360, respectively. Papers on selected NASA-sponsored research are contained in NASA CP-2321. Papers on selected Department of Defense programs in composites are contained in NASA CP-2322.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
Composite materials Composite structures Composite aircraft structures	[redacted]

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price
Unclassified	Unclassified	149	

N-305

Summary

The NASA-Boeing program goals are defined as shown. Progress to date has shown that the program goals are achievable and Boeing supports the Fuselage Development Program with confidence.

- **NASA-Boeing Program Goals**
  - **30% Shell Weight Reduction**
  - **10% Cost Reduction**
  - **20% Part Count Reduction**
- **Progress to Date Indicates Goals Achievable**
- **Boeing Supports Fuselage Development Program With Confidence**

[redacted] PAGE BLANK NOT FILMED

Figure 39

PAGE 136 INTENTIONALLY BLANK