ADVANCED COMPOSITES TECHNOLOGY PROGRAM

John G. Davis, Jr.
Manager, Structures Technology Program Office
NASA Langley Research Center

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INTRODUCTION

This paper provides a brief overview of the NASA Advanced Composites Technology (ACT) Program. Critical technology issues that must be addressed and solved to develop composite primary structures for transport aircraft are delineated. The program schedule and milestones are included. Work completed in the first 3 years of the program indicates the potential for achieving composite structures that weigh less and are cost effective relative to conventional aluminum structure. Selected technical accomplishments are noted. Readers who are seeking more in-depth technical information should study the other papers included in these proceedings.
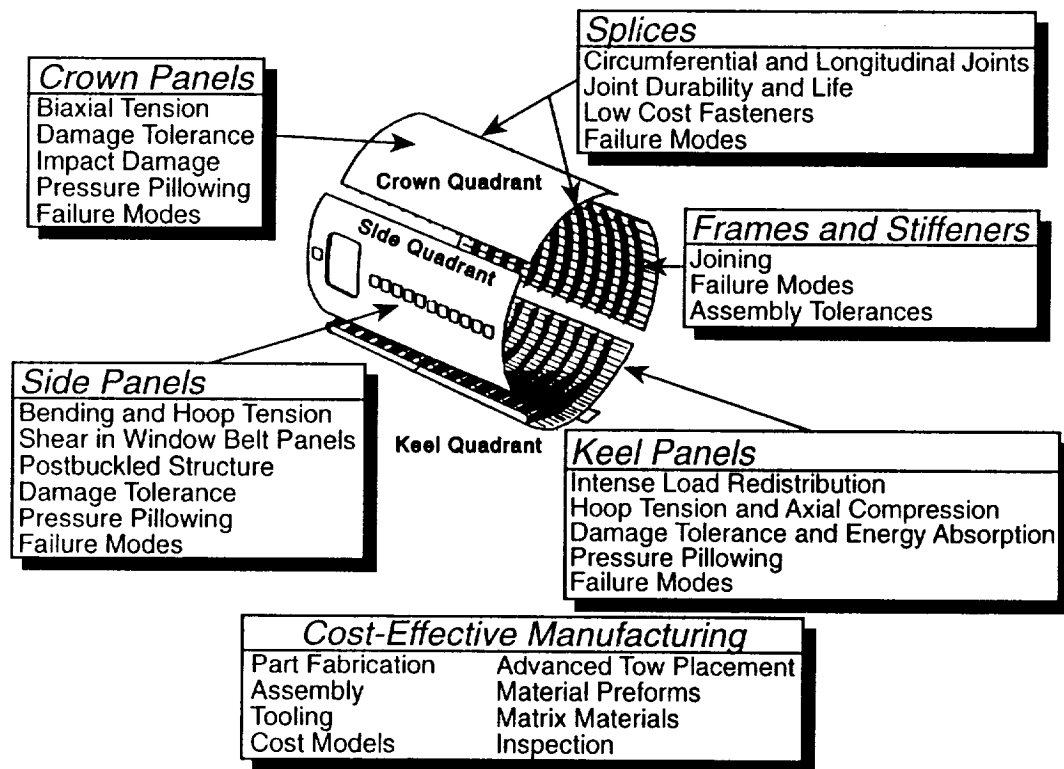
Outline

- ## Schedule
- ## Fuselage Crown
- ## Fuselage Side
- ## Fuselage Keel
- ## Wing
- ## Summary

There are numerous technology issues that must be solved to achieve cost-effective primary structure. Most are specifically related to various locations in the fuselage. This attributed to the variation in loads and design requirements for each portion of the fuselage. Biaxial tension appears to be the dominate loading for crown panels, but the other four issues listed are very important. Side panels are subjected to longitudinal bending, hoop tension, and shear, particularly around windows and doors. The keel panel is subjected to a large concentrated load at the forward end that must be redistributed to a uniform load at the aft end.
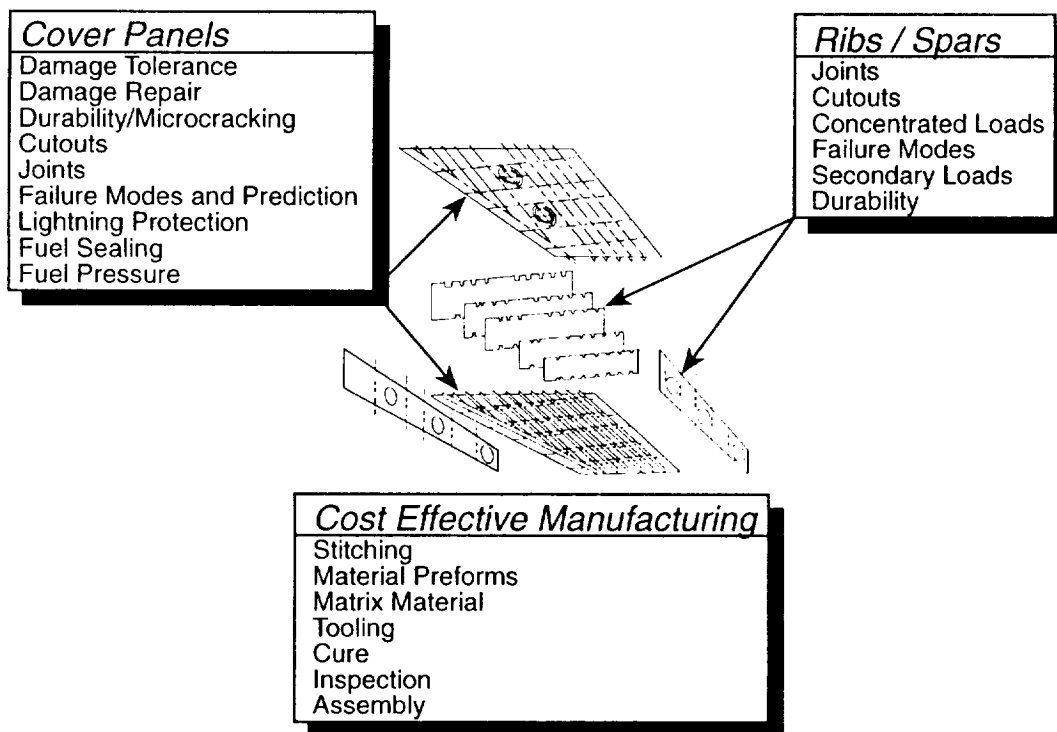
Splices, frames, and stiffeners are integral parts of each quadrant of the fuselage and directly influence structural efficiency, response to load, fabrication, and assembly cost. The most challenging issue is cost-effective manufacturing. This includes fabrication of components, inspection, and assembly, which must be competitive with metal airframe structure for wide-spread application to be achieved.

Splices
Circumferential and Longitudinal Joints
Joint Durability and Life
Low Cost Fasteners
Failure Modes

Crown Panels
Biaxial Tension
Damage Tolerance
Impact Damage
Pressure Pillowing
Failure Modes

Crown Quadrant

Side Quadrant

Frames and Stiffeners
Joining
Failure Modes
Assembly Tolerances

Side Panels
Bending and Hoop Tension
Shear in Window Belt Panels
Postbuckled Structure
Damage Tolerance
Pressure Pillowing
Failure Modes

Keel Quadrant

Keel Panels
Intense Load Redistribution
Hoop Tension and Axial Compression
Damage Tolerance and Energy Absorption
Pressure Pillowing
Failure Modes

Cost-Effective Manufacturing

Part Fabrication	Advanced Tow Placement
Assembly	Material Preforms
Tooling	Matrix Materials
Cost Models	Inspection

WING-BOX CRITICAL TECHNOLOGY ISSUES

Critical technology issues that must be solved to achieve wide-spread application
of composite structures in transport wings are related to the major sub-components.
There are nine major issues that influence design of the upper and lower cover panels.
Damage tolerance has been a dominate design requirement for conventional
two-dimensional reinforced composite panels and was the driving force for developing
toughened resin matrix materials. Ribs and spars are integral parts of the wing box and
significantly influence structural efficiency, response to load, fabrication, and assembly
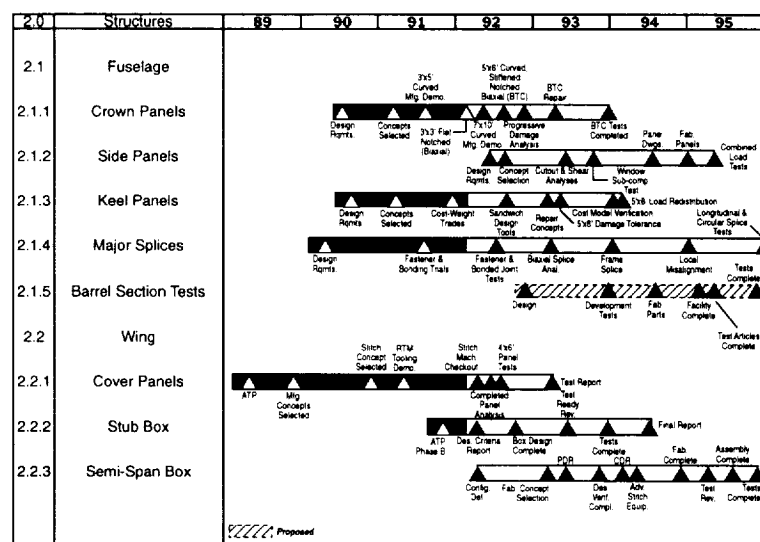cost.

Cost-effective manufacturing is also the most challenging goal for composite wing
structure. The seven specific technical issues listed are related to the dry fiber stitched
resin film infusion approach and are a major focus in the ACT Program.

Cover Panels
Damage Tolerance
Damage Repair
Durability/Microcracking
Cutouts
Joints
Failure Modes and Prediction
Lightning Protection
Fuel Sealing
Fuel Pressure

Ribs / Spars
Joints
Cutouts
Concentrated Loads
Failure Modes
Secondary Loads
Durability

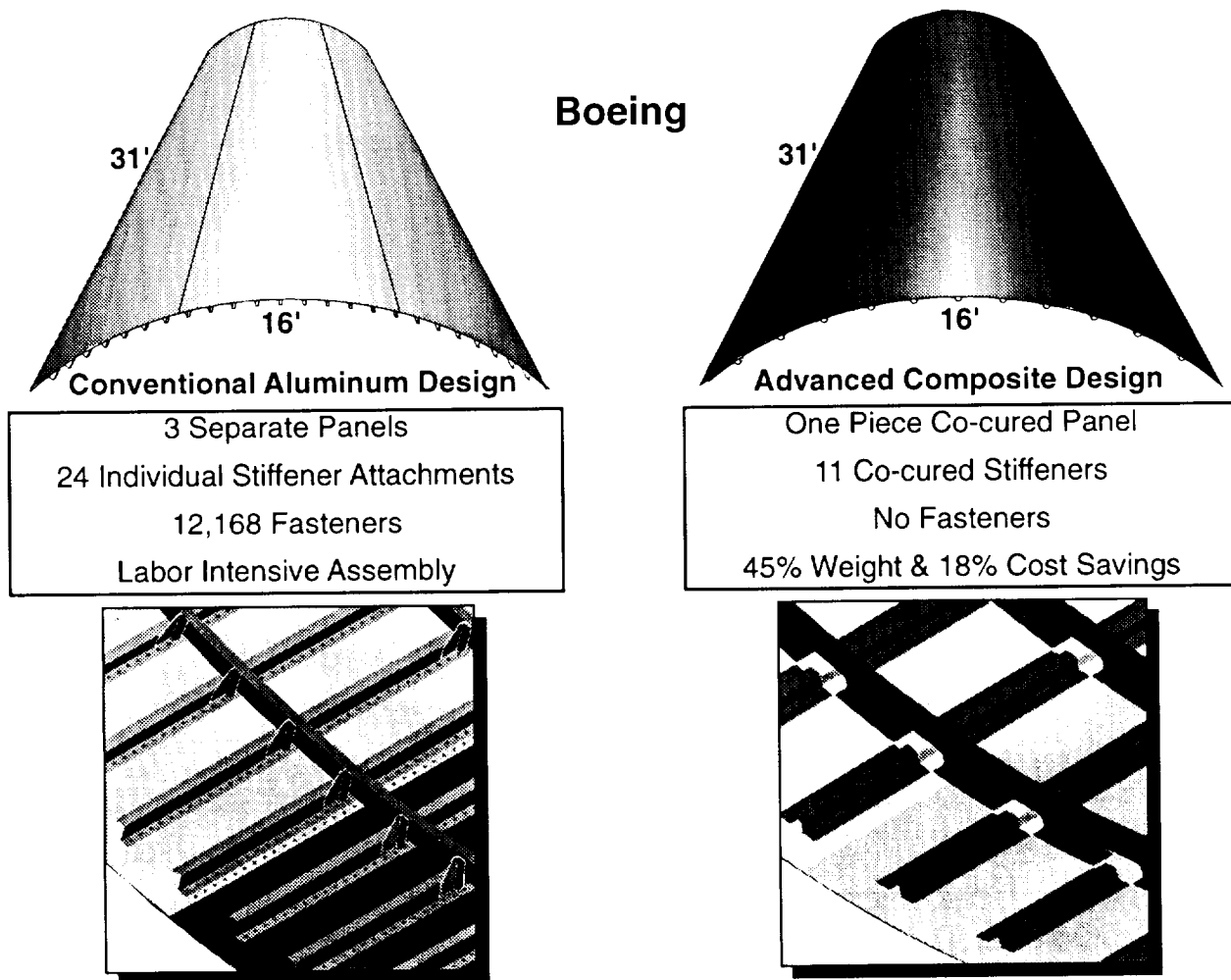| *Cost Effective Manufacturing* |
| --- |
| Stitching |
| Material Preforms |
| Matrix Material |
| Tooling |
| Cure |
| Inspection |
| Assembly |

Overall work breakdown structure, major milestones, and schedule for the ACT Program are depicted. During the first 3 years, major progress has been achieved in identifying materials, material forms, fabrication methods, and structural concepts that offer the potential for cost-effective composite primary structures. Exploitation of automated processes, particularly advanced tow placement, textile preforms, and dry fiber through-the-thickness stitched approaches has been emphasized. Major effort has been focused on development of crown quadrant panels and wing cover panels. More results are included in the proceedings for these two program elements than for the remaining elements. Curved panels, 7-feet by 10-feet in size, representative of the crown region, have been fabricated. Small and large panels have been subjected to biaxial tensile loads and a pressure-box test fixture that will be used to assess damage tolerance of skin-stiffened crown panels has been fabricated.

A wing cover panel, 6-feet by 4-feet in size, has been fabricated and tested. Dry fiber stitched resin film infusion with an un-toughened resin was used to fabricate the panel. Prior to resin impregnation, stiffeners and shear clips were attached to the external skin by machine stitching. Test results on small three-stringer and larger six-stringer panels indicate outstanding damage tolerance relative to conventional two-dimensional reinforced composite panels. Delamination was significantly reduced in the stitched panels.

| 2.0	Structures	89	90	91	92	93	94	95
2.1	Fuselage							
2.1.1	Crown Panels							
2.1.2	Side Panels							
2.1.3	Keel Panels							
2.1.4	Major Splices							
2.1.5	Barrel Section Tests							
2.2	Wing							
2.2.1	Cover Panels							
2.2.2	Stub Box							
2.2.3	Semi-Span Box							

CROWN PANEL

Aluminum and composite crown quadrant panels are depicted. The aluminum design requires three panels to cover the quadrant; contains 24 stiffeners and 12,168 fasteners; and assembly is very labor intensive. The composite design requires only one panel, contains eleven stiffeners and no fasteners for attachment of frames, stiffeners or longitudinal splices within the crown quadrant. Reduction in part count and assembly labor are major contributors to the projected cost savings of 18 percent. In addition, the composite design is estimated to weigh 45 percent less than the aluminum design.

Boeing

31' 16'
Conventional Aluminum Design

| 3 Separate Panels |
| 24 Individual Stiffener Attachments |
| 12,168 Fasteners |
| Labor Intensive Assembly |

31' 16'
Advanced Composite Design

| One Piece Co-cured Panel |
| 11 Co-cured Stiffeners |
| No Fasteners |
| 45% Weight & 18% Cost Savings |

NEW TEST FACILITY APPLIES BIAXIAL STRESS FIELDS
TO FLAT UNSTIFFENED NOTCHED COMPOSITE PANELS

A new testing machine shown in this figure was recently assembled at NASA Langley Research Center to apply biaxial stress fields to flat specimens. Specimens are 0.080-inch (11 to 13 plies) thick and 40-inch by 40-inch overall with an 18-inch-diameter test zone. Materials evaluated to date include AS4/938 graphite epoxy and hybrids of AS4/938 with various percentages of S2/938. Specimens were instrumented with 50 strain gages and included penetration-type damage simulated by a 2.5-inch notch at the center of the specimen. These tests are supplying much-needed data on composite skin panels with notches under biaxial stress fields to provide an understanding of composite behavior and damage tolerance.
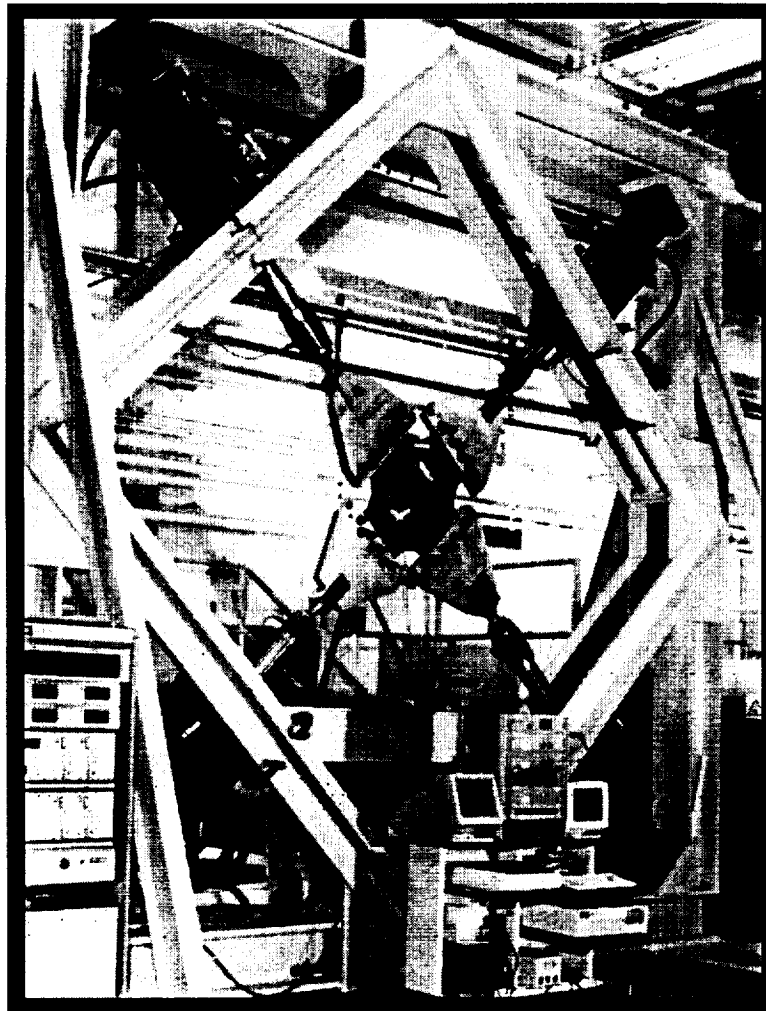
STRENGTHS OF BIAXIALLY LOADED PANELS

The behavior of flat unstiffened panels loaded in biaxial tension is shown on this chart. These panels were tested at NASA Langley Research Center with a 2-1/2-inch slit and two different load conditions. Predicted strengths of the biaxially loaded panels were made using a maximum strain criterion and test data from uniaxially loaded panels. The measured strengths compare fairly well with predicted values.
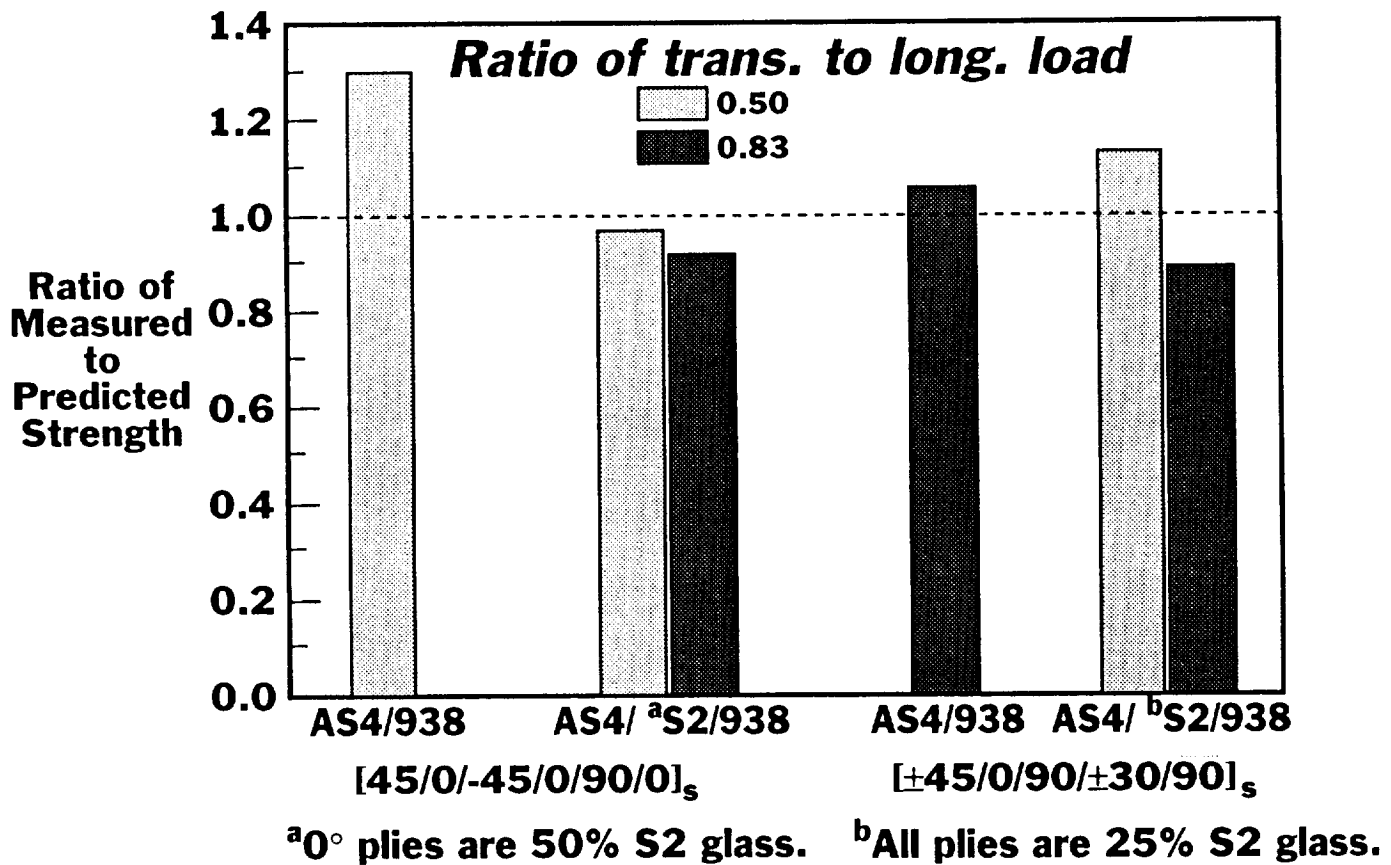
FIRST AXIAL DAMAGE TOLERANCE TEST OF
COMPOSITE FUSELAGE CROWN DESIGN

The left-hand photograph shows a five-stringer panel mounted in the testing machine. The overall panel dimensions are: 150 inches in length; 60 inches in width. Prior to testing a 14-inch-long sawcut notch which severed the center stringer and skin was made in the panel to simulate skin penetration and stringer damage. The right-hand photograph shows the notch and strain gage locations near the notch. Failure occurred when the end load reached 284 kips. Average tensile stress and strain at failure were 47 ksi and 0.0051 in./in., respectively. The margin of safety was 23 percent for the axial damage tolerance requirement. Details of these and other tests will be provided in a subsequent paper.
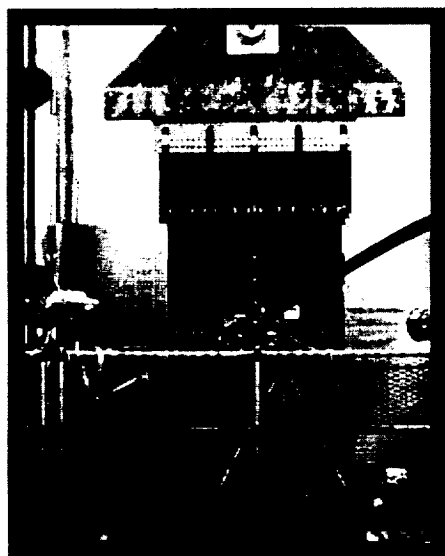
Test article mounted in testing machine
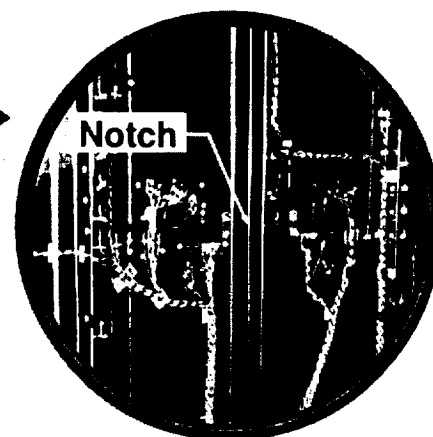
Closeup of test article showing notch and strain gauges

FUSELAGE CROWN PANEL STRUCTURE 7-FEET WIDE BY 10-FEET LONG FABRICATED WITH CO-CURED STIFFENERS

Boeing and Hercules are working jointly to develop fuselage crown structure concepts that incorporate advanced tow place skins, hot-drape formed hat stiffeners, and braided J-frames. The skin is laid on a mandrel with a computer-controlled process in which individual ribbon tows are precisely positioned with proper tension, direction, and contour. Stiffener preforms are hot-drape formed on separate male molds. Frames are braided, resin transfer molded, and cured separately. Assembly consists of placing the uncured tow-placed skin on an outer mold line caul plate, applying uncured stiffener preforms onto the skin, and positioning the cured frames over the skin and stiffeners with adhesive film between the frames and skin. The unitized skin-stiffener-frame combination is cured in a single autoclave cycle that cocures the skin and stiffeners while co-bonding the frames. The photo shows the 7-feet-wide by 10-feet-long sub-component size panel built at Hercules and representative of the section 46 fuselage crown panel for a Boeing wide-body airplane with a radius of curvature of 122 inches. The full-scale panel is 45-percent lighter and 18-percent less expensive to build than an equivalent metal crown panel. The composite crown concept also eliminates 12,200 fasteners compared to the metallic equivalent.
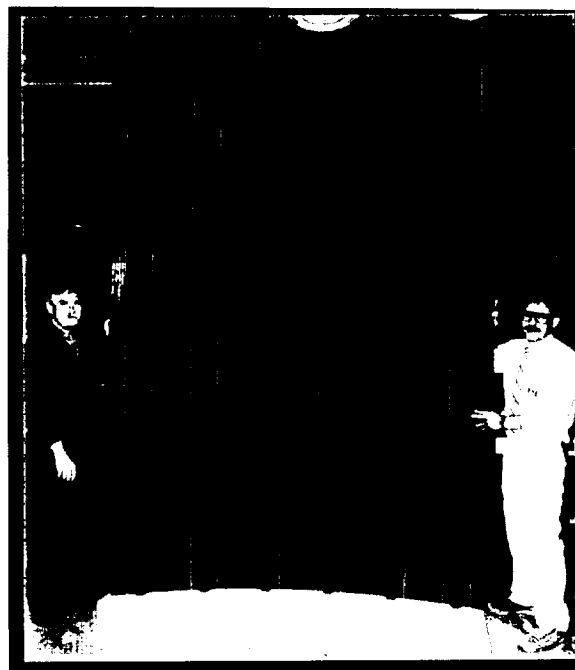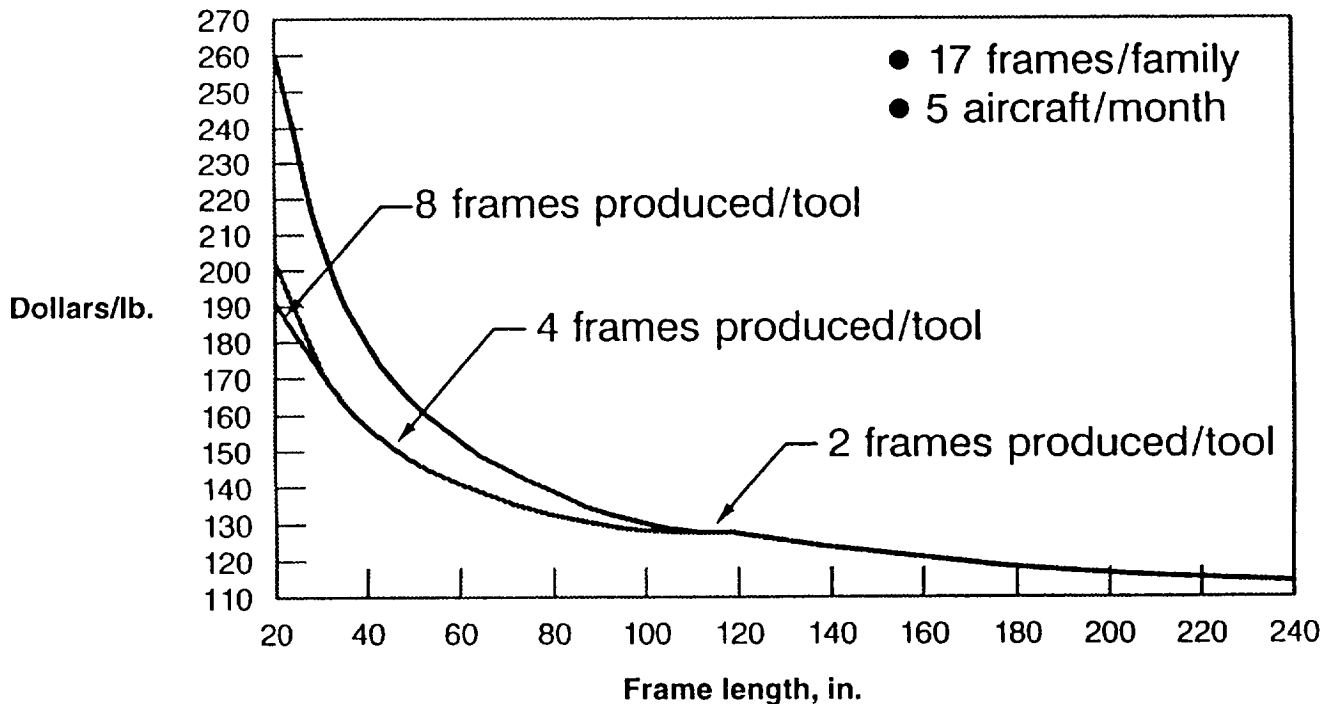
PRESSURE-BOX STRUCTURAL TEST FIXTURE

One of the important tasks in the ACT Program is to develop test fixtures that allow scale-up to large curved structures subjected to combined axial tension, hoop tension, and lateral pressure. The fixture shown in this photograph was designed and constructed at Boeing to test curved fuselage crown panels under combined loads. Maximum panel size is 72-inches long by 63-inches wide. The fixture can apply axial tension loads up to 12,000 lb/in. and a lateral pressure of 40 psi. After a series of initial tests at Boeing, the pressure box will be delivered to NASA Langley Research Center for use in the Benchmark Test Program.
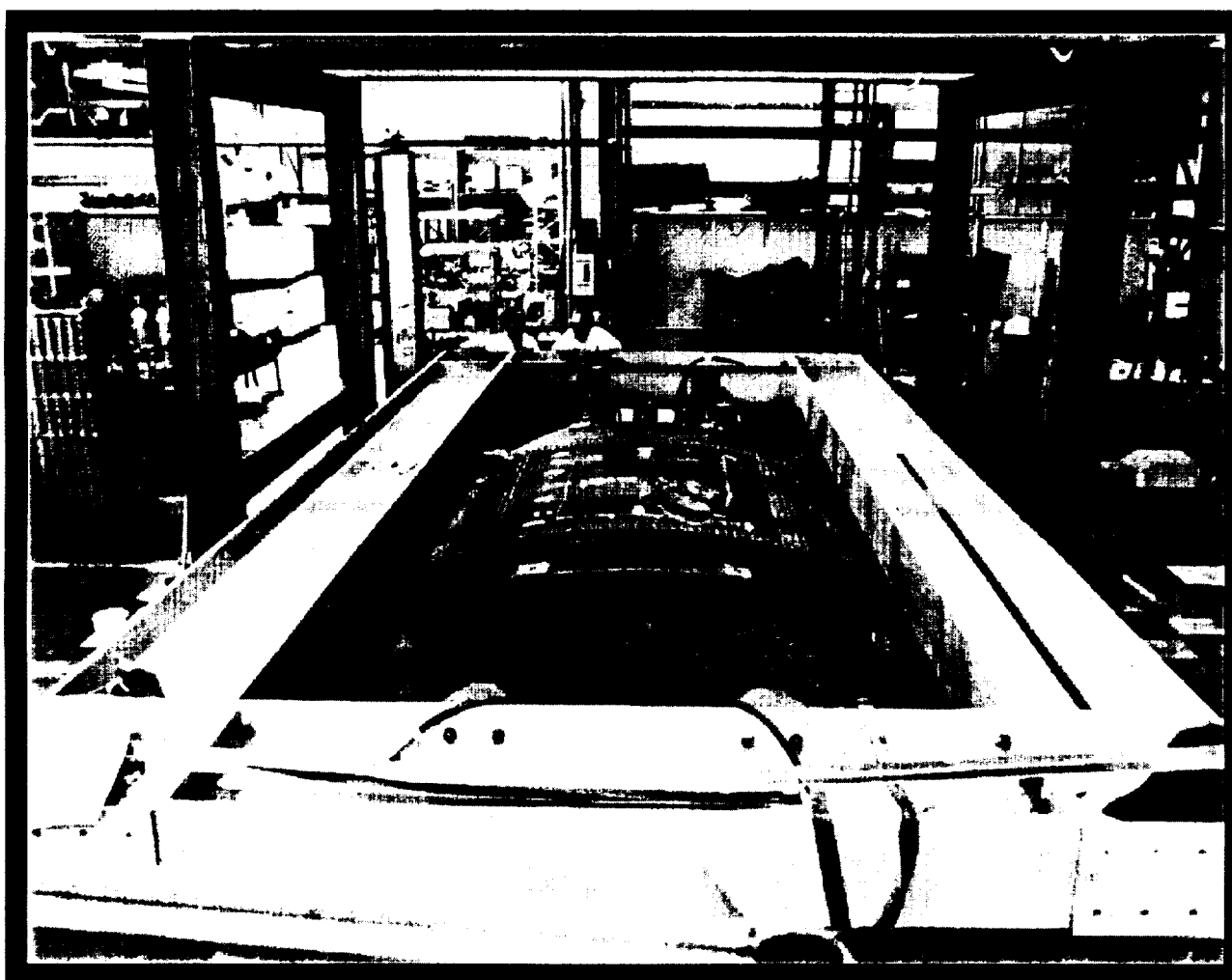
This figure shows the results for one of the cost model evaluations being developed at Boeing. The figure illustrates the strong relationship between cost and frame design details for a typical fuselage frame made by a braiding and resin transfer molding process. Cost for frame manufacture is reduced significantly for frame lengths of up to 100 inches. The figure includes the effect of process operations, amortizing tool set-up and batching of multiple frames per tool. Considerable details are provided on these cost studies in two subsequent papers.

Braid/RTM "J" Fuselage Frames

Chart: Dollars/lb. (vertical axis, 110 to 270) versus Frame length, in. (horizontal axis, 20 to 240).

Legend:
- 17 frames/family
- 5 aircraft/month

Curves labeled:
- 8 frames produced/tool
- 4 frames produced/tool
- 2 frames produced/tool

DESIGN/COST MODEL TRADE STUDIES
FOR THE FUSELAGE CROWN QUADRANT

This figure illustrates the application of the Boeing COSTADE designers' cost model to local optimization of a composite crown panel. The designers' cost model is based on simplified preliminary analysis design tools for tension damage tolerance, panel warpage, and buckling assessment. For small stiffener spacings the postbuckled design was more cost and weight efficient than the design constrained to resist buckling. The area labeled "transition zone" in the figure illustrates where the critical design constraints are changing. Beyond the transition zone, the design is driven by axial damage tolerance and buckling constraints. For larger stiffener spacings, postbuckling is no longer an effective way to save weight since significant material must be added to satisfy minimum buckling constraints. The incentive for developing the COSTADE model is demonstrated in this chart where the time needed to generate the analysis trends for various load cases, design criteria with cost considerations would be cumbersome if done by hand.

Problem Definition and Assumptions
- ATCAS fuselage crown geometry (122-in radius, 17.6 x 33.2 ft), loads, and criteria
- AS4/938 tow-placed laminates for skin and stiffener
- ATCAS crown panel designs ("stiffened sandwich" is a hybrid of families C and D)
  - Family C (cocured hat stiffeners with cobonded "J" frames)
  - Family D (cobonded "J" frames)
- ATCAS crown panel manufacturing plans (families C and D)

Note that relative trends in the graph are dependent on the body-bending loads.

GLOBAL OPTIMIZED CROWN MANUFACTURING
PLANS TO MINIMIZE ASSEMBLY COST

This chart illustrates the manufacturing detail that Boeing is evaluating to obtain cost estimates for composites versus metallic structure. The figure shows fastening approaches for longitudinal and circumferential splices of larger crown panels and frame splice close-out fastening for final assembly. The assembly cost contributors are shown in the bar chart as a percentage of total cost.
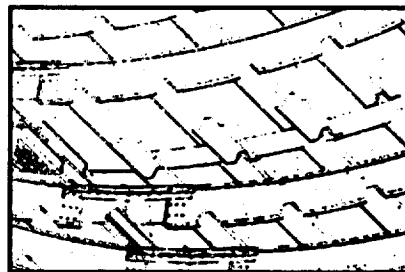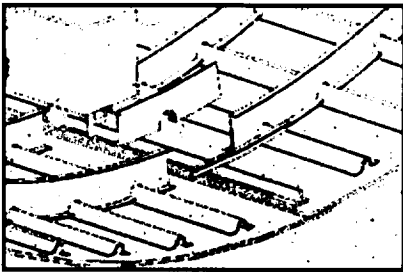

- Globally optimized family C
  - Includes major splices
  - Assumes 300 shipsets

NOTE: Percentages are of toal costs
(Recurring plus nonrecurring)
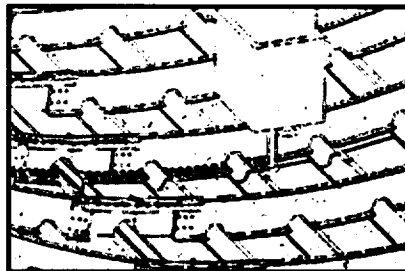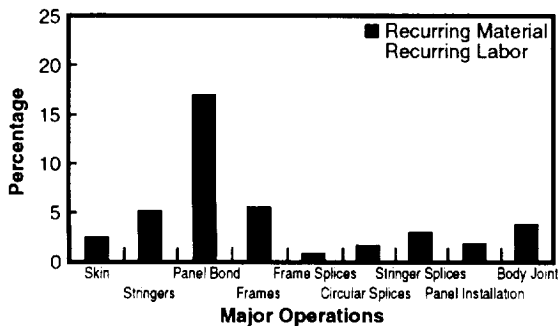
TAPERED KEEL PANEL

The figure shows a manufacturing demonstration panel for one concept Boeing is evaluating for keel structure. The panel has a 42-ply-thick laminate with several ply drop-offs to evaluate the effectiveness of the Hercules tow placement process in dropping plys without stopping the machine. This approach simulates the manufacturing process needed to make a single autoclave cure panel that transitions the high loads from the wing-body carry-through section to the more lightly loaded aft cargo area. Details of the Boeing baseline keel design are discussed in a subsequent paper.

AS4/8553 AFP
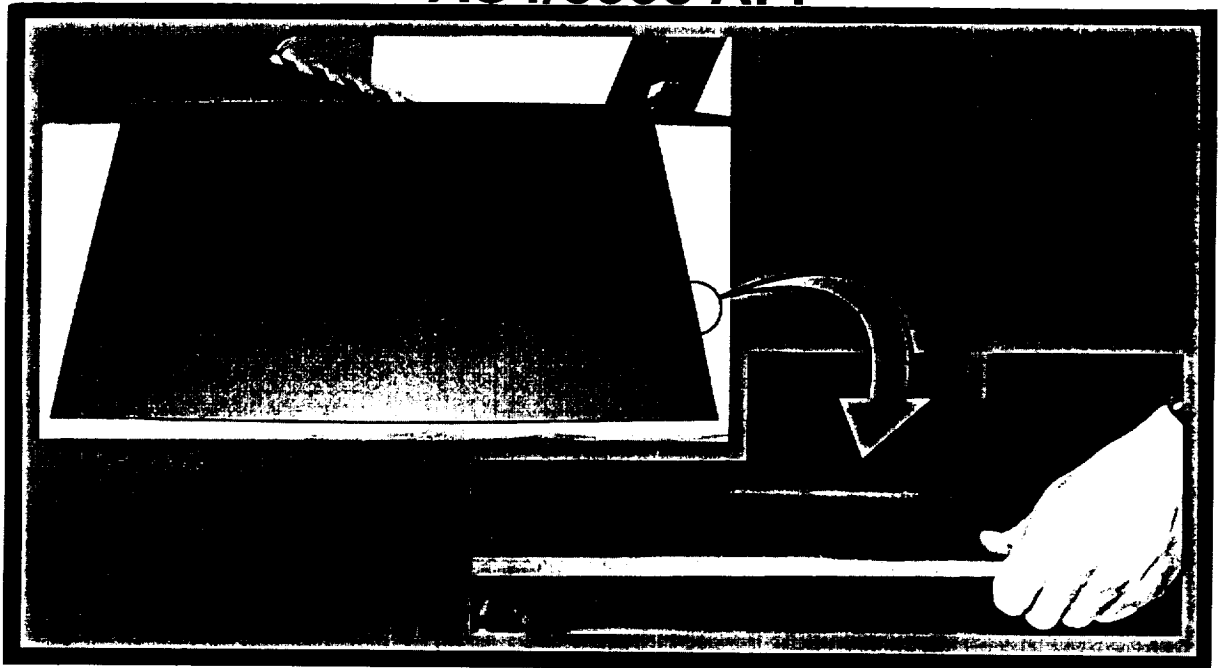
WINDOW-BELT REINFORCEMENT

The window-belt area offers the greatest potential to take advantage of the benefits offered by textile composites. The loading in the window-belt area requires reinforcement around the windows and adequate strength to transfer loads through the skin, the frames, and the stringers. Textile composites offer the best means of providing the needed strength and stiffness in this area. Lockheed is considering several textile composite window-belt concepts in their design trade studies. The concepts include 2D and 3D braiding of window-belt preforms to be attached to ATP skins as shown in the sketch. Their frames and preforms will be cocured or mechanically attached. Final details of these designs, and selection of the preferred concept, are pending the completion of detailed design studies and tests.

ATP Skin

One-Piece Window
Belt Reinforcement
3-D Woven

Frame - Precured & Cobonded
With Skin & Reinforcement
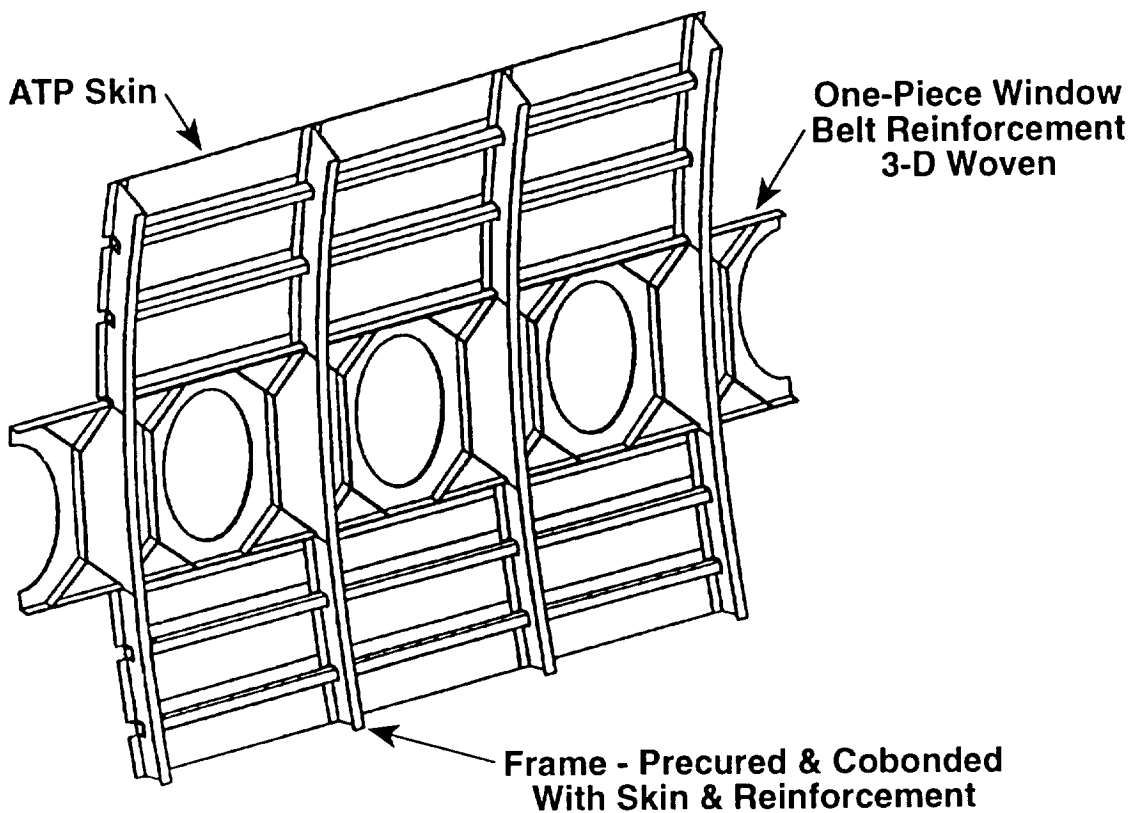
New processing techniques that offer potential for producing lower price material are being investigated. One of these processes is dry powder coating of fiber tows, and this sketch shows an approach being worked at Langley. This technique differs from the conventional cloud chamber method in that the powder is dropped directly onto the spread tow by a curtain, or waterfall, feeder. The tow in the chart is shown starting at the supply spool, running through the curtain feeder and the oven, then reversing and running back through another feeder and the oven again. The new system features a simple gravity feed, faster run speeds than the old cloud chamber method, reduced dust (fugitive powder) and improved safety for the operator, better control over resin content, and adaptability to multi-tow production. For multiple tows, the curtain is simply made wider to handle up to 40 or more tows instead of one.

## Powder Curtain Towpregging
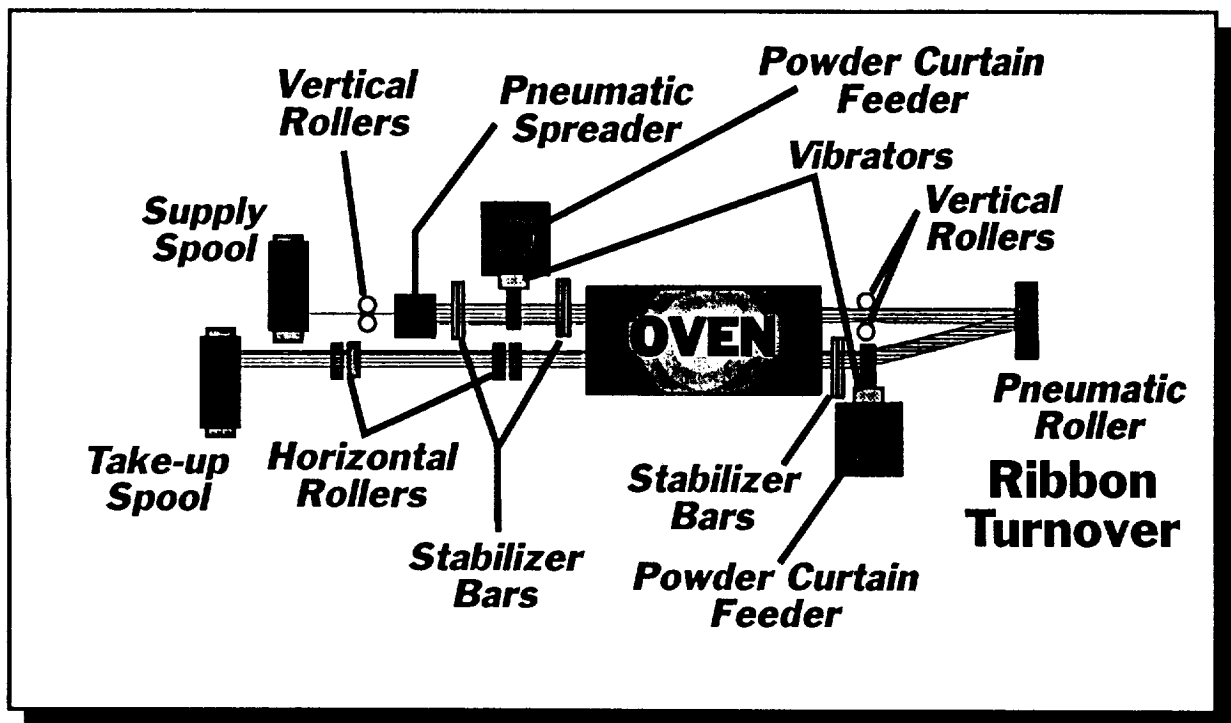
Double-Back Powder Coating Schematic (Top View)

CROSS-STIFFENED MEMBER

One of the advantages offered by textile composites is continuous through-the-thickness fiber reinforcement. For stiffeners that run perpendicular to each other, this could be used to make composite structure with superior strength over conventional laminated or bonded components. The panel preform shown in the figure is a proof-of-concept part woven of IM7 graphite fibers and is to be fabricated using RTM. The fibers in the stiffeners are continuous through the intersection, and the flanges of the stiffeners have continuous through-the-thickness fibers at the flange attachments to the skin. Such a fiber-strengthened arrangement is possible with innovative weaving techniques. Although test articles have been made, there are some obstacles to be addressed in scaling the process up to production sizes and quantities.

Woven Stitched IM7
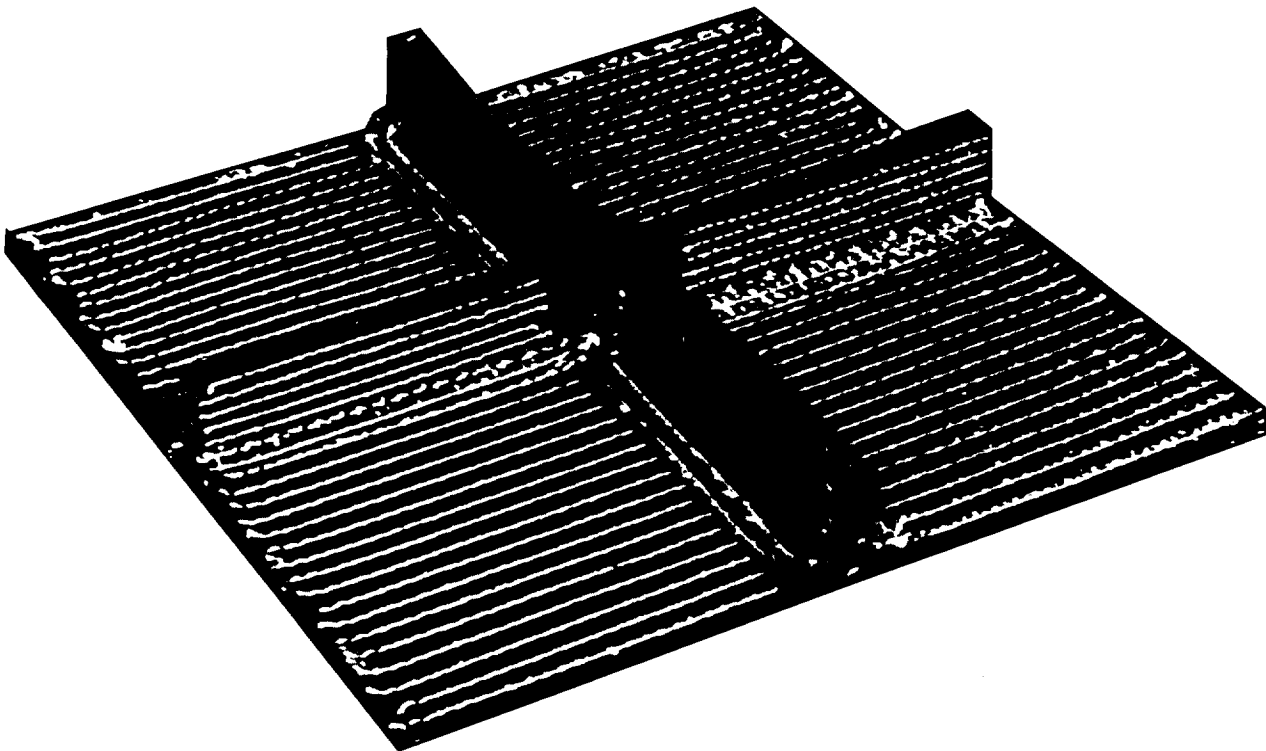
STRENGTH PREDICTIONS FOR BRAIDS USING DIAGONAL BRICK MODEL
(TECA)

Part of the effort in textile composites has been developing analysis methods to predict the strengths of various braided preforms. Shown in the figure are measured tension and compression strengths plotted for four braided configurations. The first type of braid has 40-percent axial tows, and the remainder are braided yarns. The second type has all braided yarns. Each of these two types is made using two braiding processes: 2D process with multiple layers of triaxial patterns; and 3D process which yields a single integrated layer. Strength predictions of these four configurations were made using a diagonal brick model or three-dimensional truss model. The predicted strengths were good compared with tests. The braids with axial tows typically show higher strength. The 2D-layered braids typically show higher strengths than the corresponding 3D single-layer braids.
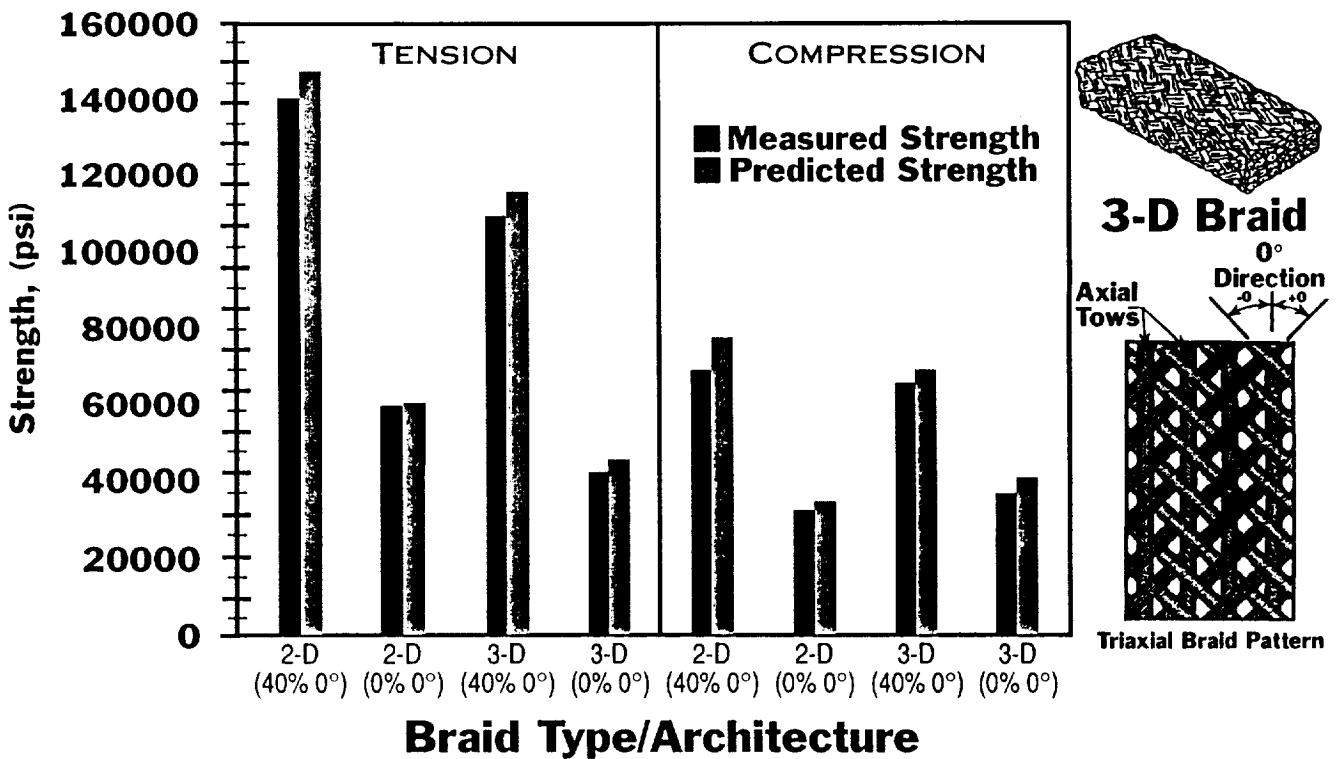
PREDICTIONS OF MODULII FOR 2D BRAIDS

Part of the effort in textile composites has been developing analysis methods to predict the modulus of braided materials. Shown in the figure are measured modulii for tension and compression in various 2D braids. All of the braids tested contained multiple layers of triaxial braid and had essentially straight yarns in the longitudinal direction. All of the braids had equal fiber volume fractions. Some braids used 6K yarn and some 12K yarn, and all of the braids were made over cylindrical mandrels with differing diameters. Both longitudinal and transverse modulii were measured, and the experimental results were essentially the same across the board. The predictions were made with three distinct theories: diagonal brick model (three-dimensional truss); composite lamination theory corrected for undulating braided yarns; and finite-element analysis that discretized yarn and matrix. All of the theories yield approximately the same predicted results. Except for the one transverse case, the predictions were very good compared to experiment.
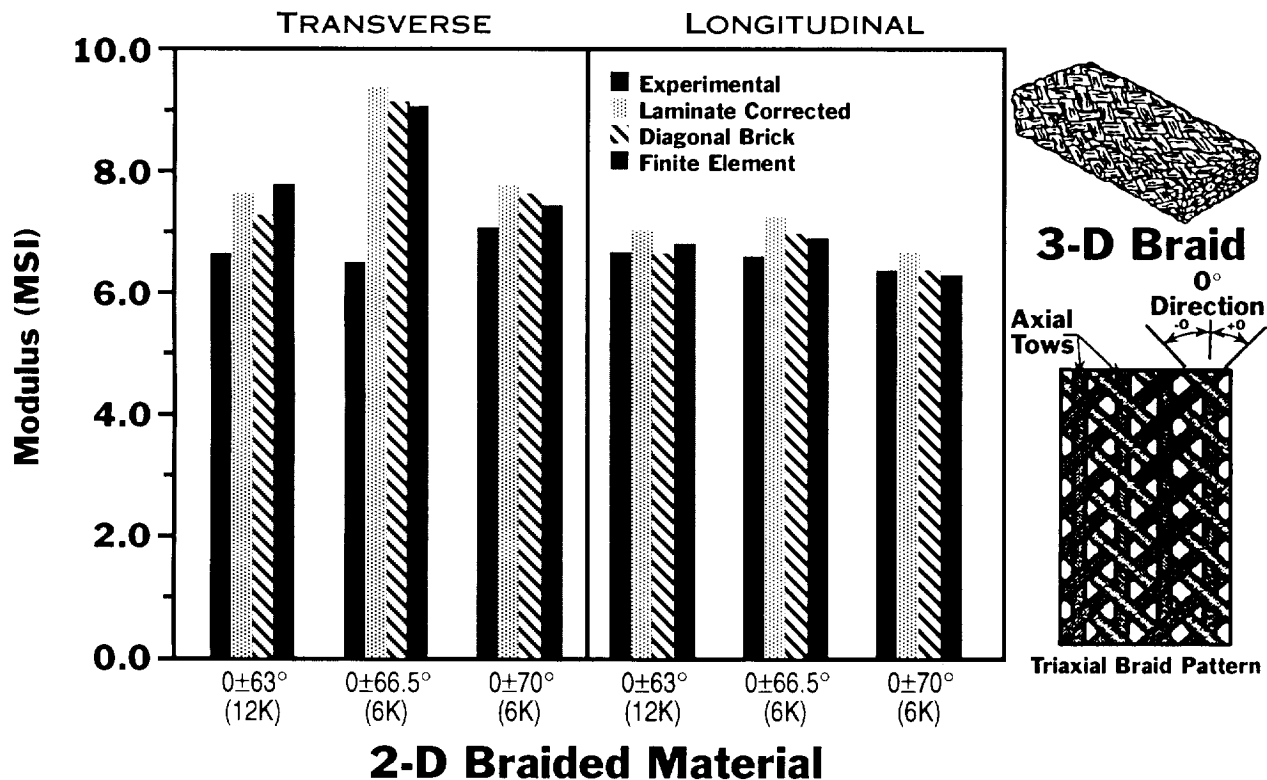
PROCESS CONTROL OF TEXTILE COMPOSITES

A recent workshop was held at NASA Langley Research Center with representatives from the textile industry and airframe manufacturers to assess where we are with quality control and where we need to be. While it is generally concluded that current post-process NDE methods can find most commonly occurring defects, this method is very time consuming and costly. Research is needed to develop effective in-process inspection methods that will preclude costly post-process requirements.
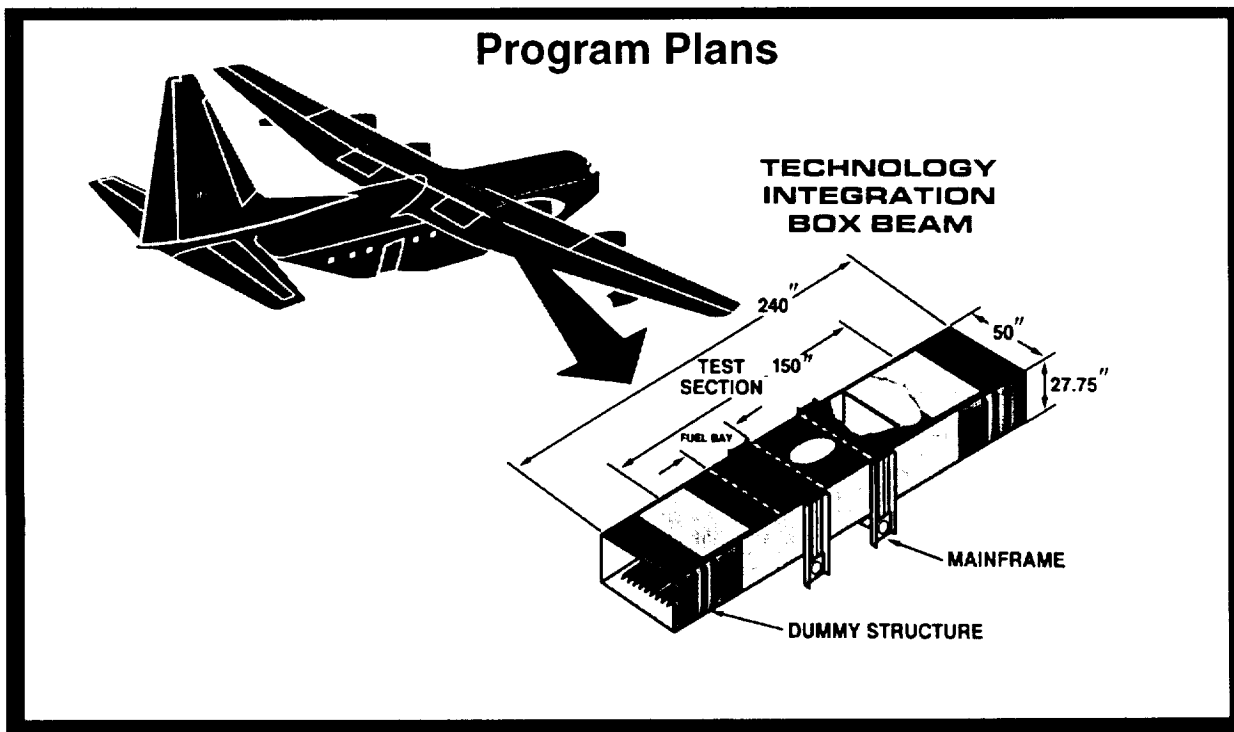
Industry consensus that:

- Current post-process NDE can find any of the commonly occurring defects but is not cost effective

- Post-process NDE must give way to "in-process" inspection/QA control procedures to remove "art" from shop floor

- Research is needed to develop in-process inspection/QA control techniques that will reduce the need for post-process NDE of each part

ADVANCED COMPOSITE STRUCTURAL CONCEPTS
## AND MATERIAL TECHNOLOGIES
TASK 5--BOX BEAM

Under Task 5 of the ACT contract NAS1-18888 with Lockheed, a wing box was built that represented a half-scale center box for the C-130 to understand the trade-offs between cost and weight of composite wing structure. The center wing-box beam for the C-130 aircraft was selected as the baseline for a typical heavily loaded wing structure. The box is approximately 240-inches long, including load introduction fixtures, and incorporated the best graphite composite fabrication technology available at the time. Design/manufacturing integration has provided an understanding of weight and manufacturing cost trade-offs for composite structures. This type of information will aide in developing the capability to design cost-effective composite structures that offer improved performance over aluminum structure.

Program Plans

TECHNOLOGY
INTEGRATION
BOX BEAM

240"

TEST 150"
SECTION

FUEL BAY

50"

27.75"

MAINFRAME

DUMMY STRUCTURE

TECHNOLOGY INTEGRATION BOX BEAM TEST DEMONSTRATES IMPORTANCE OF LOAD INTERACTION

The approach was to substantiate the strength of the structure using analysis and sub-component tests. The sketch in the upper left of the figure shows a separate cover assembly that was fabricated and tested prior to building the complete box beam. The cover specimen successfully supported ultimate load in compression. Subsequently, the cover was subjected to impact damage in two critical areas near the elliptical access hole. The damaged cover assembly also survived loading in compression to ultimate load. Fabrication of the box beam was then carried out, and tests were conducted at Lockheed's Kelly Johnson Research Center at Rye Canyon, California. In the final test, the box was loaded in up-bending and torsion. Failure occurred across the right side of the upper cover at 124 percent of design limit load (83 percent of ultimate), causing catastrophic damage to the right side cover and both spars. Since the cover panel alone survived ultimate load, component and load interaction apparently caused a compression failure adjacent to a transverse rib on the upper skin near the end of a truncated hat stiffener. Complete results of an analytical investigation of the TIBB were reported in the presentation entitled "Technology Integration Box Beam Failure Study," by Shuart, Ambur, Davis, Davis, Farley, Lotts, and Wang at the Third NASA ACT Conference, Long Beach, California, June 8-11, 1992.
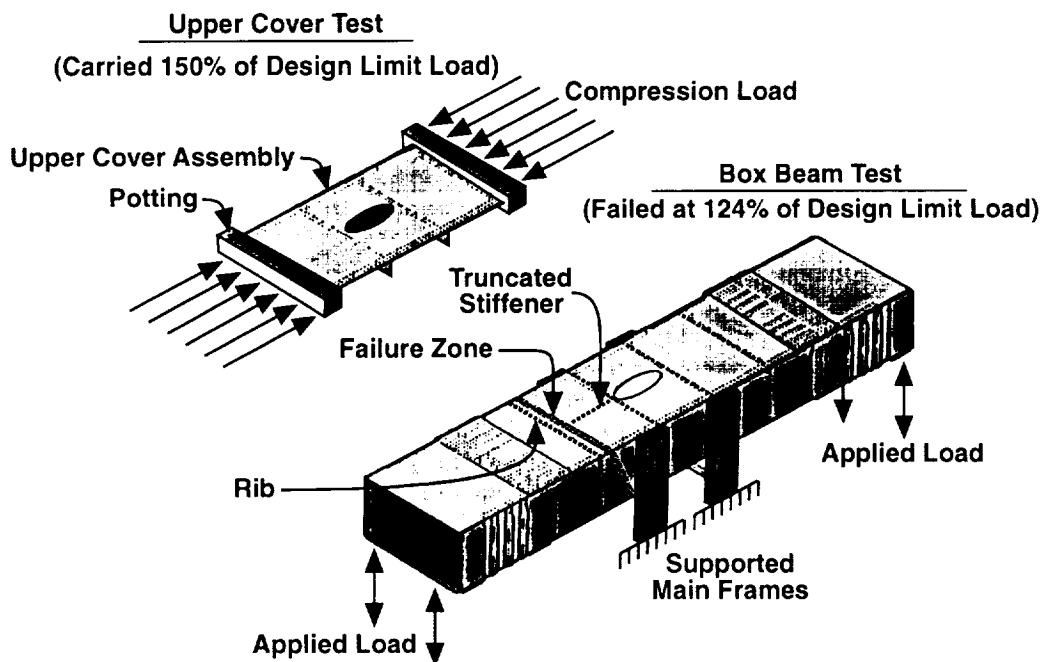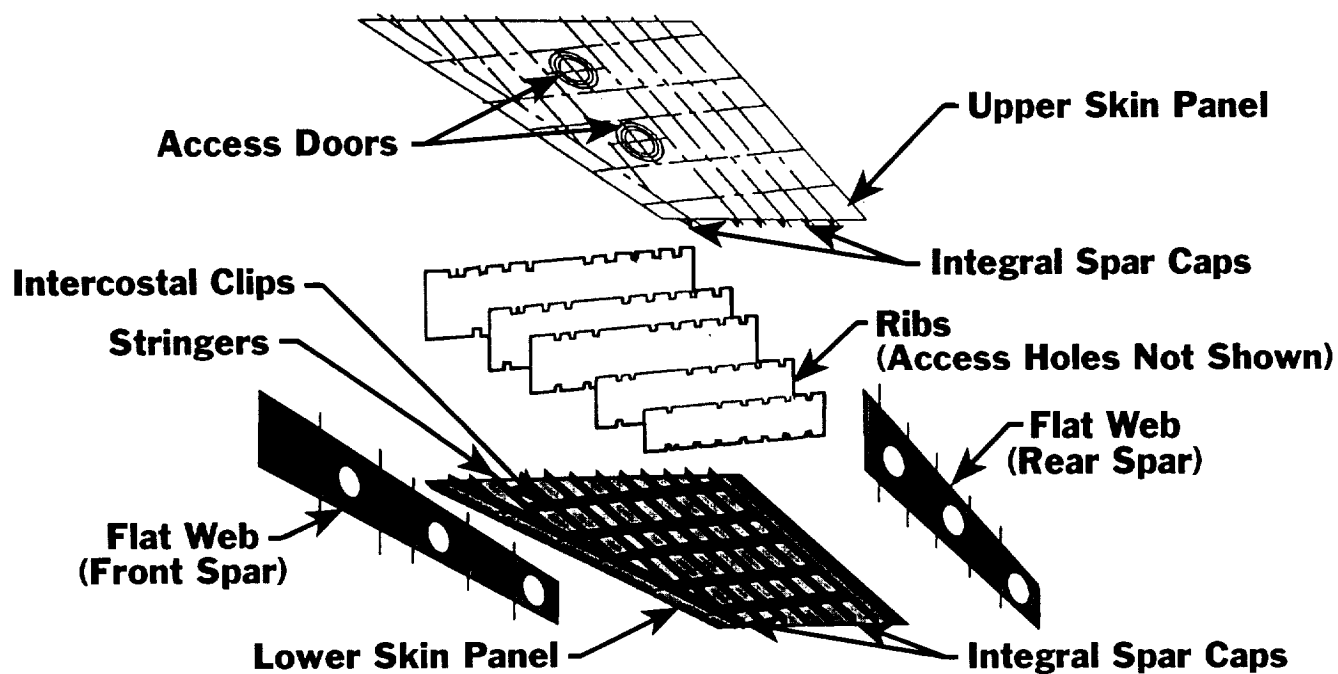
AXIAL DISPLACEMENT OF SRTS

An intense investigation of the failure that occurred in the TIBB was undertaken. The failure occurred in the TIBB upper cover at the point where the central hat-stiffener runs out adjacent to a rib. A Stiffener Run-out Test Specimen (SRTS) was cut from the undamaged side of the TIBB that was representative of the region that failed. The figure shows the axial displacement of the SRTS as it underwent axial compression in a uniaxial test machine. The end shortening analysis correlated well with test, with the analysis being stiffer than test. This was expected because the mesh size and element properties were slightly stiffer than the actual SRTS material they represented. Also, some rib rolling occurred in the SRTS test which was constrained in the analysis. The rib rolling became severe at the predicted failure load causing strain at one of the two ends of the stiffener run-out to intensify and the strain at the other end to lessen. Overall agreement between test and experiment for the SRTS was good, and the results supported the hypothesized mode of failure in the TIBB. Complete results of an analytical investigation of the TIBB was reported in the presentation entitled "Technology Integration Box Beam Failure Study," by Shuart, Ambur, Davis, Davis, Farley, Lotts, and Wang at the Third NASA ACT Conference, Long Beach, California, June 8-11, 1992.

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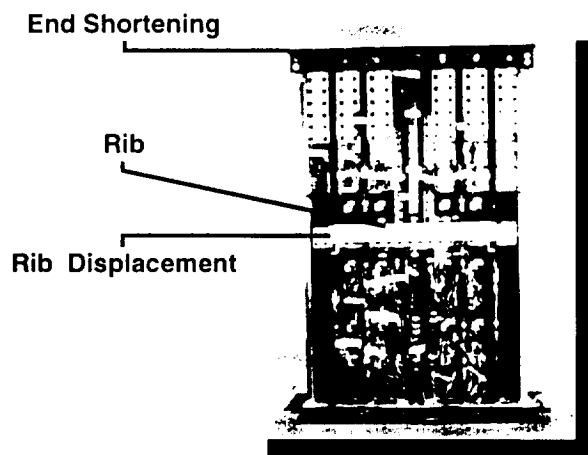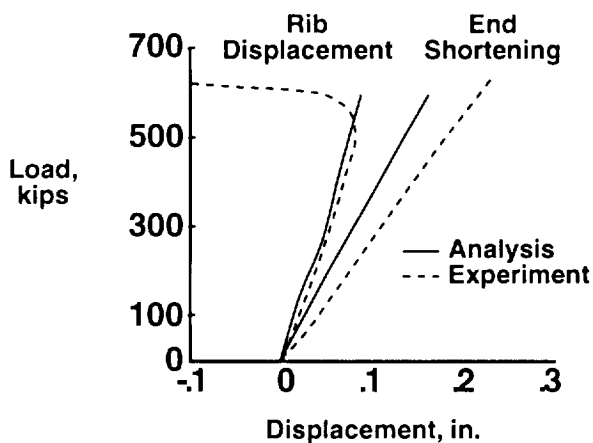
GTU AND ICAPS ACCELERATED WING DEVELOPMENT

Douglas Aircraft Company is the primary contributor to technology for transport wings. This figure shows the components of the ground test article (GTU) which will be manufactured and tested to validate RTM/stitched technology. This test unit, which is 8 feet at the root and has a span of 12 feet, is scheduled for demonstration tests beginning January 1994. Successful completion of these tests will pave the way for scale-up to a full-scale semi-span wing component. Specific details of this program will be provided in subsequent papers.

ICAPS Wing Box Ground Test Unit (GTU) Assembly

Access Doors, Upper Skin Panel, Integral Spar Caps, Intercostal Clips, Stringers, Ribs (Access Holes Not Shown), Flat Web (Rear Spar), Flat Web (Front Spar), Lower Skin Panel, Integral Spar Caps
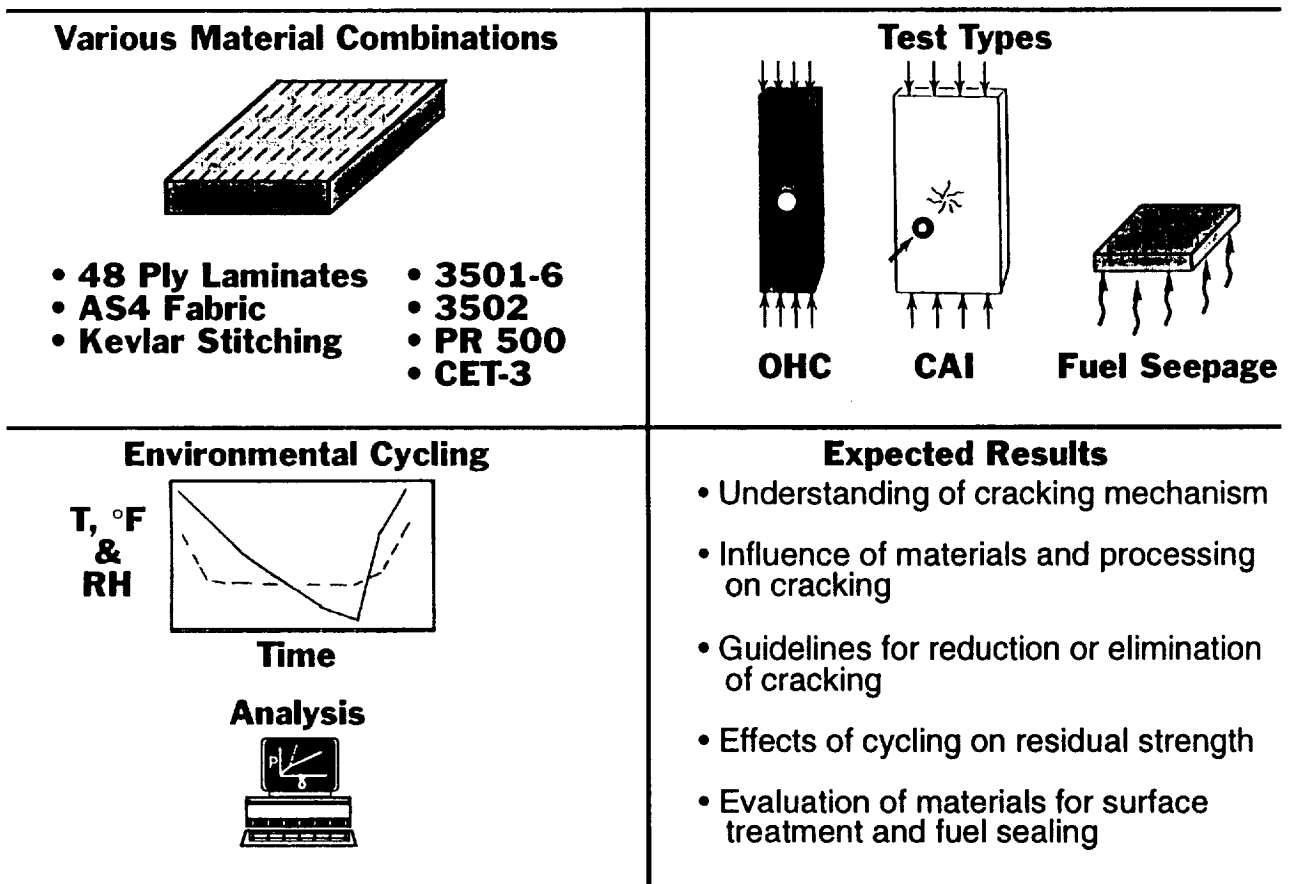
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FAILED THREE-STRINGER WING PANEL

One of the major advantages of dry stitching and resin transfer molding is the inherent ability of the structure to resist damage due to impact and, thereby, avoid large strength reduction because of delamination. This figure shows the failure mode of an RTM/stitched panel impacted at 100 ft-lb at mid-bay and tested to failure in compression. The suppression of delaminations results in a well-defined failure mode even for this relatively brittle material system.

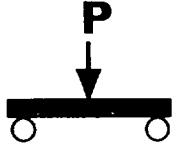## AS4/3501-6 Stitched Impacted 100 ft-lbs at Mid Bay
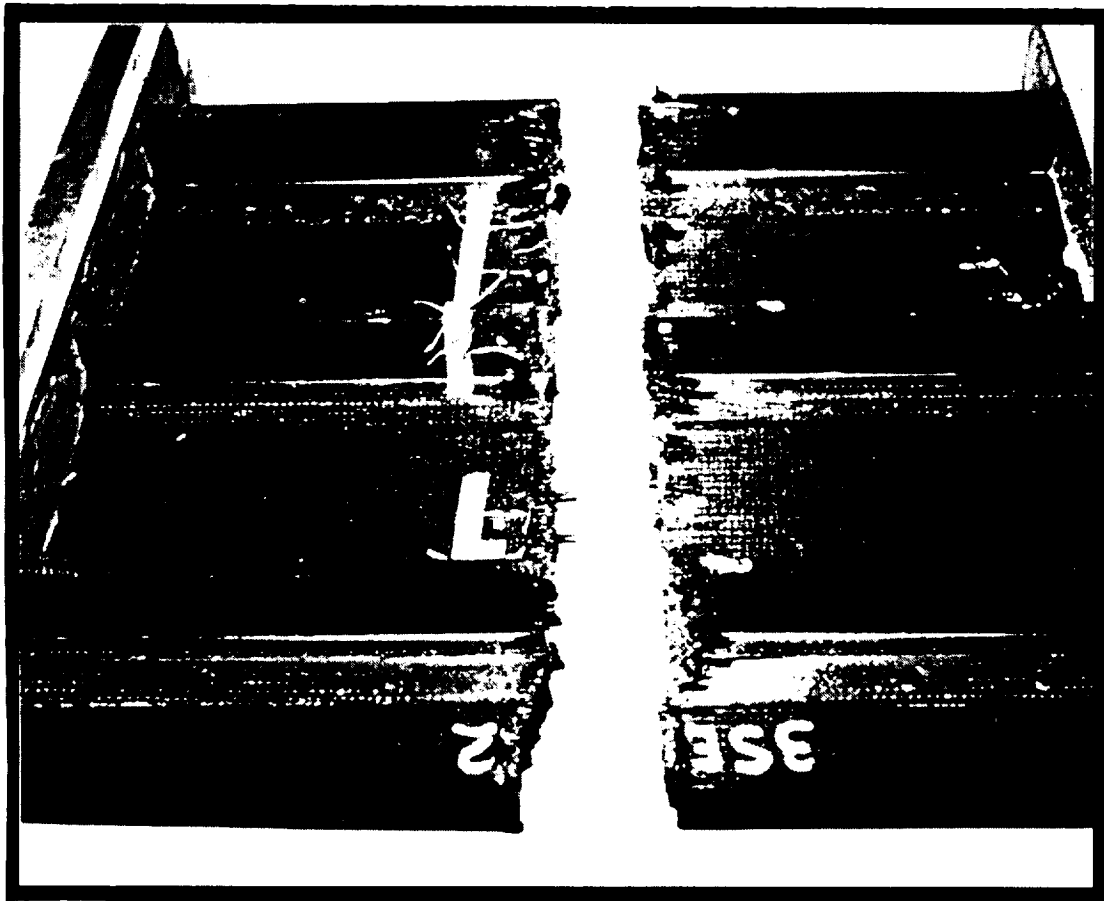
STUDY OF MICROCRACKING IN STITCHED COMPOSITES

Some early standard tests of RTM/stitched components have revealed microcracking believed to be associated with the stitching fibers. Though strength reduction due to microcracking seems to be insignificant and may not be a concern, the influence of microcracking over the long term does need to be evaluated. To better understand the cracking mechanism, a series of tests is being performed on different material systems, as shown on this figure. These tests will include the influence of material processing as well as environmental cycling.

| **Various Material Combinations**	**Test Types**
• **48 Ply Laminates** • **3501-6** • **AS4 Fabric** • **3502** • **Kevlar Stitching** • **PR 500** • **CET-3**	**OHC** **CAI** **Fuel Seepage**
Environmental Cycling **Time** **Analysis** 	**Expected Results** • Understanding of cracking mechanism • Influence of materials and processing on cracking • Guidelines for reduction or elimination of cracking • Effects of cycling on residual strength • Evaluation of materials for surface treatment and fuel sealing

AIRCRAFT FLUIDS AND FUEL EXPOSURE TESTS

The search for new and improved resin systems has resulted in several candidate materials suitable for resin injection. Tests are being conducted to study the influence of exposure of these materials to numerous fluids and aircraft fuel commonly used, as indicated on this figure. Significant details and results of this investigation will be presented in a subsequent paper.

Materials

Carbon Fabric
• 3K AS4 at 145 g/sq m

RTM Resins
• Hercules 3501-6 (Baseline)
• Dow CET-2 Epoxy
• Dow CET-3 Epoxy
• Shell 862 Epoxy
• Shell 1895 Epoxy
• BP E905 L Epoxy
• CG 5292 Bismaleimide

Test Specimen Fabrication

Resin Transfer Molding

8 Ply Tension

Fluids and Fuel Exposures

• Water at 160°F
• JP-4 Jet Fuel at RT
• Hydraulic Fluid at 160°F
• Turbine Oil at 160°F
• MEK Cleaner at RT
• Paint Stripper at RT
• Deicing Fluid at RT

Results

• Strength Data:
  - Cross plied tension at RT and 180°F
  - 0° shear at RT and 180°F

• Consistent comparative data on the behavior of state-of-the-art resins

ICAPS SEMI-SPAN WING DEVELOPMENT

In reviewing the technical needs of the scale-up from the stub box to the semi-span, there were several required sub-components to design, fabricate, and test. This illustration shows the tension and compression repair panels, similar durability panels, spar cap, and rib clip load transfer areas, the engine pylon attach areas, and two intermediate box specimens representing an inboard and outboard area of the semi-span. The box specimens are approximately 84-inches long and 24-inches wide with typical rib and spar attachments. Tests on these specimens should validate the structural integrity of the stitched/RTM concepts subjected to three-dimensional loadings.

Spar Cap

Pylon/Front Spar Attachment

Rib Clip Attachment

Intermediate Box Specimens

Compression

Tension

Major Repair Panels
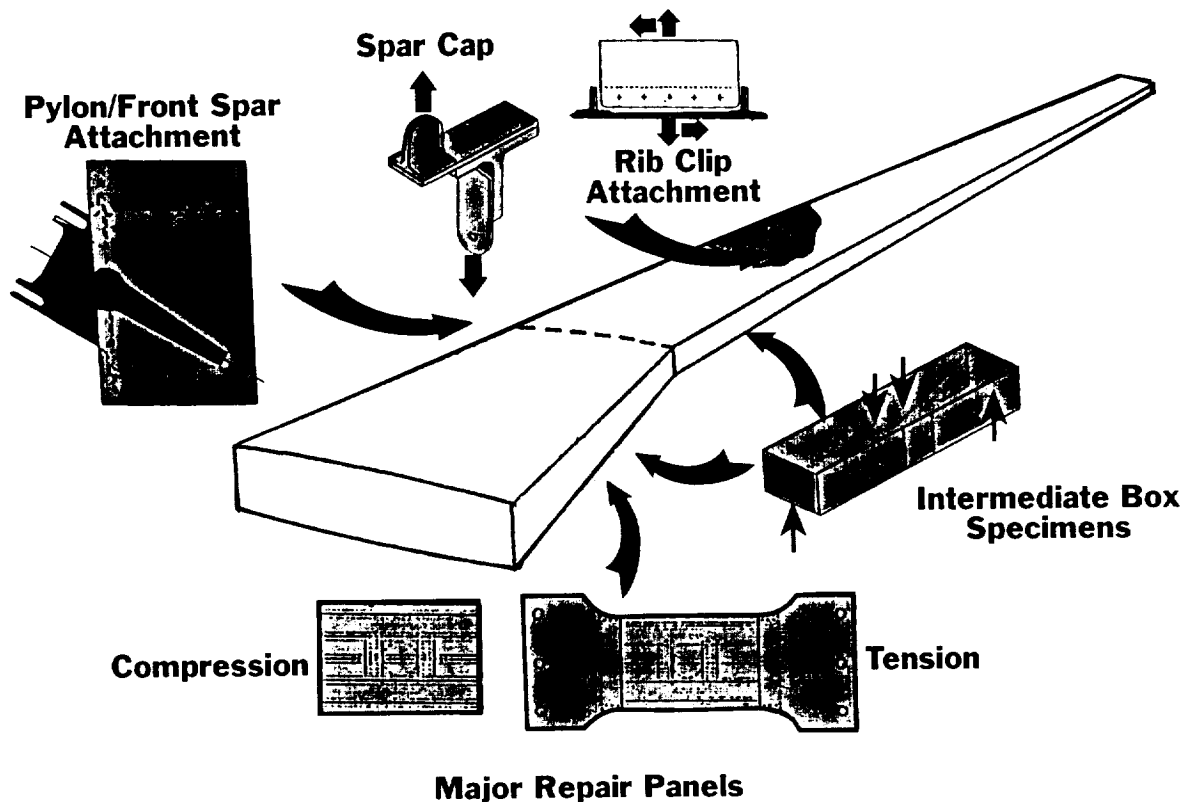
COMPOSITE SEMI-SPAN WING GROUND TEST UNIT

The successful culmination of the composite wing program will rely on the demonstration of technology with ground tests of a semi-span wing component shown on this figure. The wing is about 5.5 feet at the root and over 26 feet in span. Fabrication of this component will demonstrate the scale-up and cost effectiveness of RTM/stitched technology and should provide a wealth of data for analysis verification. This test program is expected to be completed by late 1995.
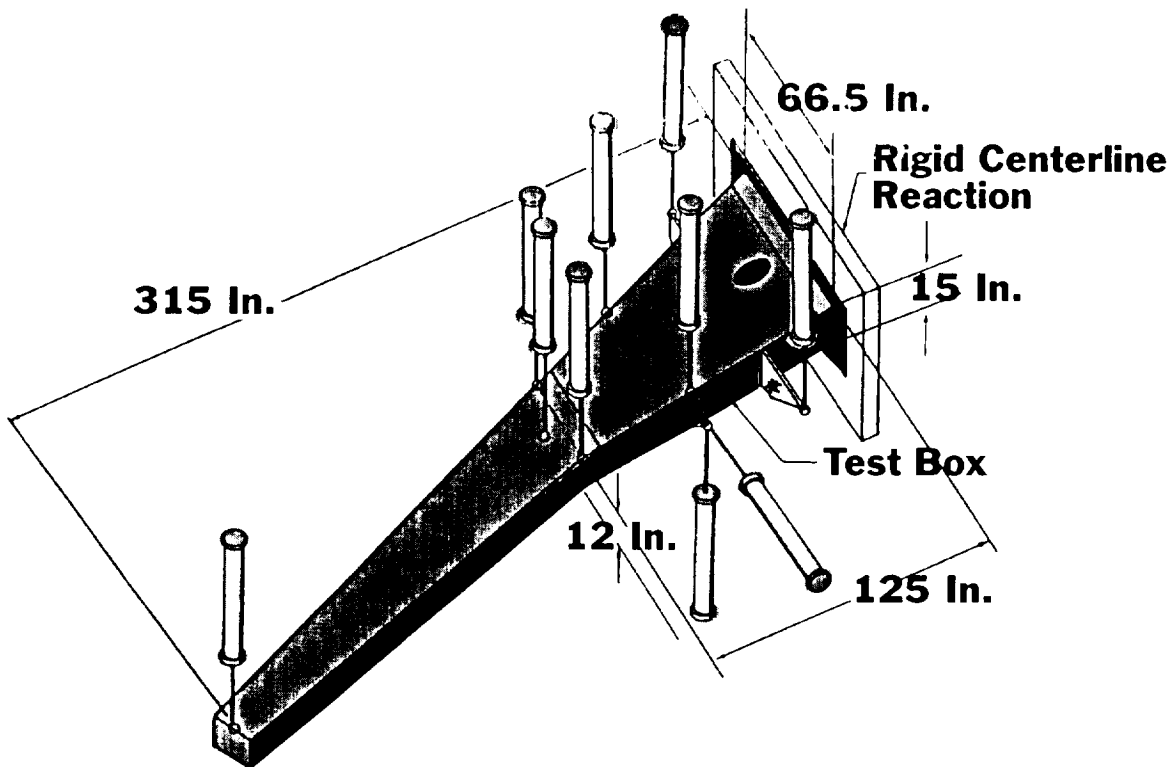
SUMMARY

Progress in the development of cost-effective composite structures over the first 3 years of the ACT Program has been very good. While the program has focused on wing and fuselage primary structure, the application of the three principal technology areas of automated tow placement, RTM/stitched, and textile preforms has provided significant insight into their weight and cost potential. The advantages of these technologies will be clearly delineated over the next 2 years.

Through close coordination with our transport manufacturers, a Phase C plan is evolving. This plan is expected to be approved beginning in 1995 and is designed to provide full-scale verification of an integrated data base grounded on analysis methodology, cost-effective manufacturing, test validation, and certification. The Phase C plan should provide the confidence and experience required for successful application of composites to fuselage and wing components of large transport aircraft. This plan should be well defined before our next conference which will be jointly sponsored by the Air Force and held in June 1993.

- Excellent progress is being made on development of cost-effective primary composite structures

- Development of Phase C Program Plan is underway

- NASA and Air Force will hold a joint Advanced Composite Technology Conference in June, 1993