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DEVELOPMENT OF STITCHED/RTM COMPOSITE PRIMARY STRUCTURES

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INTRODUCTION

The goal of the NASA Advanced Composites Technology (ACT) Program is to provide the technology required to gain the full benefit of weight savings and performance offered by composite primary structures. Achieving the goal is dependent on developing composite materials and structures which are damage tolerant and economical to manufacture. Researchers at NASA Langley Research Center and Douglas Aircraft Company are investigating stitching reinforcement combined with resin transfer molding (RTM) to create structures meeting the ACT program goals. The Douglas work is being performed under a NASA contract entitled Innovative Composites Aircraft Primary Structures (ICAPS). The research is aimed at materials, processes and structural concepts for application in both transport wings and fuselages. Empirical guidelines are being established for stitching reinforcement in primary structures and test data are reported in reference 1. New data are presented in this paper from evaluation tests of thick (96-ply) and thin (16-ply) stitched laminates, and from selection tests of RTM composite resins. Tension strength, compression strength and post-impact compression strength data are reported. Elements of a NASA Langley program to expand the science base for stitched/RTM composites are discussed.

Evaluation of Stitch-Reinforced Composites

Although great advances have been made in carbon fiber-reinforced composites, innovative concepts are needed to overcome the performance and cost barriers that limit the application of composites in aircraft primary structures. Thermoplastics and toughened epoxies provide improved damage tolerance and structural efficiency, but are considered too expensive for widespread application. Composite manufacturing methods used on production aircraft are still costly and labor intensive. In an effort to enable affordable and damage tolerant composite structures, Douglas Aircraft Company has adopted the approach shown in figure 1. Layers of dry carbon fabric are stacked in the desired ply orientation and the plies are stitched together using Kevlar or glass thread for through-the-thickness reinforcement. The stitched preform is then impregnated with resin and cured in a resin transfer molding (RTM) process.

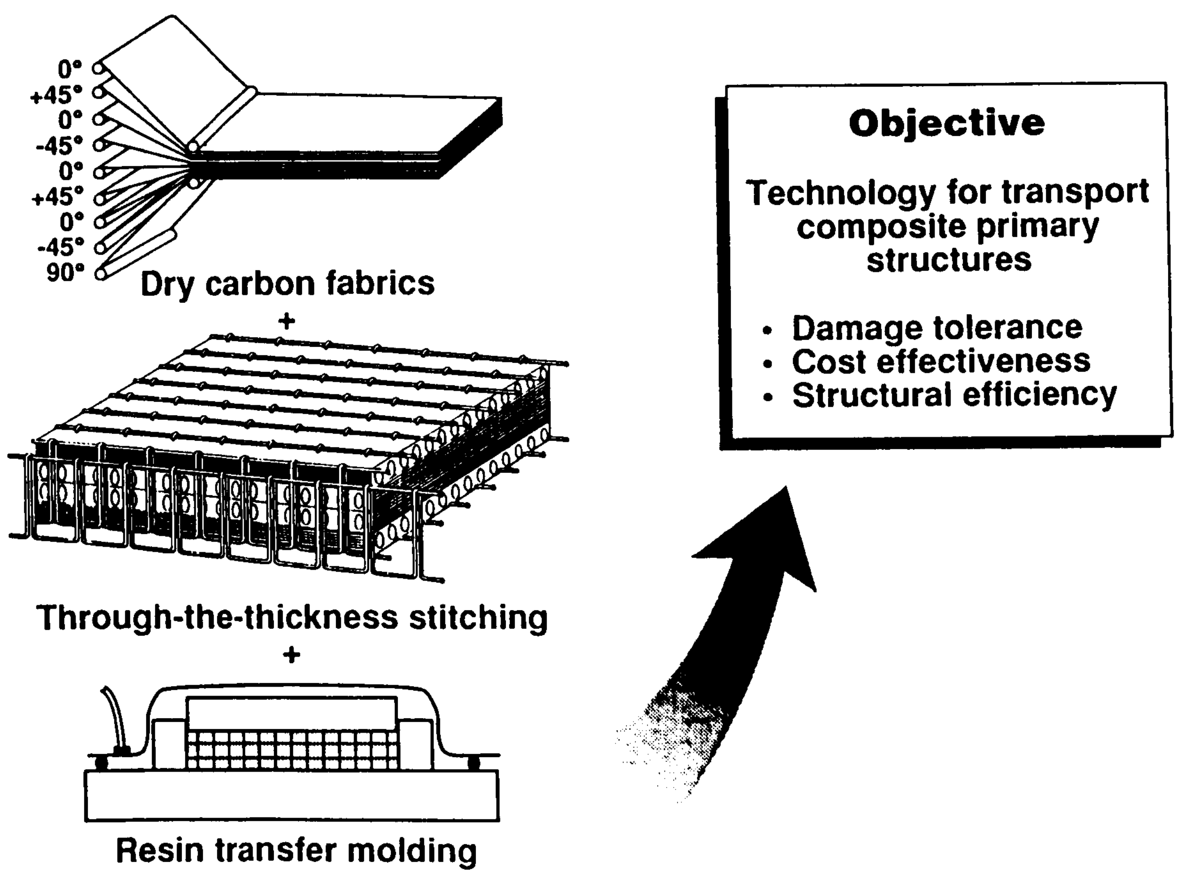


Figure 1

Development of Stitched/RTM Primary Structures

As part of the NASA Advanced Composites Technology (ACT) Program, Douglas Aircraft Company is developing unique composite materials and processes for transport aircraft primary structures. An outline of the Douglas contract is shown in figure 2. Phase A - Concepts Development is currently in progress and involves stress analysis of stitched composite aircraft structures and the establishment of a supporting database of stitched/RTM composite properties. Two RTM processes are being developed, one for wings and one for fuselage structures. For heavy wing structure, the process is resin film infusion with autoclave curing. For fuselage structure, the process involves fixed volume tooling and pressure RTM. Details of these RTM processes are given in the paper by A. Markus titled "Resin Transfer Molding Technology for Composite Primary Wing and Fuselage Structures." Testing of the wing and fuselage elements is currently in progress.

The planned Phase B - Technology Verification will consist of building and testing a 12 ft. by 8 ft. wing box with stitched upper and lower cover panels. A fuselage barrel section (150 inches long by 100 inches in diameter) will be built and tested. Two benchmark fuselage panels, a lower side panel and a crown panel, will be built for testing at NASA Langley Research Center. Two other major airframe manufacturing companies will also build benchmark panels for NASA Langley tests.

Douglas Aircraft Co. Contract NASI-18862

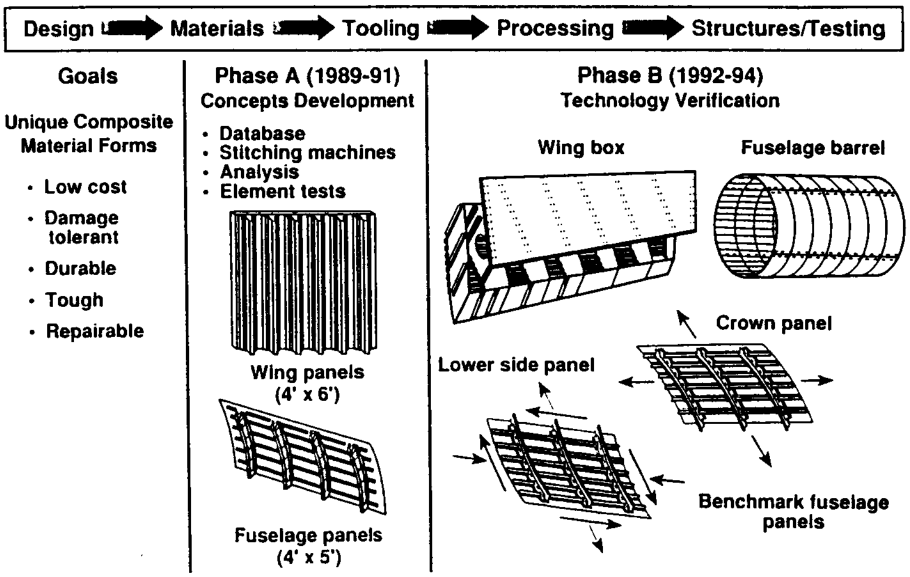
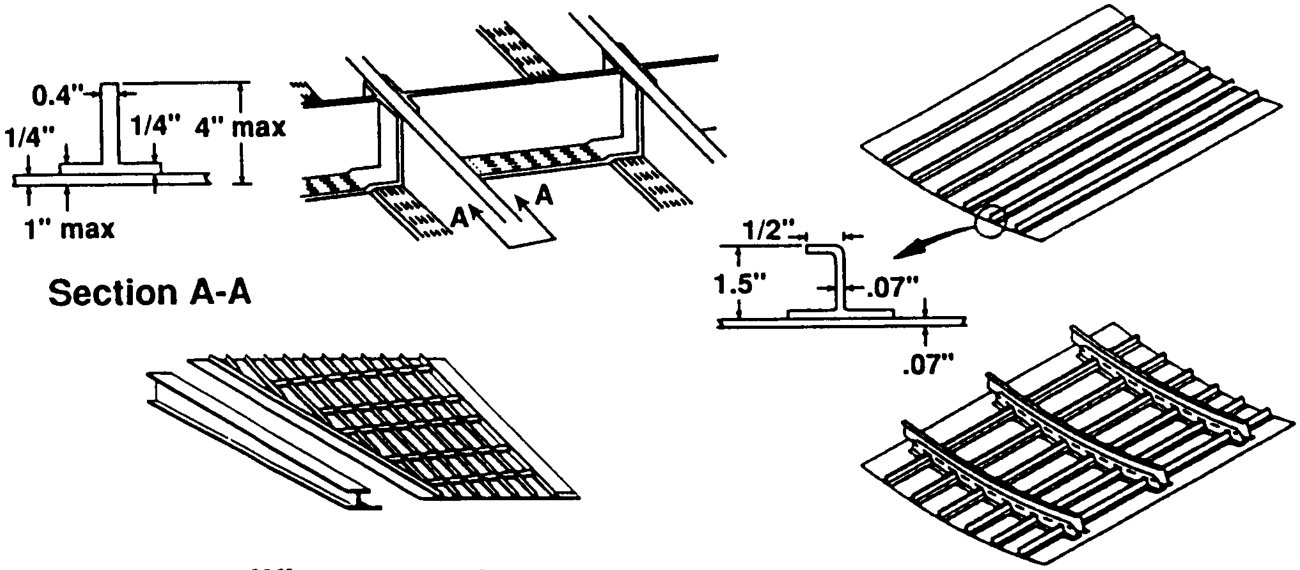


Figure 2

Candidate Concepts for Transport Wing and Fuselage Structures

Design concepts selected by Douglas for stitched/RTM wing and fuselage panels are shown in figure 3. The wing panels incorporate blade stiffeners which were selected for structural efficiency combined with manufacturing simplicity. The design was developed under a previous NASA contract and details are presented in reference 2. In the stitched/RTM wing panels, the skins have a dense array of through-the-thickness stitching and flange-to-skin stitching is used with the stiffeners and intercostals. All elements have the same layup of 44 percent 0° plies, 44 percent +/-45° plies and 12 percent 90° plies.

Like the wing panels, the fuselage design was also developed under a previous NASA contract, see reference 3 for details. Fuselage longerons are "J" sections selected for structural efficiency. In the current stitched/RTM design, only the longeron flanges are stitched to the skin. The layup for skin and longerons has equal percentages, or 33.3 percent each of 0°, +/-45° and 90° plies.



Wing concept

- Blade-stiffened covers
- Stitched skins, blades and intercostals

Fuselage concept

- "J"-stiffened skin
- Stitched longerons

Figure 3

Organizations involved in the NASA/Douglas ICAPS effort are shown in figure 4. Douglas has the lead role in the design, fabrication and stress analysis of stitched/RTM structures, as well as the tabulation of cost data and program documentation. The Materials Division at NASA Langley has a major role in database testing. In addition, Langley has a sizable program to advance the mechanics and technology of stitched composites. The elements of this program are described later in this paper. Under a subcontract to Douglas, researchers at William and Mary College and Virginia Polytechnic Institute are developing RTM process and flow models and processing guidelines for various resin systems, in addition to designing cure monitor instrumentation. Ketema, Inc., another Douglas subcontractor, stitched the fabric preforms for the database test coupons. Pathe, Inc., a third Douglas subcontractor, is designing and building new automated single needle and multi-needle stitching machines.

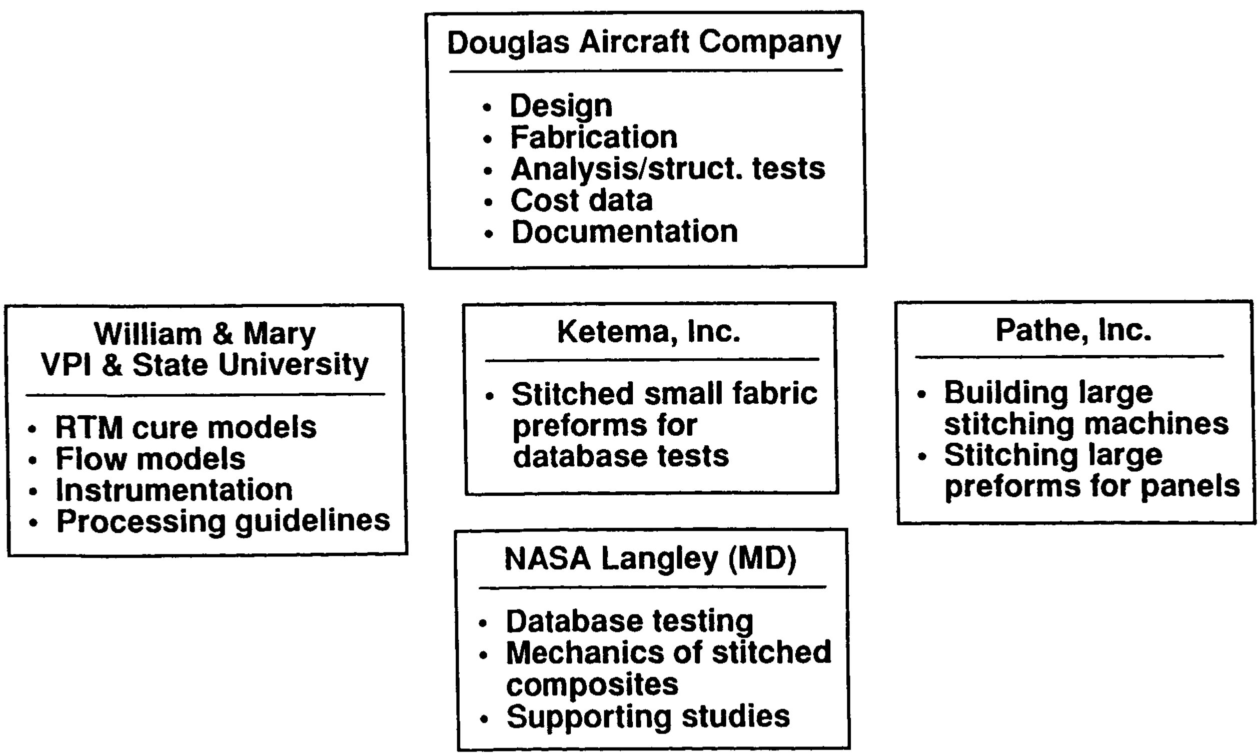


Figure 4

Stitched Materials and Structures Database
Work in Progress

Figure 5 shows work in progress at Douglas and NASA Langley under the ACT contract. Douglas work includes fabrication of stitched and resin transfer molded coupons for database testing, as well as wing and fuselage elements and subcomponents. Douglas is also performing compression tests of thick wing panels and single stringer crippling tests. For structural analysis of stitched composites, Douglas has adopted a modified laminate theory and a macro-mechanics/semi-empirical approach. Douglas is using NASTRAN to model the behavior of "J" stiffened fuselage panels. Douglas is also responsible for checkout of the single-needle and multi-needle stitching machines being developed at Pathe.

Laminate coupon testing is being done at NASA Langley and the data is provided to Douglas for their structural analysis. Tests include tension, compression and compression after impact as well as stiffener pull-off tests and compression testing of fuselage "J" stiffened panels. Tests in progress include stitched stiffener pull-off specimens and stitched "J" stiffened panels.

- **Under the ACT Contract:**
  - **-Fabrication of stitched coupons, elements and subcomponents**
  - **-Compression testing of thick wing panels**
  - **-Single stringer crippling tests**
  - **-Structural analysis**
  - **-Checkout of new stitching machines**
- **At Langley:**
  - **-Lamina coupon testing for analysis data**
  - **-Strength tests of wing and fuselage coupons (tension, compression, CAI)**
  - **-Stiffener pull-off tests**
  - **-Compression testing of fuselage element panels (21 in. x 15 in.)**

Figure 5

Six Stringer and Three Stringer Wing Panels

Douglas has built several three-stringer structural element panels, figure 6(a), using the resin film infusion process. These element panels will be tested in compression to investigate damage tolerance and to provide data for correlation with structural analysis. The test panels are 21 in. wide by 15 in. long.

The first six stringer wing panel successfully built by Douglas using the resin film infusion process is shown in figure 6(b). The skin has 54 plies with ply orientation of $[0°/45°/0°/-45°/90°/-45°/0°/45°/0°]_{3S}$, and the stringers are 72 ply laminates with the same layup as the skin. The panels were resin transfer molded using 3501-6 epoxy resin. The lightly shaded areas visible in Figure 6(b) have been sanded and cleaned for secondarily bonding the intercostals to the wing skin and stringers. In all future panels, the intercostal preforms will be stitched to the skin and the entire assembly will be resin transfer molded.

Figure 6(a)

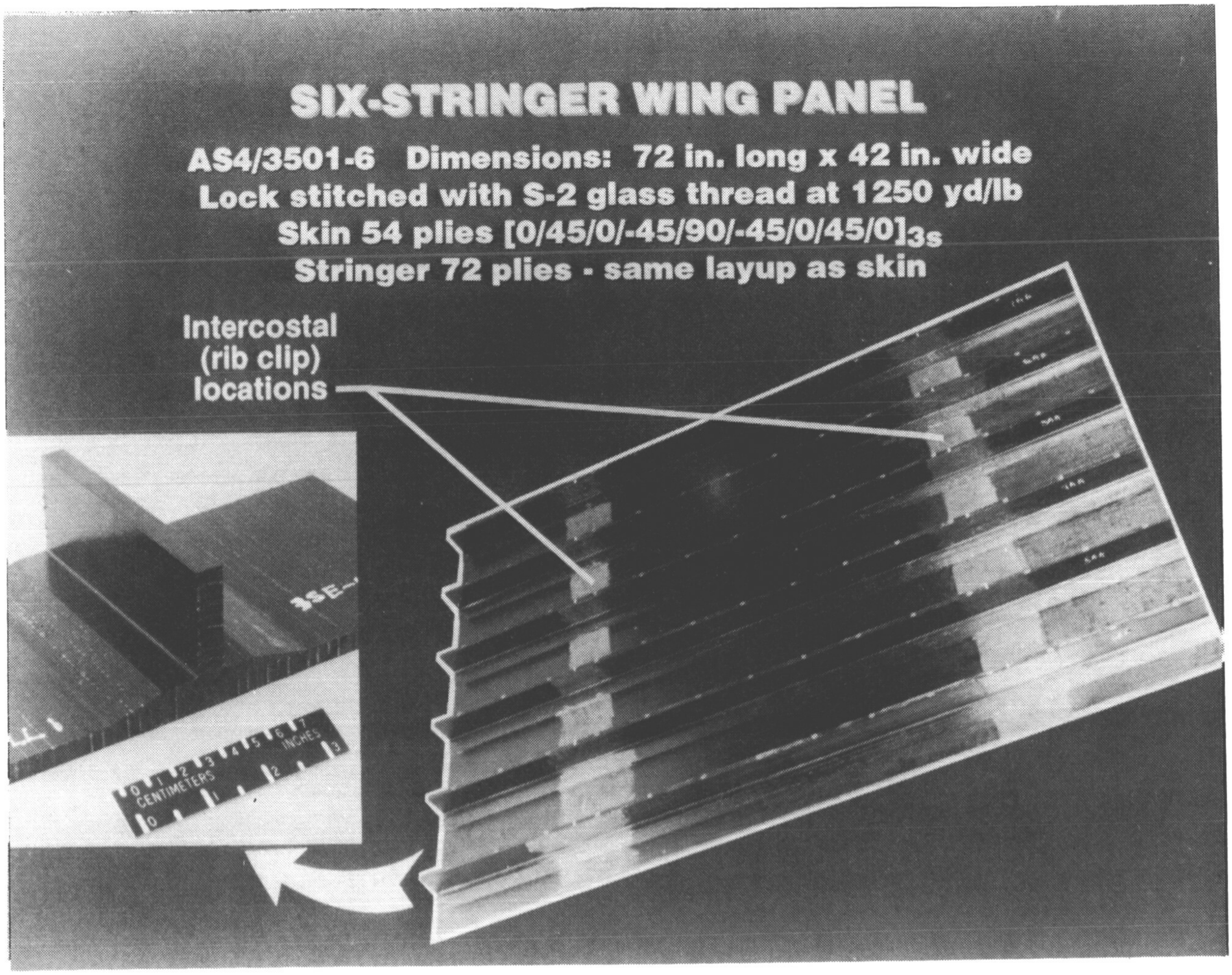

SIX-STRINGER WING PANEL
AS4/3501-6 Dimensions: 72 in. long x 42 in. wide
Lock stitched with S-2 glass thread at 1250 yd/lb
Skin 54 plies [0/45/0/-45/90/-45/0/45/0]3s
Stringer 72 plies - same layup as skin
Intercostal
(rib clip)
locations
CENTIMETERS
INCHES

Figure 6(b)

Figure 7(a) shows the fabrication procedure for making preforms for wing panels. The 54-ply skin is made by stitching together six 9-ply subelements in the desired orientation of 0°, +/-45° and 90° plies. The stiffener is made by stitching together eight 9-ply subelements to form the web section. The flanges are formed by folding out 4 subelements on each side and cutting them at varied lengths to provide taper. A filler of prepreg tape is placed in the flange to web joint and the flanges are then stitched to the skin. A completed AS4 fabric single stringer preform is shown in figure 7(b), ready for resin impregnation and cure.

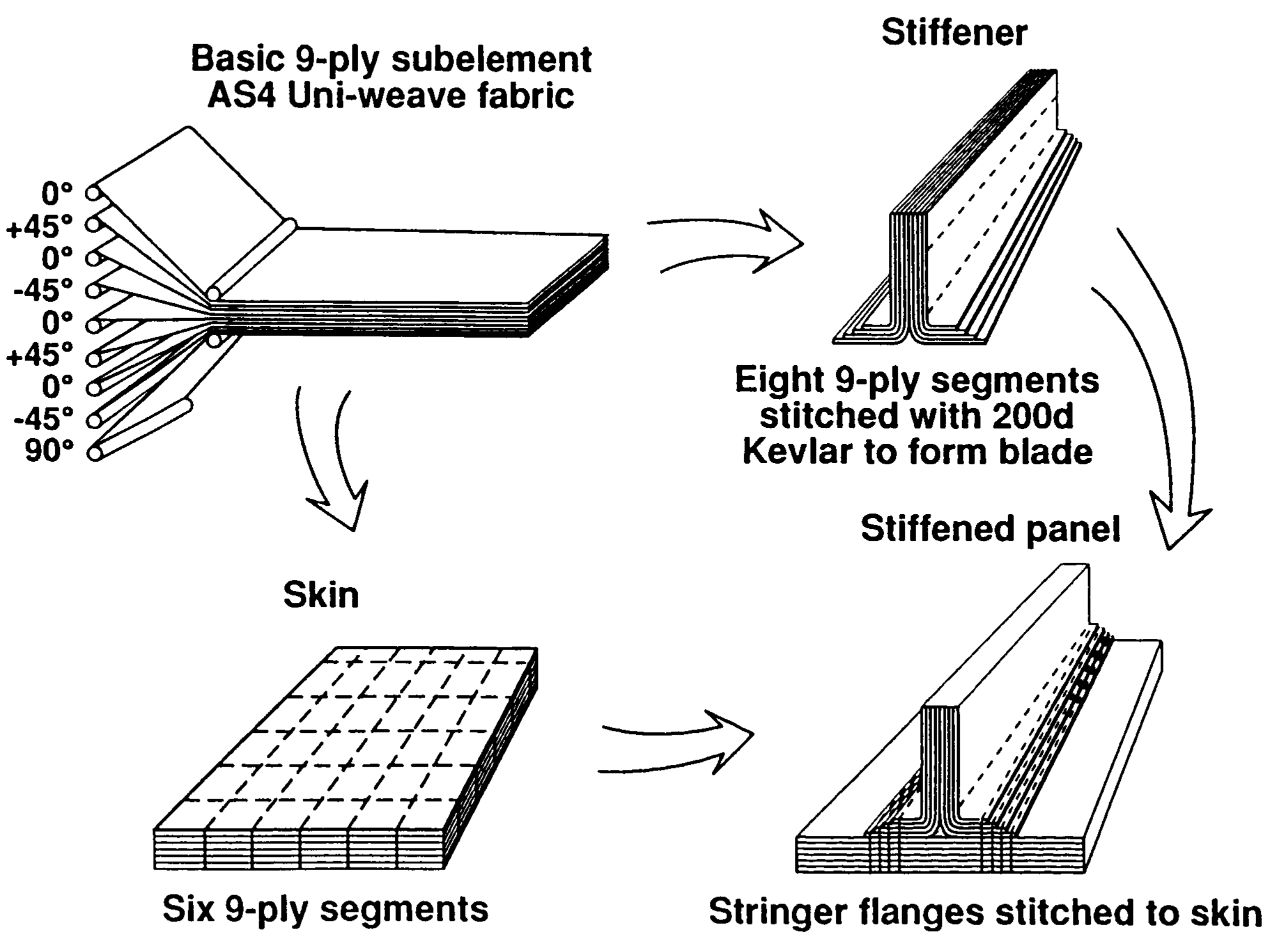


Figure 7(a)

DRY STITCHED CARBON FABRIC PREFORM

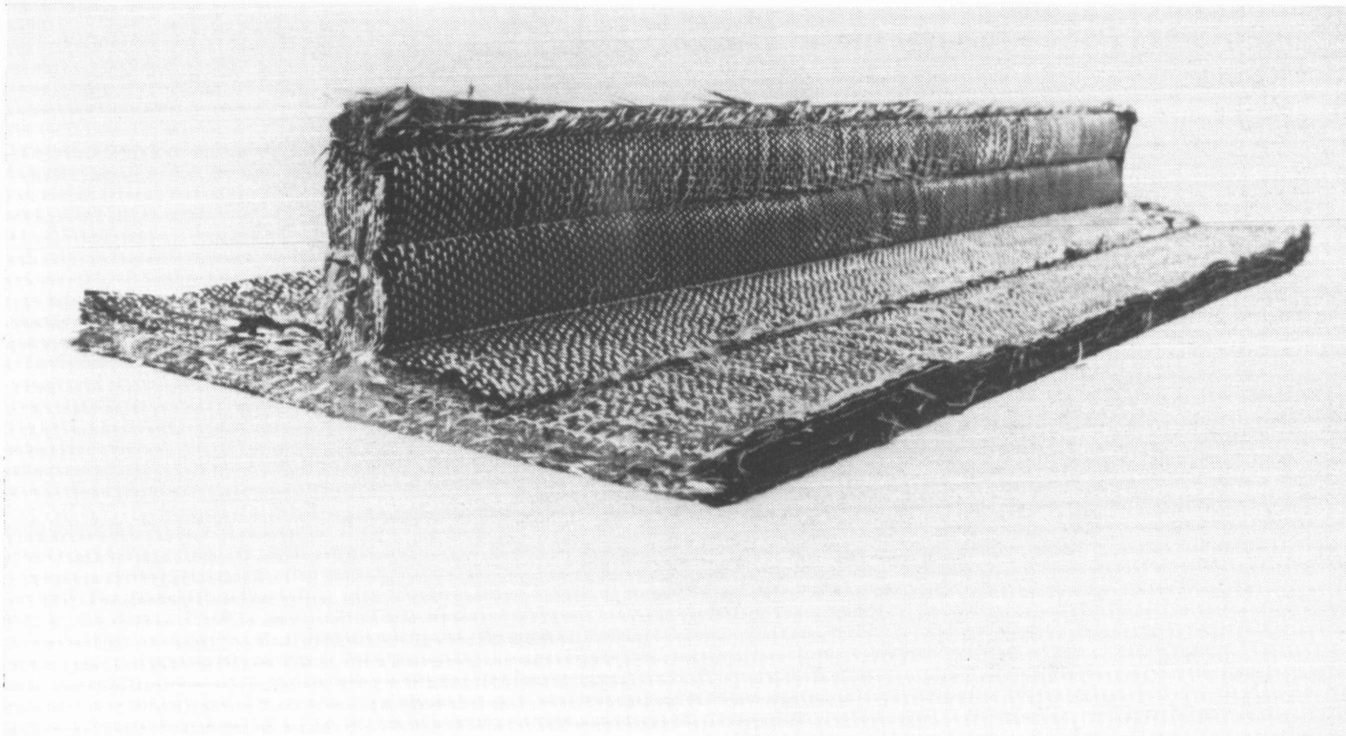


Figure 7(b)

Single Needle Lock Stitch Machine

Preforms for all database laminates and the stiffened panels, figure 6, were stitched on the manual single needle lock stitching machine shown in figure 8. The machine features adjustable needle and bobbin thread tensions and variable stitch pitch and stitch speed. An adjustable guide rail reset after each pass controls the row spacing. With an arm reach of five feet, the machine can accommodate a 0.5 inch thick by 5 feet wide dry preform of any length. Because the stitching speed is slow and the operator must move the preform by hand, this machine is not suitable for the economical stitching of large preforms.

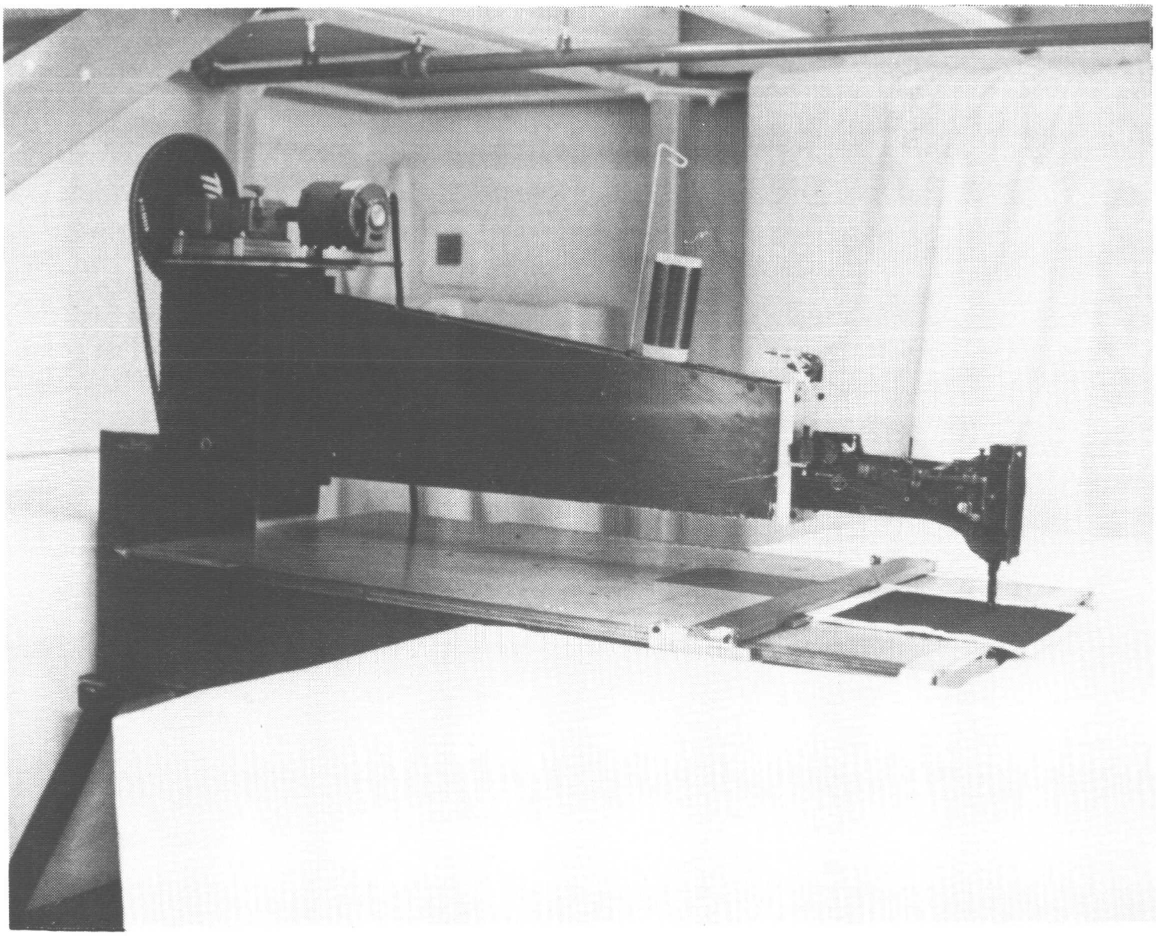


Figure 8

Automated Single Needle and Multi-Needle Stitch Machines

Single needle manual stitching machines have proven to be invaluable tools in the development of damage tolerant stitched composite structures, but more efficient and cost effective methods of stitching preforms must be developed. Figures 9(a) and 9(b) show two stitching machines being developed by Pathe under the Douglas ICAPS contract. The multi-needle stitching machine with up to 256 needles, is mechanically controlled and can accommodate a 128 inch wide preform. In its current design, the multi-needle machine will perform both light and heavy density stitching. The single needle stitching machine features computer controlled motion of the stitching head with a work area of eight feet by fifteen feet. Both machines are limited to lock stitching only. In the overall scheme, the multi-needle machine will stitch together single plies to make a wing skin, for example; then stiffeners will be stitched to the skin using the single needle machine. The current manual machine, figure 8, would require about 400 hours to stitch an eight by twelve foot preform, whereas the new automated multi-needle machine would reduce that time to about one hour (ref. 4).

DRY PREFORM MULTI-NEEDLE STITCHING MACHINE
Cam/Gear Control - 256 Needles in Two Rows
Stitching Width 128 Inches - Lock Stitching

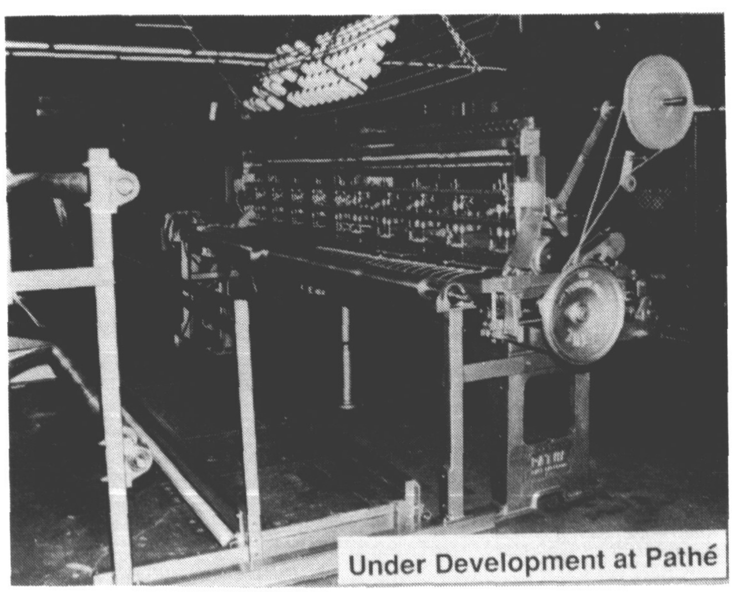


Figure 9(a)

DRY PREFORM SINGLE NEEDLE STITCHING MACHINE

Computer Controlled X-Y Motion of Stitching Head
Working Area 8' x 15' - Lock Stitching

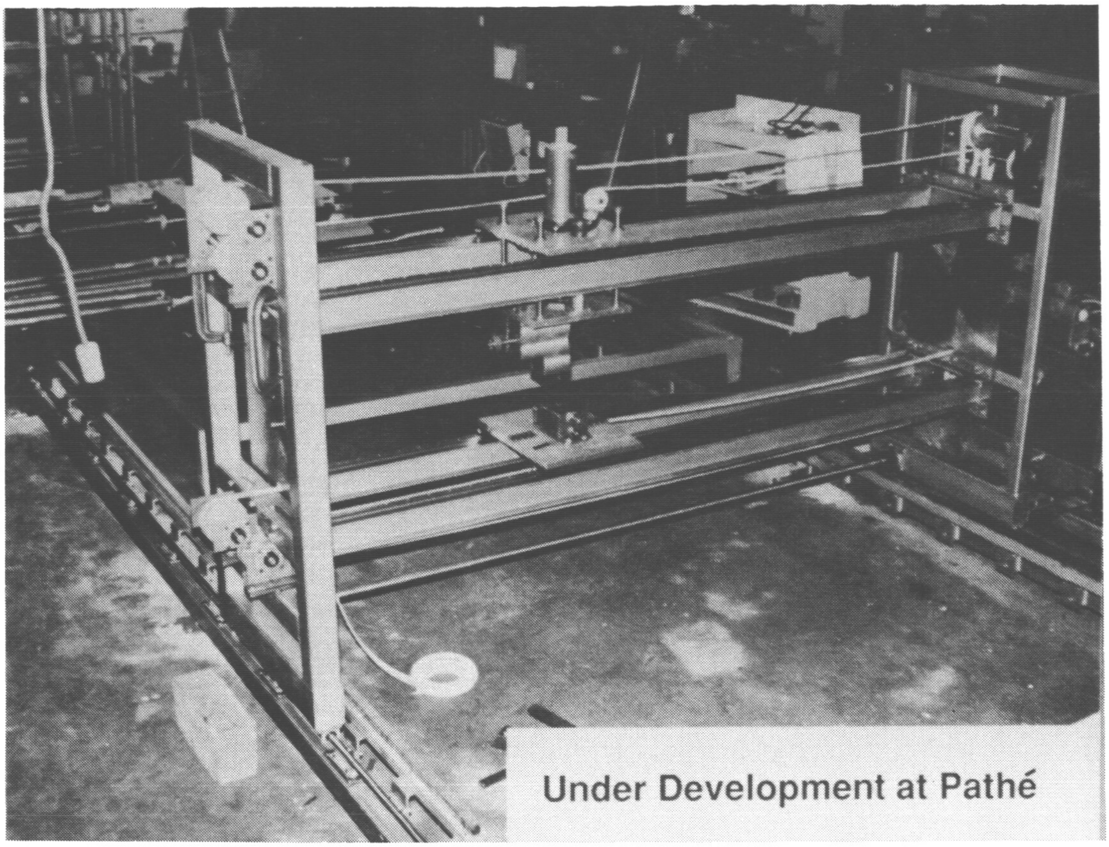


Figure 9(b)

Specimen Fabrication for Database Testing

The overall scheme for fabricating and testing specimens for the stitching/RTM database is shown in figure 10. Laminates were made with AS4 uniweave fabric that contained 97.5 weight percent 0° (warp) carbon fibers and 2.5 percent 90° (fill) glass fibers. The glass fill fibers were used merely to stabilize the 0° carbon fibers and facilitate handling of the fabric. Individual plies of the uniweave fabric were cut and stacked in a $[45°/0°/-45°/90°]_{2S}$ sequence to form a 16-ply, quasi-isotropic laminate and in a $[45°/0°/-45°/90°]_{12S}$ sequence to form a 96-ply laminate. The 16-ply laminates were chosen for testing to simulate fuselage structure and the 96-ply laminates were chosen to represent built-up areas of the wing skins. The dry fabric preform stacks were lock stitched with S-2 fiberglass and Kevlar 29 threads of various weights, then resin transfer molded with 3501-6 resin. The AS4 fabric and 3501-6 epoxy resin were chosen as the baseline materials because they have been well characterized and, compared with other fiber/resin systems, are among the least expensive. The resin evaluation specimens were quasi-isotropic unstitched laminates of uniweave fabric that were resin transfer molded using either Shell 1895 or 862 resins or the British Petroleum E905L resin.

Test specimen configurations are shown in the lower left quadrant of figure 10. The 1.75-inch by 1.5-inch short block compression specimen is a NASA Langley configuration suitable for tests of angle ply laminates. For the tension tests, a tabbed 9-inch by 1-inch specimen was used and the compression after impact tests were performed using the 10-inch by 5-inch specimen shown, as recommended by reference 5. Results of previous laminate property and stitching guideline tests are given in reference 1.

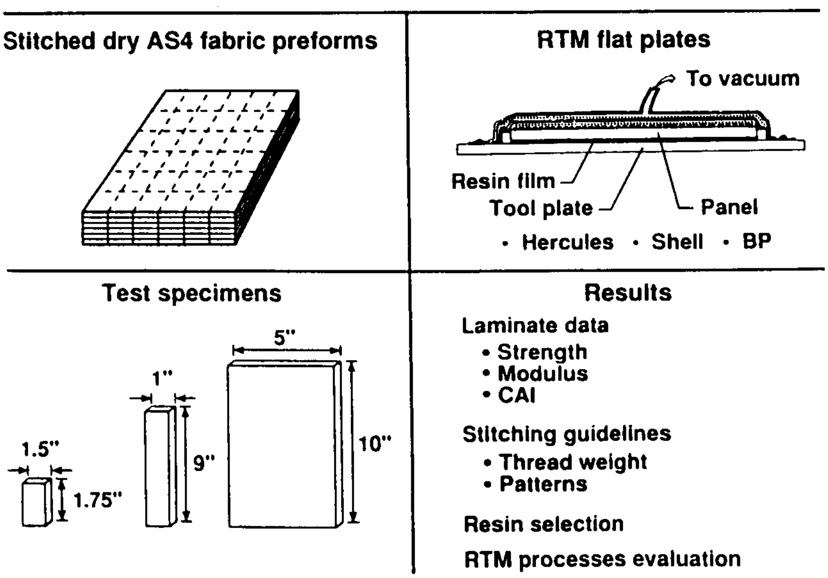


Figure 10

Strength and stiffness data for tension and compression tests of thin (16-ply, 0.096-inch nominal thickness) and thick (96-ply, 0.576-inch nominal thickness) stitched laminates are shown in figure 11. Values shown represent the average of three test specimens. Two different stitching patterns were used: 1/8" and 3/16" row spacing each with eight penetrations per inch and stitch rows parallel to the 0° carbon fibers, (designated 0° stitching). The thick compression specimens were stitched using the same two patterns, but with an additional pattern having stitch rows perpendicular to the 0° carbon fibers (90° stitching). All specimens were stitched with a 200 denier (d) Kevlar 29 needle thread and with bobbin threads as indicated in figure 11. Denier is the weight in grams of a 9000 meter length of thread. In the designation for S-2 glass, the number '449' refers to the epoxy compatible sizing on the fibers, and '1250' refers to the thread weight in yards per pound.

For the thin laminates, the data show that using S-2 glass bobbin thread, which is six times heavier than the 600d (7448 yards/pound) Kevlar bobbin thread, gives much lower in-plane properties. The highest strengths were obtained using the 3/16" x 8 x 0° stitch pattern and Kevlar 29 600d bobbin thread. For the thick laminates, compressive properties of thick laminates with 0° stitching were higher than for 90° stitching. Additionally, better properties were obtained using the 3/16" x 8 x 0° stitch pattern and 3000d (1488 yards/pound) Kevlar bobbin thread. These results are in agreement with data presented in reference 1.

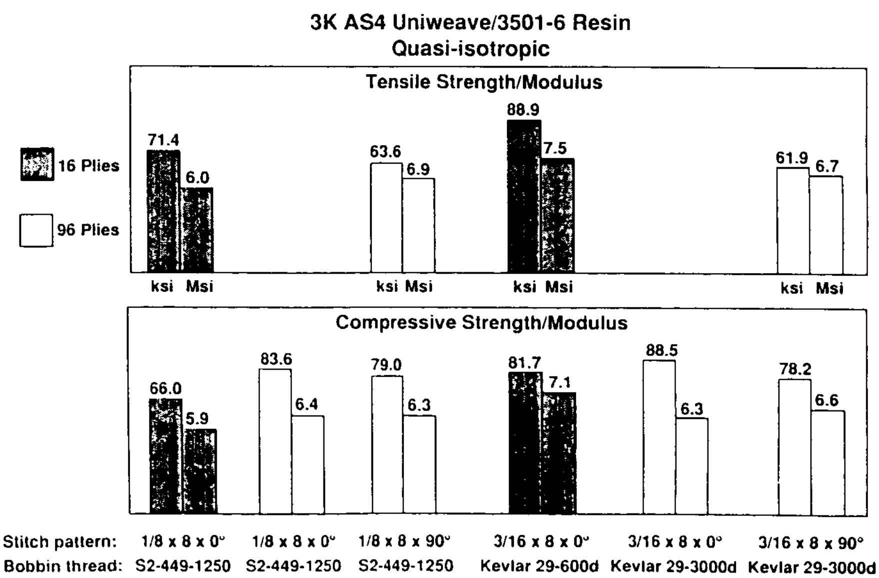


Figure 11

Figure 12 shows the results of compression after impact (CAI) tests on thin (16-ply) and thick (96-ply) stitched laminates. The thin specimens were impacted using a 0.5-inch diameter steel tup attached to a 10-pound drop weight at the energy levels indicated. The thick laminates were impacted using a 1.0-inch diameter steel tup attached to a 20-pound drop weight at the energy levels shown. These energy levels were chosen to assess the damage tolerance of stitched laminates under severe conditions.

The results for the thin laminates show that using the stronger S-2 glass bobbin thread (breaking strength: 59 pounds) and the 1/8" x 8 x 0° stitching gives outstanding CAI strength retention when compared to laminates stitched with 600d Kevlar bobbin thread (breaking strength: 24 pounds) and 3/16" x 8 x 0° stitching. However, as shown in figure 11, lower in-plane properties were obtained using S-2 glass and 1/8" x 8 x 0° stitching. Based on results presented here and in reference 1, the best compromise for stitching thin laminates would be the 3/16" x 8 x 0° stitch pattern and 600d Kevlar thread.

C-scans (not shown) of the thick panel with 1/8" x 8 x 90° stitching indicated the presence of manufacturing defects prior to being impacted at 100 ft-lbs, and would explain the lower CAI strength than those impacted at higher energies. There were no indications of manufacturing defects in the other thick panels. The results for the thick laminates indicate that there is no real advantage in using either one of the thread/stitch pattern combinations tested. When compared to the results presented in figure 11, however, the best combination of in-plane properties and CAI strength retention was obtained using 3000d Kevlar thread (breaking strength: 124 pounds) and the 1/8" x 8 x 90° stitch pattern. These results are in agreement with those presented in reference 1.

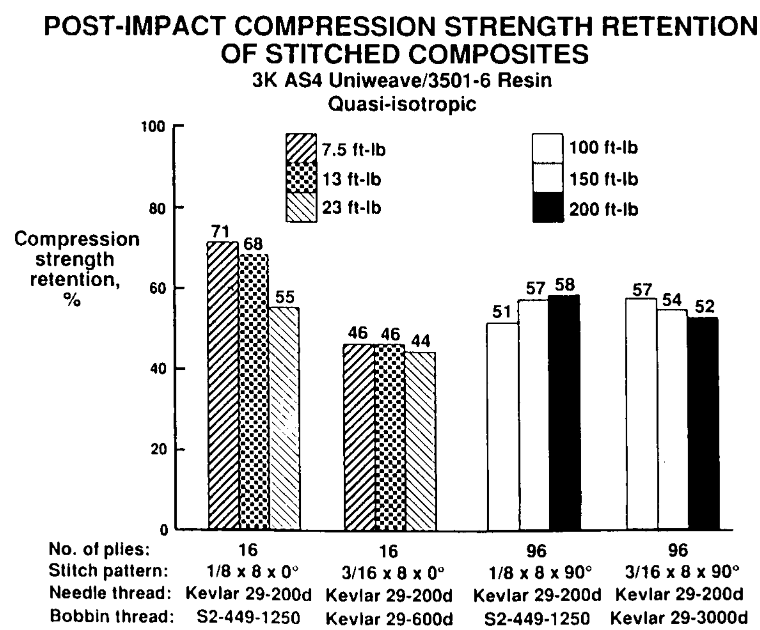


Figure 12

RTM Resin Evaluation and Selection for Pressure RTM

The purpose of this investigation was to select a low viscosity resin for resin transfer molding fuselage panels. The 3501-6 resin used in wing structure was not considered for the pressure RTM process because of its short pot life at elevated temperature. Selection was focused on two-component resins formulated specifically for resin transfer molding. Figure 13 shows the results of tension and compression tests on quasi-isotropic unstitched laminates. The 8-ply test specimens were fabricated from AS4 uniweave fabric with ply orientation of $[45°/0°/-45°/90°]_S$, then resin transfer molded using the three resins shown. Both Shell resins, 1895 and 862, showed comparable tension properties, but the 1895 resin had the best room temperature, dry (RTD) and hot, wet compression properties of the three resins tested. The 1895 and 862 laminates, each with an average thickness of 0.051 inch, were thinner than the E905L laminate, which was 0.059 inches thick. The difference in thickness might indicate that the E905L laminate had a lower fiber volume fraction and thus explain its lower properties. Both Shell resins, at five and ten dollars per pound, are more economical when compared with 3501-6 resin ($36 per pound) or toughened resins ($100 or more per pound). The best combination of performance and cost was provided by the Shell 1895 resin, which was selected by Douglas for pressure RTM of the fuselage elements. Ongoing RTM resin evaluation tests at Langley will further verify these results.

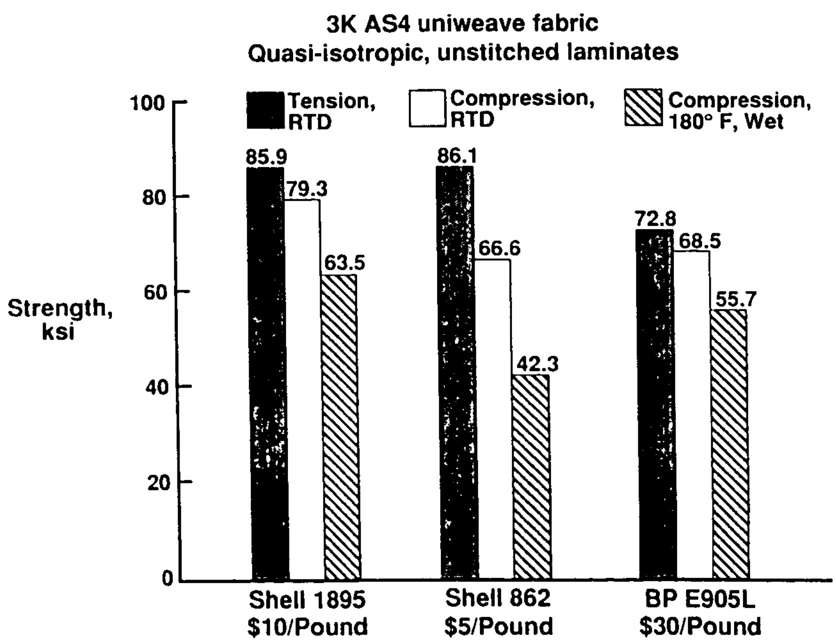


Figure 13

Expanding the Technology for Stitched/RTM Structures

NASA Langley Research Center's in-house program to expand the science-based technology for stitched/RTM composite structures is outlined in figure 14. Within the Materials Division, the Mechanics of Materials Branch will be involved with Douglas in cooperative research to model the mechanics of stitched laminates. The Applied Materials Branch will investigate the effects of stitching parameters on structural performance, new stitched materials concepts, and environmental effects on stitched laminates. Each element of the program will be discussed further in succeeding figures.

- **Mechanics modeling of stitched composites**
- **Effects of stitching parameters on structural performance**
- **Effects of moisture and thermal cycles**
- **New stitched material concepts**

Figure 14

Figure 15 shows the areas to be investigated in a NASA/Douglas cooperative research program on mechanics of stitched composites. Researchers will study failure modes such as Euler buckling, "micro" buckling and sublaminate buckling, as well as bolted joint failure modes such as net tension and bearing. Interlaminar toughness testing will also be included, along with the effects of ply drops on fatigue properties.

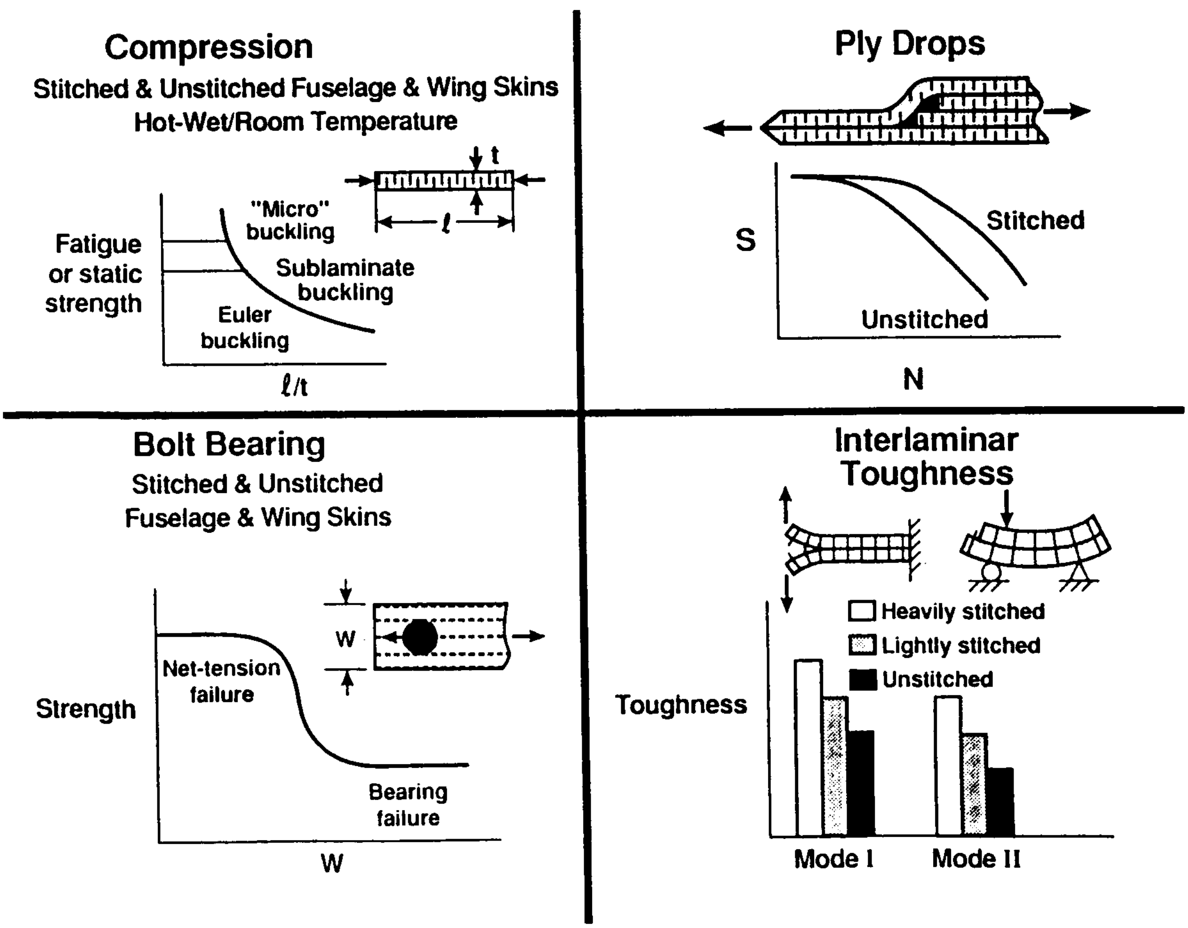


Figure 15

Stitched Composites Parametric Investigation

A test program shown in figure 16 will investigate five different stitched composite parameters: laminate thickness (number of plies), stitch pitch, row spacing, thread material and thread strength. Tension, compression and compression after impact tests will be performed. A design of experiments (Taguchi) approach will be employed to provide significant information with a minimum number of tests. The resulting parametric database will be used to develop predictive models and stitching guidelines. The guidelines will include the laminate thickness/stitch parameter interactions and the trade-offs between in-plane strength loss and improved damage tolerance. British Petroleum E905L resin was chosen for this study before the RTM resin evaluation results presented in figure 13 were available. Additional resins may be included in this work as more resin evaluation tests are completed.

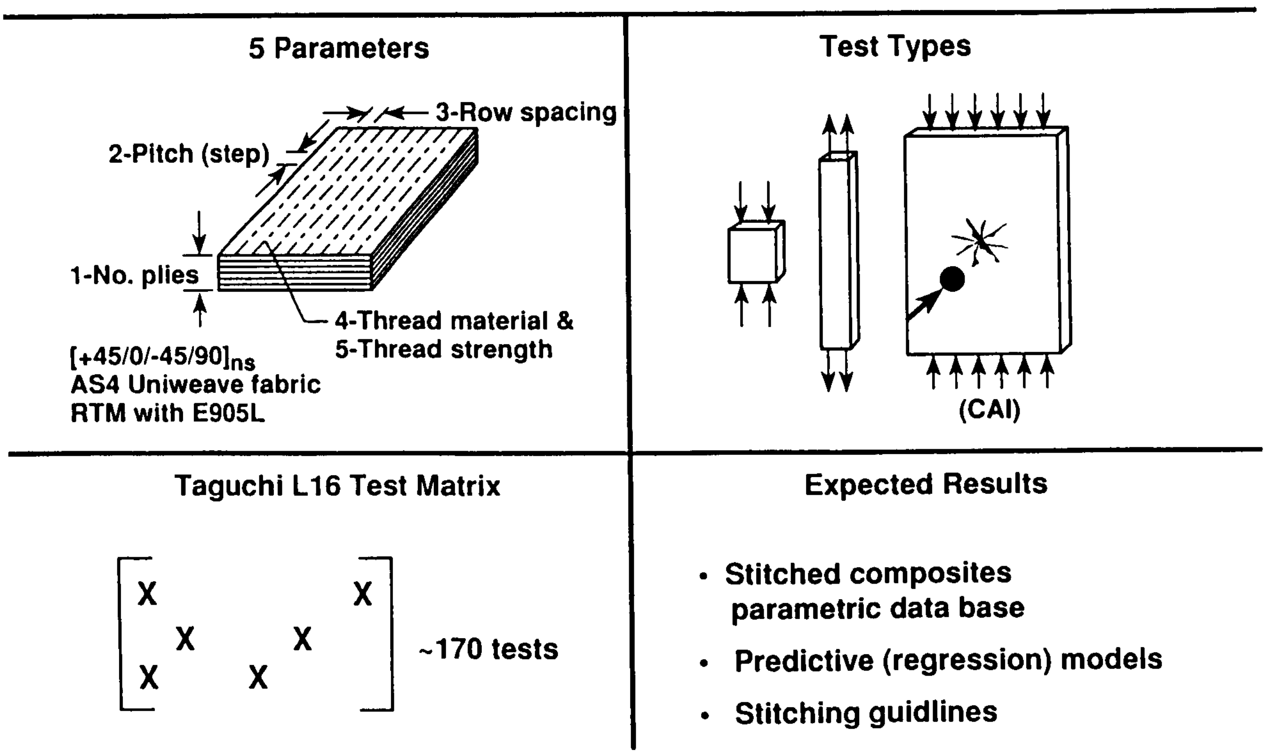


Figure 16

Environmental Effects on Stitched Composites

Figure 17 illustrates another Langley research program that is currently investigating environmental effects on stitched composites. Test panels are 32-ply quasi-isotropic laminates of AS4 uniweave fabric, with ply orientation of $[45°/0°/-45°/90°]_{4S}$. Three groups of specimens will be tested: unstitched laminates, and laminates stitched with 1500 yd/lb S-2 glass thread or 1000d Kevlar 29 thread. All test panels have been resin transfer molded with 3501-6 resin, cut into test coupons as shown, and are being subjected to an environmental cycling regime of +60° C to -54° C and 0 to 100 percent relative humidity. The 3501-6 resin was chosen because it has been well-characterized, and it is the resin selected for thick, heavily loaded wing panels of the type shown in figure 6. The expected results of this study include diffusion coefficients that will better define the moisture absorption of stitched composites, a greater understanding of microcracking mechanisms, especially around the stitch threads, and residual strength properties as a function of environmental history. A companion study that investigates the effects of jet fuels, hydraulic fluids, and other chemicals on RTM resins is nearly completed.

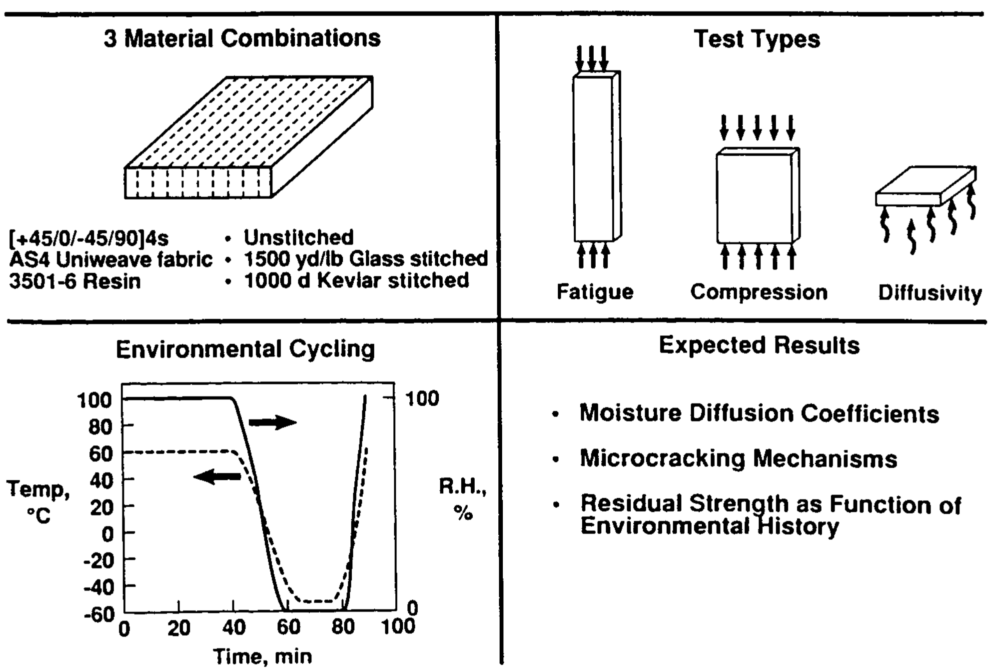


Figure 17

Improved Damage Tolerance of Composites Using Glass Buffer Strips

Figure 18(a) illustrates a third Langley research program aimed at utilizing existing materials combined with stitching and resin transfer molding to create innovative damage tolerant materials. This research will use glass buffer strip fabric made of AS4 uniweave, with half-inch strips of the 0° carbon (warp) fibers replaced with S-2 glass as shown in Figure 18(b). The glass "softening" buffer strips, less stiff than the surrounding carbon fibers, have been shown to effectively arrest crack growth in composites (ref. 6), but their compression properties have not been adequately characterized. The 40-ply quasi-isotropic laminates will be laid up with glass buffer strips in every layer and ply orientation of $[45°/0°/-45°/90°]_{5S}$. The panels will then be stitched, resin transfer molded and cut into test specimens as shown in Figure 18(a). British Petroleum E905L resin was chosen for this study before the RTM resin evaluation results presented in figure 13 were available. Additional resins may be included in this work as more resin evaluation tests are completed. Tension, short block compression, open hole compression, compression after impact and bearing test results are expected to demonstrate the best combination of buffer strip orientation and stitching for improved damage tolerance and bearing strength.

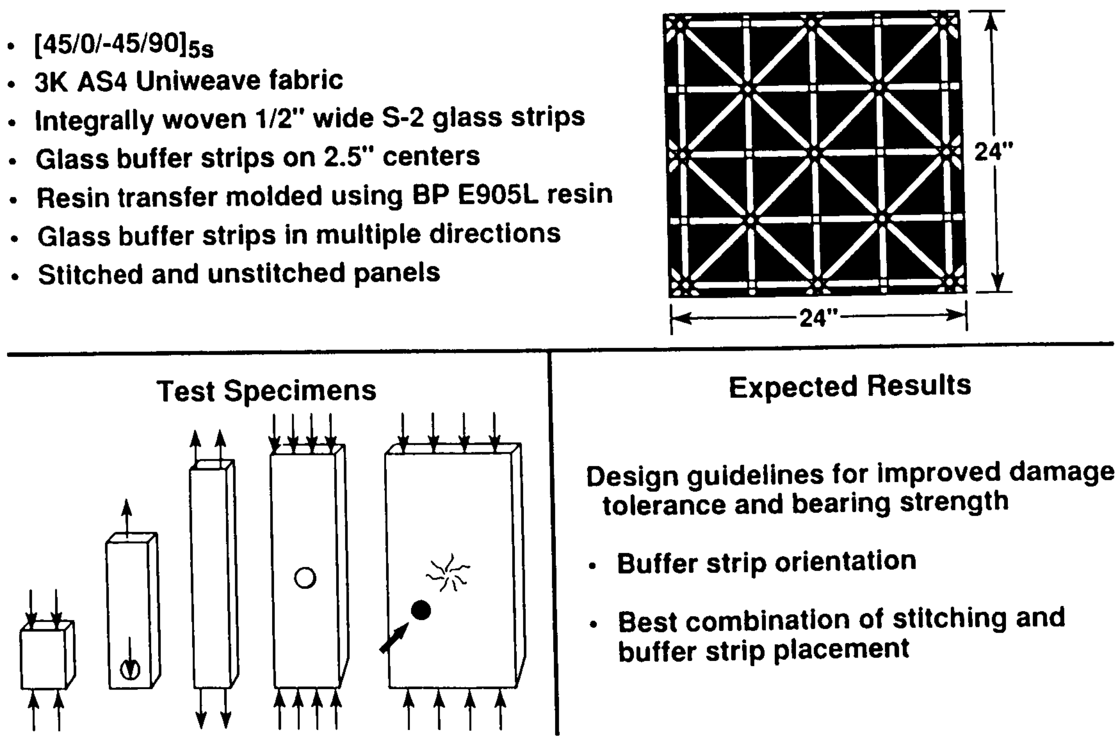


Figure 18(a)

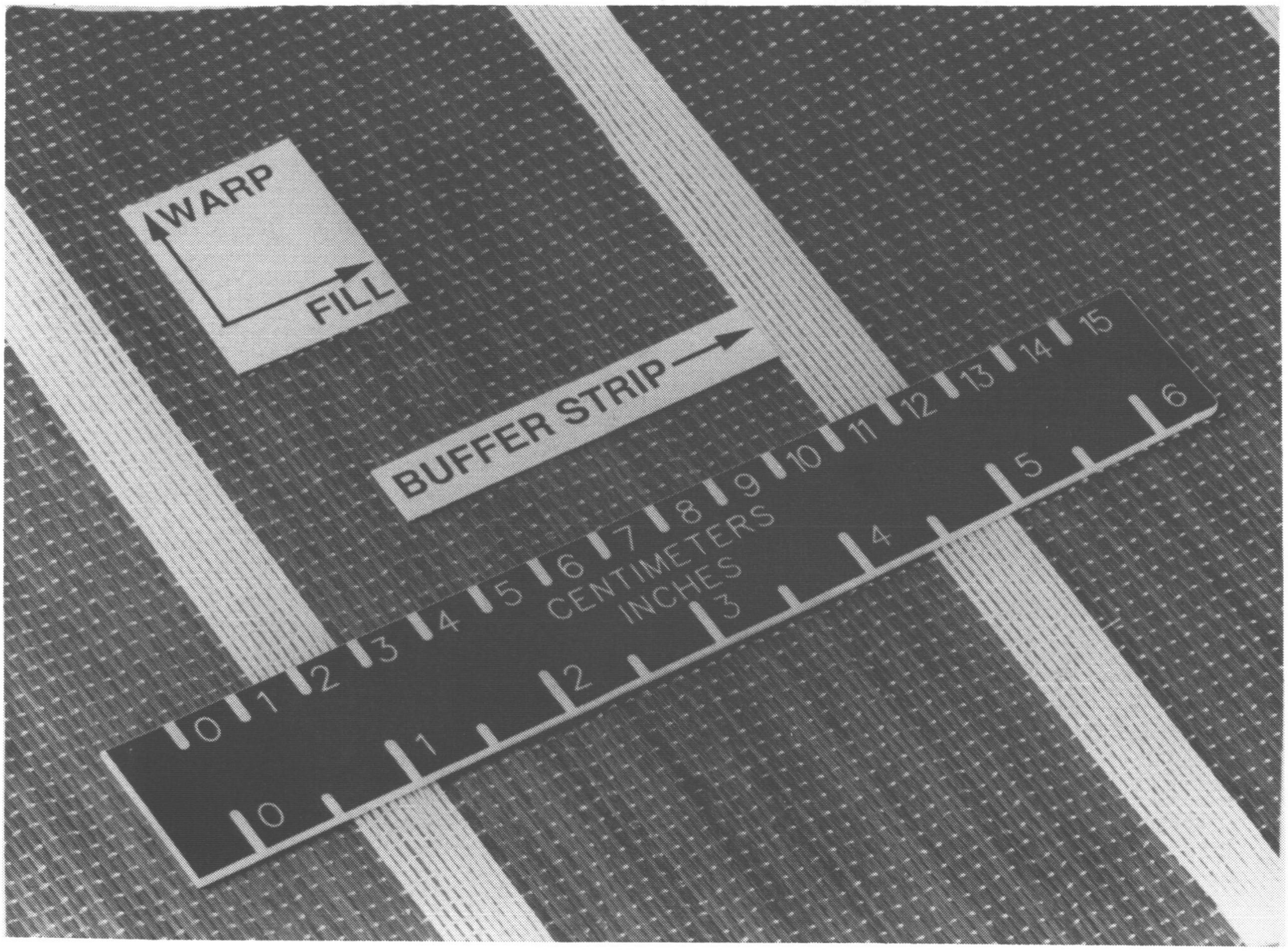

WARP
FILL
BUFFER STRIP
CENTIMETERS
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Figure 18(b)

Concluding Remarks
The Case for Stitching

The research to date on stitched/RTM composites supports the conclusions listed in figure 19. Stitched composites show outstanding damage tolerance, as indicated by their post-impact compression strength retention. Stitched composites also demonstrate acceptable fatigue behavior and hot, wet performance as reported in reference 1. Stitching and resin transfer molding provide near net shape molding of integral structures requiring very little machining to final size and reduce the need for mechanical fasteners. Lower cost fibers and resins can be used in stitched and resin transfer molded structures, making them more cost-effective than toughened resin composites and traditional prepreg tape composites. In summary, stitched and resin transfer molded composites afford strong potential to achieve the benefits of weight savings and performance offered by composite primary aircraft structures.

- **Completed tests on stitched laminates showed:**
  - **-Outstanding damage tolerance**
  - **-Acceptable fatigue behavior**
  - **-Acceptable hot, wet performance**
- **Provides near net shape molding of integral structures**
- **Accommodates lower cost fibers and resins**
- **Reduces need for mechanical fasteners**
- **Potential breakthrough technology for composite primary structures**

Figure 19

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