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TECHNO-ECONOMIC REQUIREMENTS FOR COMPOSITE AIRCRAFT COMPONENTS

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This paper presents information obtained in the performance of NASA contract NAS1-18862, "Innovative Composite Aircraft Primary Structure" (ICAPS).

INTRODUCTION

- WHY COMPOSITES
- AIRCRAFT APPLICATIONS
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- NEAR NET SHAPE DRY FIBER PREFORM
- RESIN INFUSION
- DAMAGE TOLERANCE
- COSTING ESTIMATES

Why Composites?

The primary reason for use of composites is to save structural weight. A well designed composite aircraft structure will usually save 25-30 percent of a well designed metal structure. The weight savings then translates into improved performance of the aircraft in measures of greater payload, increased flying range or improved efficiency - less use of fuel.

Composite materials offer technical advantages. Key technical advantages that composites offer are high stiffness, tailored strength capability, fatigue resistance, and corrosion resistance. Low thermal expansion properties produce dimensionally stable structures over a wide range of temperature. Specialty resin "char" forming characteristics in a fire environment offer potential fire barrier application and safer aircraft.

The materials and processes of composite fabrication offer the potential for lower cost structures in the near future.

- **SAVE STRUCTURAL WEIGHT**
- **HIGH STIFFNESS**
- **IMPROVED STRUCTURAL PERFORMANCE**
- **TAILORED STRENGTH**
- **FATIGUE RESISTANCE**
- **CORROSION RESISTANCE**
- **LOW THERMAL EXPANSION**
- **RETAINED STRENGTH IN FIRE**
- **COMPLEX SHAPE FABRICATION**
- **POTENTIAL FOR LOWER COST**
- **UNLIMITED BUSINESS OPPORTUNITIES**

AIRCRAFT

COMPOSITE

APPLICATIONS

The first military aircraft production composite part was the stabilizer on the Grumman A-4 in 1970. This part, not shown on the view graph, represented only about 1 percent of the structural weight of the aircraft.

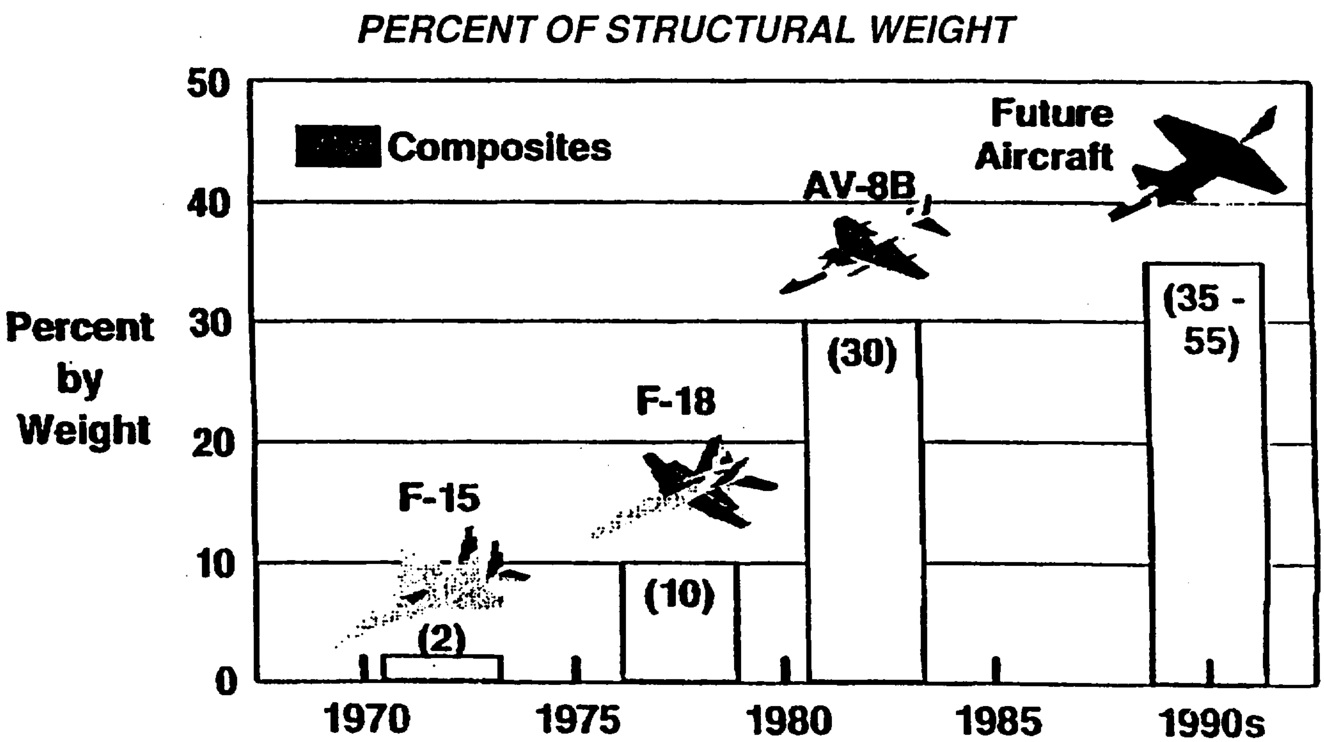
In 1972, McAIR introduced the F-15 applications of speed brake, horizontal and vertical stabilizer skins as composite materials that represented about 3 percent of the aircraft structural weight.

In 1978 McAIR introduced additional composite applications on the F-18. Horizontal and vertical stabilizer skins, wing skins, and control surfaces that represented almost 11 percent of the aircraft structural weight.

In 1982 McAIR added more composite materials than ever before to the AV8-B fighter aircraft. Wing skins and sub-structure, horizontal stabilizer skins and sub-structure, nose fuselage structure, control surfaces made of composite material represented almost 30 percent of the total primary structure. The vertical stabilizer reverted to metal structure to allow its use as an antennae.

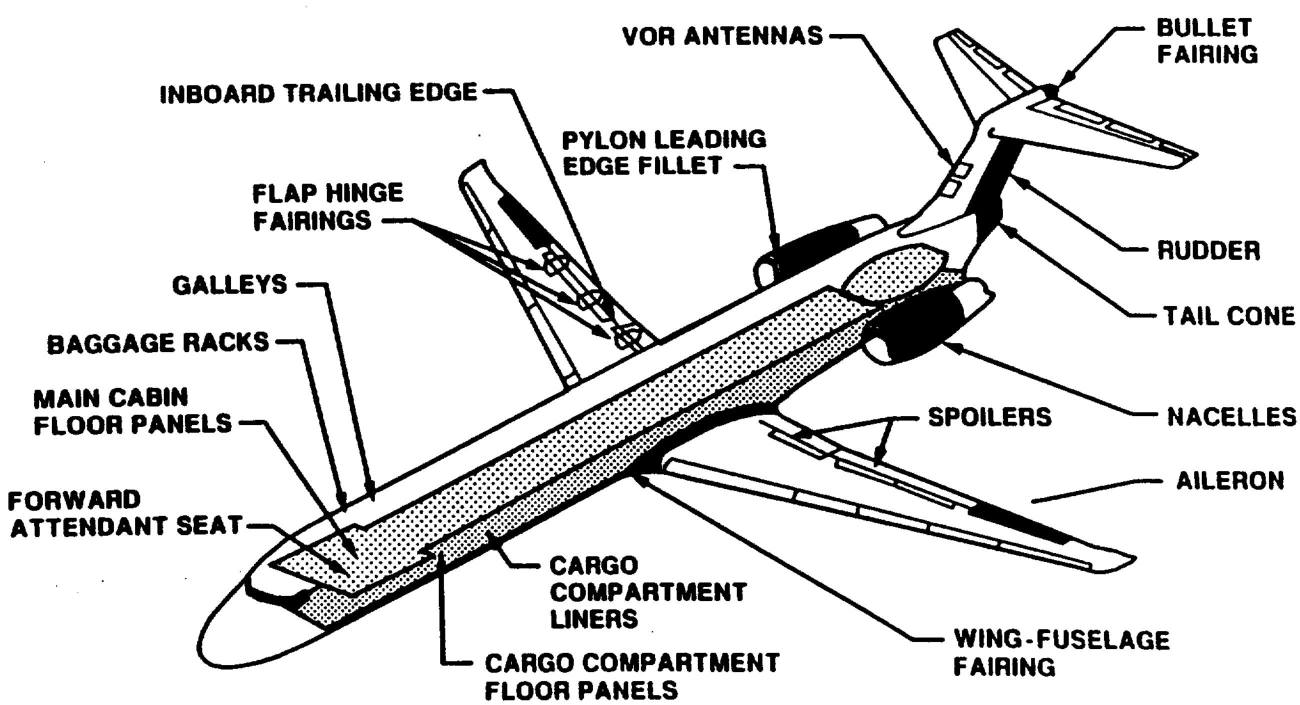
It is predicted that the next generation fighter aircraft will have over 50 percent composite material structural weight.

McAIR COMPOSITES - EXPERIENCING GROWTH



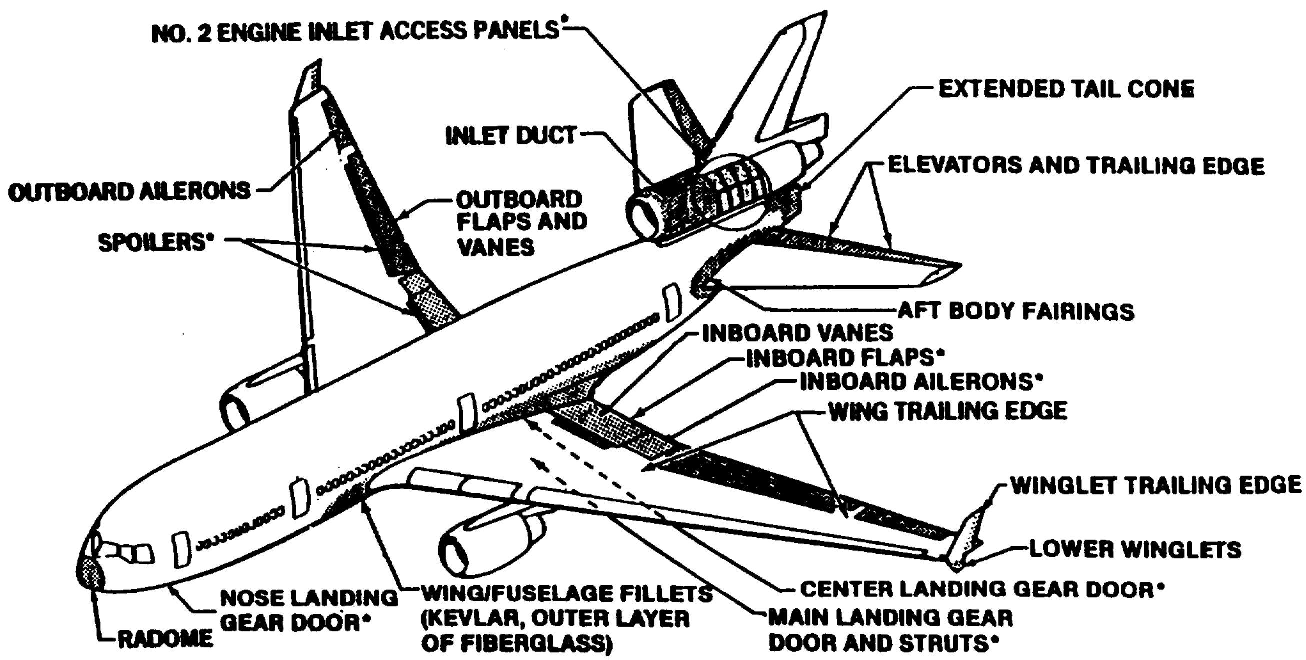
The original MD-80 series aircraft was an all metal primary structure design. Once an aircraft is in production, it is very difficult to change to a more expensive, even though lower weight component made from composite materials. Many of the *feathers* of the aircraft, representing 3 percent of the overall structural weight, were eventually converted to composite materials that included spoilers, ailerons, rudder, engine nacelles, wing trailing edges, and tail cone. No primary structure is of composite materials.

MD-80 ADVANCED COMPOSITES



Composite secondary structural components were designed into the original MD-11 production design that represents approximately 5 percent of the overall structural weight of the aircraft. Most control surfaces such as outboard ailerons, flaps and vanes, and spoilers and wing trailing edge panels are carbon fiber composites. Horizontal stabilizer elevators and trailing edge panels are composites. Winglet skins are carbon fiber composites, Wing fuselage fairings and aft body fairings are Kevlar fiber composites. All considered, composites are still considered as "feathers" of the aircraft and there is no true primary structure.

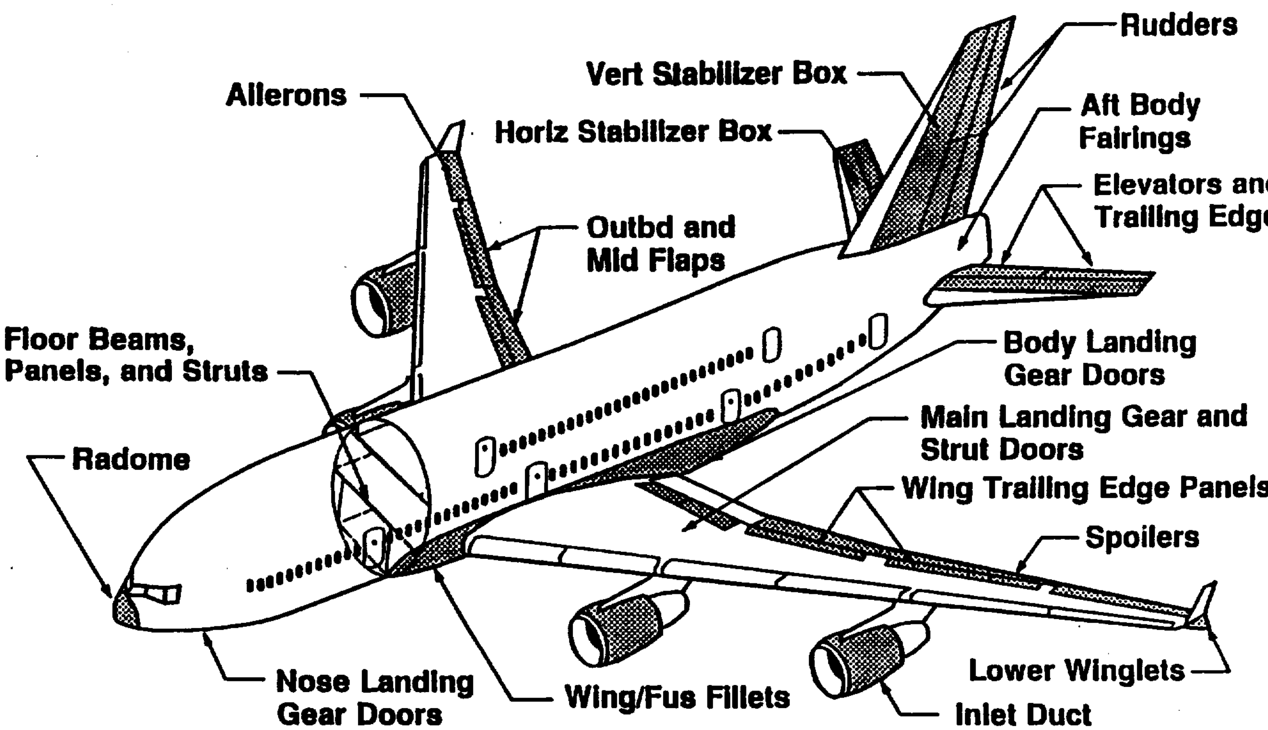
MD-11 COMPOSITE CONSTRUCTION



*UNDER CONSIDERATION

The final design of the MD-12X has not yet been completed. However, almost 11 percent overall composite structural weight is being proposed for this aircraft. All control surfaces are proposed to be carbon fiber composite. Fairings and various "feathers" are also proposed in a manner similar to the MD-11. In addition, primary structure (carbon fiber) is proposed for horizontal and vertical stabilizers. Internal fuselage cargo and passenger floor beams and support struts are proposed carbon fiber.

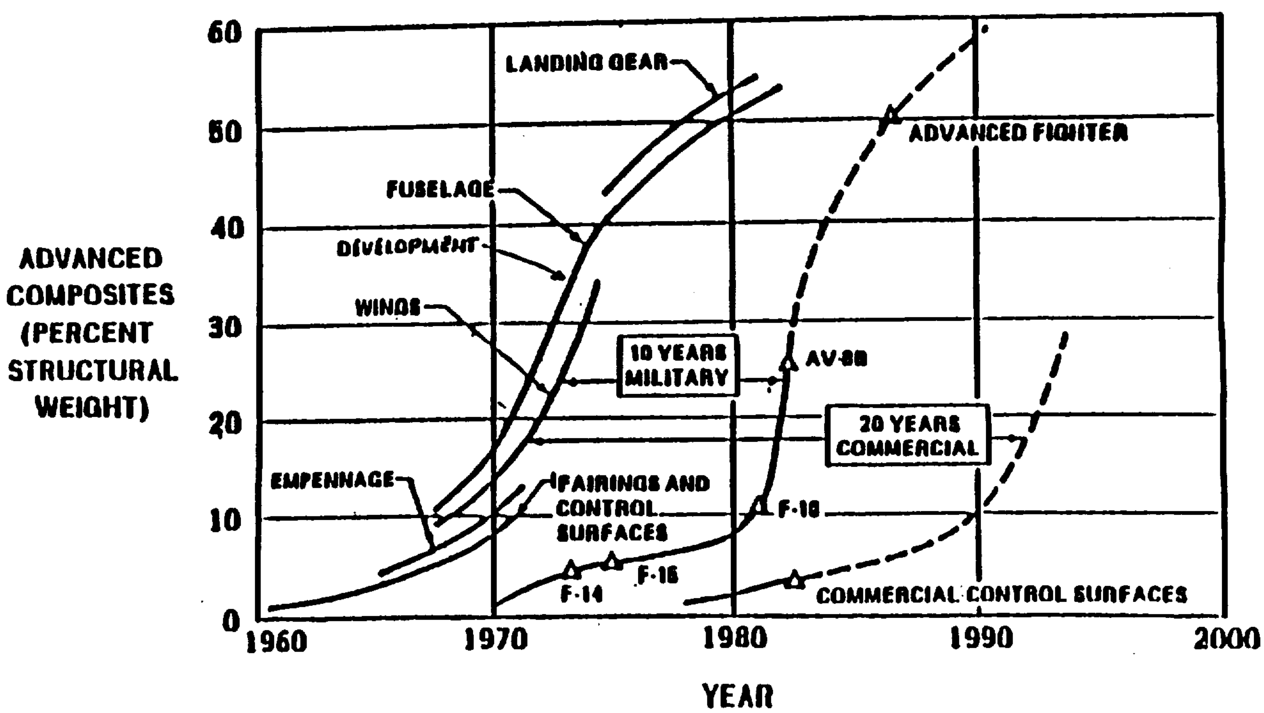
MD-12X COMPOSITE STRUCTURES



The primary message in this chart is that it has taken approximately 10 years from start of composite development activities at Wright Field until equivalent structure was applied to production military fighter aircraft. Grumman aircraft placed the first production composite structure in service, the Tom Cat stabilitator in 1970 and McDonnell Aircraft Company placed the F-15 speed brake and stabilizer skins in production in 1972. Primary wing structure took the same 10-year period from development in 1970 at Wright Field to F-18 and then AV-8B wing structure at McDonnell Aircraft Company in 1978-1982 time period. It is projected that the next fighter aircraft will have over 50 percent primary structure of composite material.

It took almost 20 years from initial development at Wright Field until voluntary commercial production of carbon fiber composite control surfaces started at Boeing and DAC transport divisions. If this 20-year trend holds true, and NASA ACT activity and support continues, it appears that a major jump into primary commercial transport wing application can be expected in the mid 1990's.

IMPLEMENTATION OF ADVANCED COMPOSITES ON AIRCRAFT



The greatest restraints to application of composites in commercial aircraft are the high cost of composite structure (materials plus fabrication and qualification costs), and the marginal resistance to damage tolerance and reduced compression after barely visible surface impact damage.

GREATEST RESTRAINTS TO APPLICATION OF COMPOSITES ON AIRCRAFT STRUCTURE

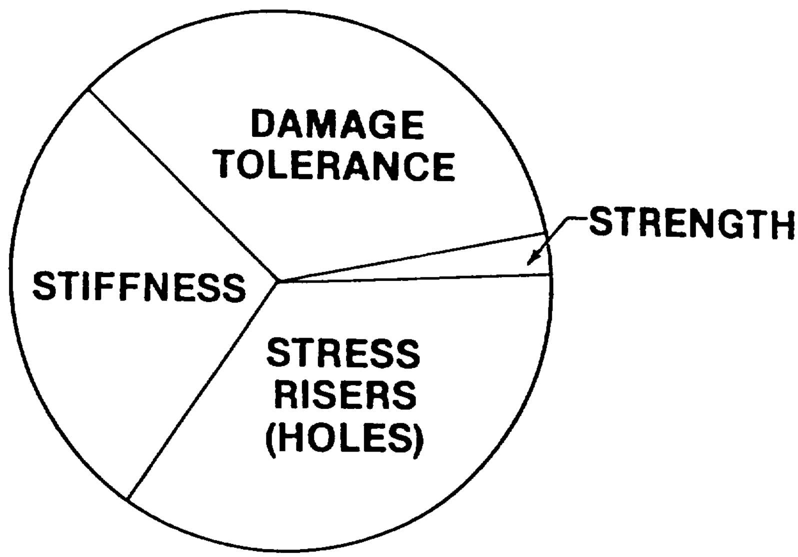
- **HIGH COST**

- **DAMAGE TOLERANCE**

Technical factors were identified that will limit the application of composites to transport aircraft primary structure. The analysis was based on the strength/stiffness capability of a 35m modulus, 520,000 tensile fiber, and a typical production resin system, Hercules 3501-6.

The analysis indicated that approximately one-third of the overall primary structure was critical in stiffness, one-third was critical in stress risers (cut-outs or bolted joints), one-third was critical in damage tolerance, and that only 2 or 3 percent was critical in strength.

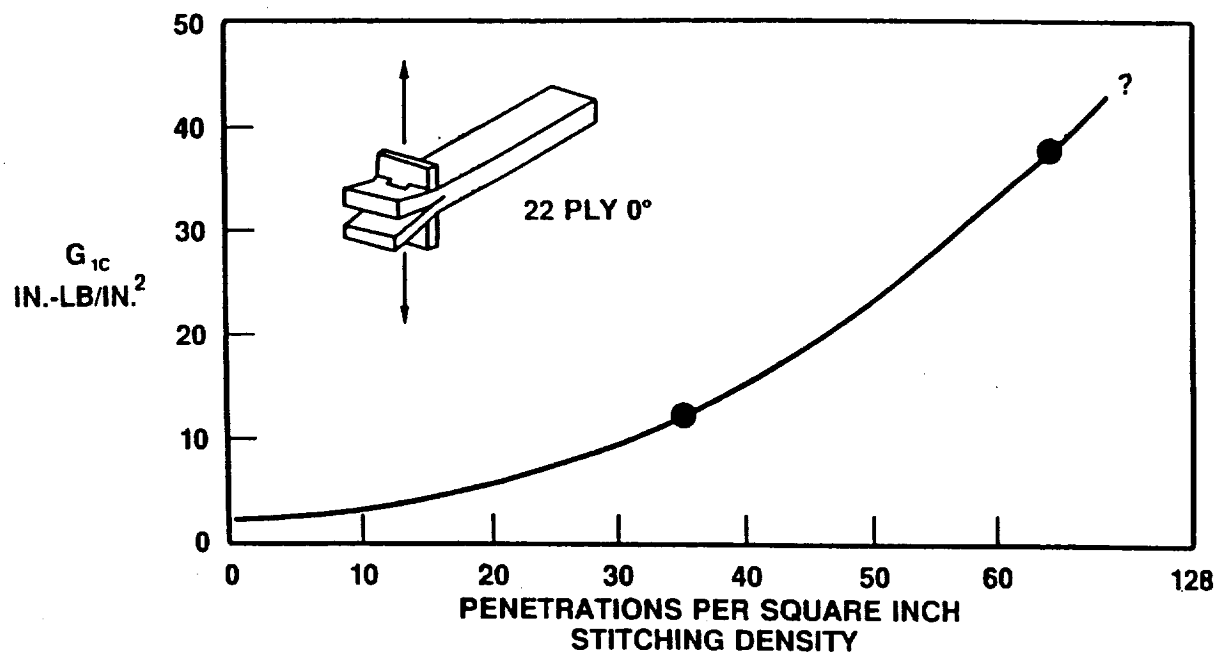
FACTORS THAT LIMIT APPLICATION OF COMPOSITE STRUCTURES IN A TYPICAL TRANSPORT AIRCRAFT



BASED ON CARBON FIBERS WITH 520,000 PSI TENSILE/ 35 M PSI MODULUS AND PRESENT PRODUCTION RESIN SYSTEMS (HERCULES 3501-6/AS-4)

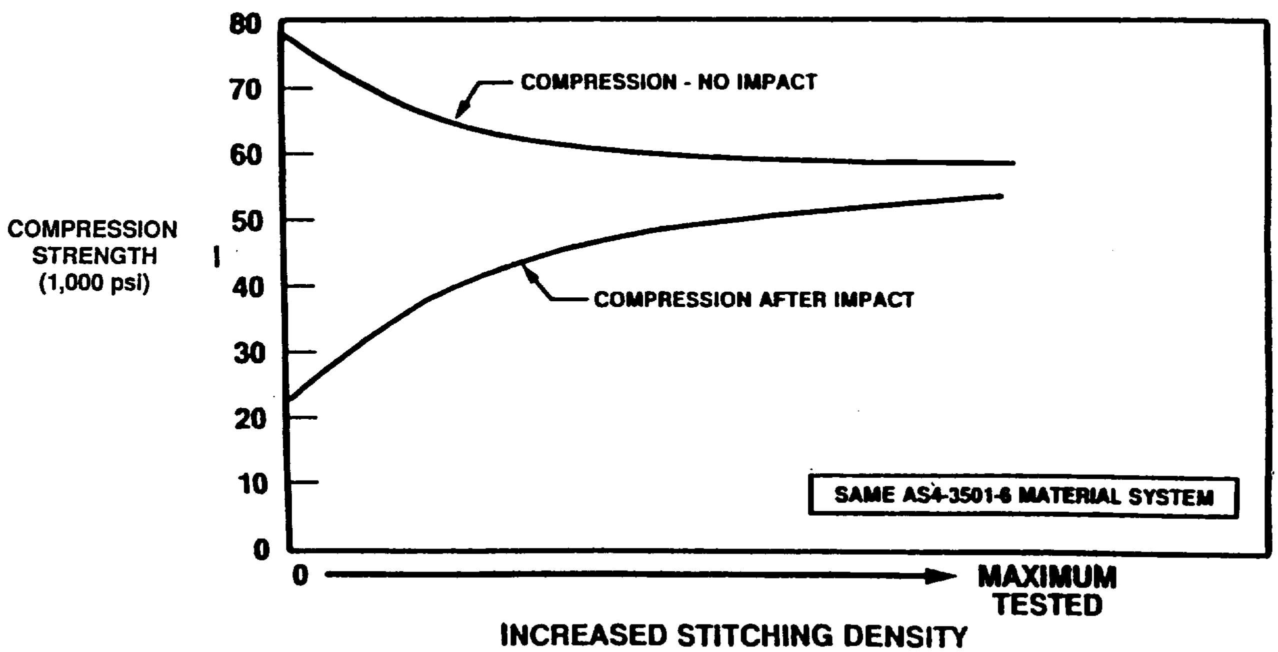
One measure of damage tolerance is resistance to peel or delamination forces. G_{IC}, crack growth rate, is a measure of delamination force. Through thickness stitching, using high strength glass or Kevlar thread, can improve G_{IC} from less than 1 in. lb/in. to over 36 in. lbs/in. while using the same structural fiber and matrix resin system. At near 36 in. lbs/in. G_{IC} peel force, specimen failure changes from peel or delamination to flexure failure at a row of stitching. Thus, it appears that stitching can eliminate G_{IC} or peel as a mode of propagation of failure in a composite laminate.

G_{IC} VERSUS STITCHING DENSITY



Retained strength in compression after barely visible impact damage is a critical mechanical property. As shown in this viewgraph, an undamaged compression panel with approximately 80,000 psi compression strength will have only about 20,000 psi compression strength after just visible impact damage. Through thickness "Z" axis stitching, with increasing stitch penetration density (penetrations per square inch) can improve CAI to over 55,000 psi while using the original reinforcement fibers and matrix resin system. Thus, structure that is damage tolerance critical can be much lighter in weight when the through thickness stitched fiber reinforcement is included in the design and fabrication.

INCREASED STITCHING DENSITY VERSUS COMPRESSION AFTER IMPACT



There are two major objectives in the DAC/NASA composites development program.

1. *Reduce Manufacturing Costs by 50 Percent*

   The cost is compared to the best procedures for "B" stage material layup and autoclave cure. This includes hand layup and automated equipment tape, tow or filament layup. The final goal is to produce composite structure that is comparable in cost to aluminum structure and gain a weight savings of 25 to 30 percent.

2. *Improve damage Tolerance by 100 Percent*

   The goal is to use the same lower cost resin system/fiber combination, and with the addition of "Z" axis stitched through thickness fibers, improve CAI by 100 percent. This will allow lighter weight structure to be designed where damage tolerance is critical.

OBJECTIVES

REDUCE MANUFACTURING COST BY 50 PERCENT

IMPROVE DAMAGE TOLERANCE BY 100 PERCENT

Many methods of producing near net shape preforms are under investigation by various developers. At DAC, after a study of each of the listed potential process, we have selected the knitting/stitching and the weaving/stitching as having the best overall potential to produce the large complex shape preforms required for the proposed transport aircraft structural components.

METHODS FOR NEAR NET PREFORM FABRICATION

- TWO-DIMENSIONAL ADVANCED WEAVING
- THREE-DIMENSIONAL WEAVING
- WEAVING KNITTING
- KNIT / KNIT
- IN-LINE MULTI-PLY KNIT
- BRAIDING
- IN-LINE MULTI-PLY THERMOPLASTIC HEAT SET
- KNITTING / STITCHING
- WEAVING / STITCHING
- FILAMENT WINDING

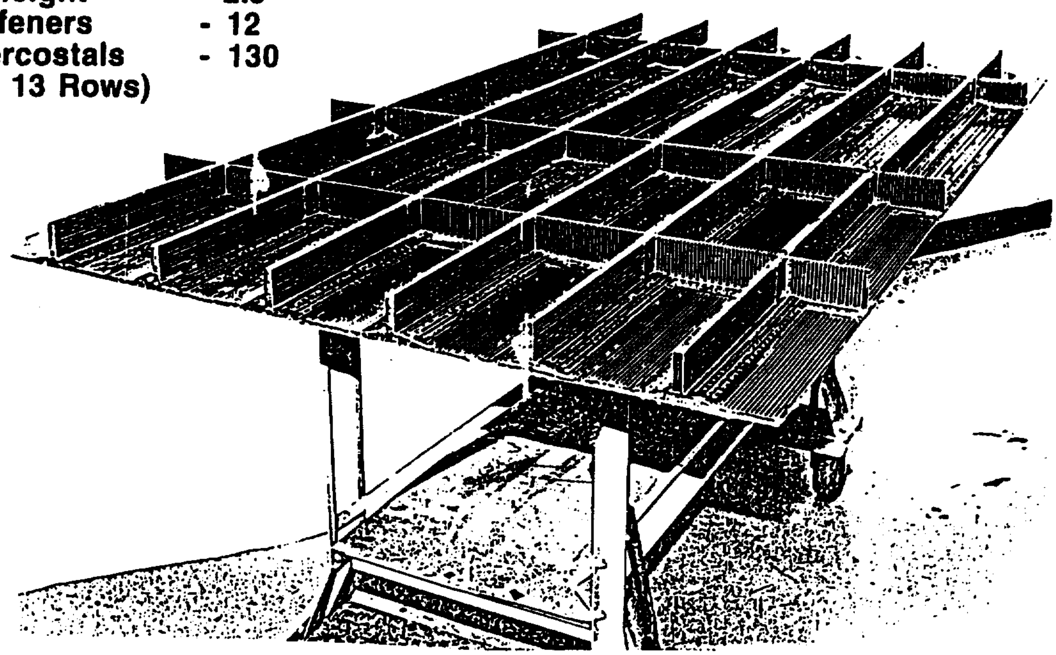
The blade stiffened panel design with 90° intercostal blade reinforcements used to demonstrate the dry fiber near net shape preform/resin infusion process (RIP) is shown in this viewgraph. This NASA demonstration wing subcomponent panel is 4-ft. x 6-ft. with 6 blade stiffeners and 3 rows of 7 individual intercostals that were all stitched together and then impregnated by the RIP (developed at DAC) and cured in the autoclave.

A similar wing panel design 8-ft. by 20-ft. with 12 stiffeners and 130 intercostals was used for later comparative cost estimating purposes.

SELECTED STIFFENED PANEL DESIGN

ASSUME:

Stiffened Panel	**- 8' X 20'**
Skin Thickness	**- 0.360"**
Stiffeners Thickness	**- 0.480"**
Stiffener Height	**- 2.5"**
Intercostals Thickness	**- 0.120"**
Intercostal Height	**- 2.5"**
Number Stiffeners	**- 12**
Number Intercostals (10 Rows x 13 Rows)	**- 130**



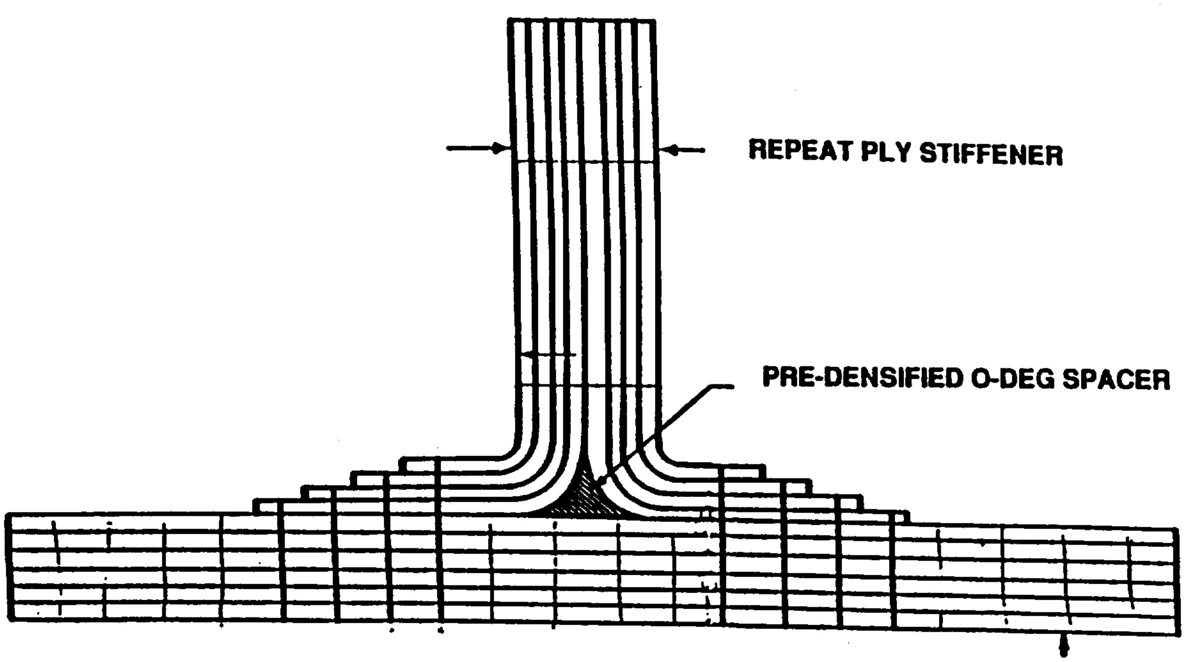
The basic design concept for the stiffened panel is shown in this viewgraph. The design is based on a 9-layer repeat pattern that is "lightly" stitched together as the first processing step. The skin is then made from 6 layers of this 9-ply material (54-ply total) and is high density stitched together to secure the 54 total layers and add damage tolerance to the skin.

The stiffener shown consists of 8 layers of 9-ply material (72-ply total) that are high density stitched together in the web area of the stiffener. The stiffeners are trimmed from the 72-ply sheet and flanges folded left and right and trimmed with four 9 layer steps. The flanges of the stiffener are then high density stitched to the skin for stiffener location and damage tolerance of stiffener to skin bond.

TEXTILE PREFORM - RTM

STITCHING OF BLADE STIFFENER TO SKIN

EXAMPLE



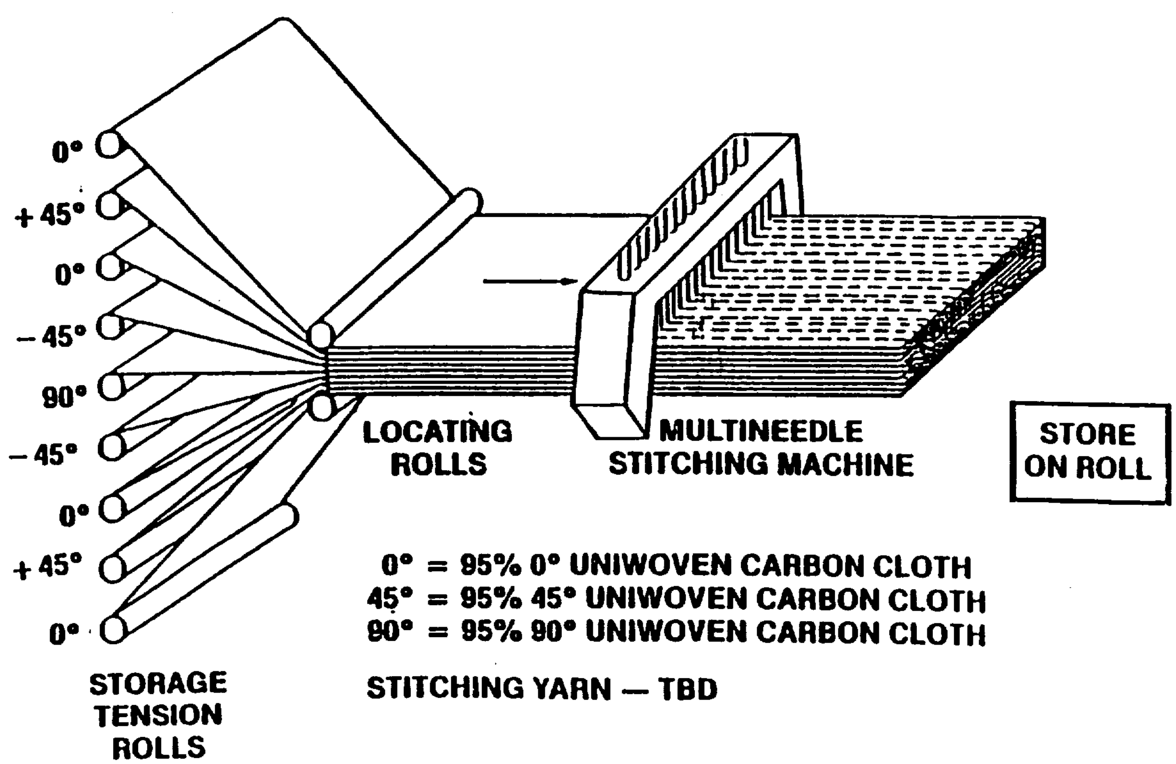
Uniwoven carbon fiber fabric can be manufactured by the weaving companies with 95 to 98 percent of the structural fibers in the 0°, the 45° or the 90° fiber directions. The remaining 2 to 5 percent fibers are fiberglass or polyester material to tie the fabric together.

This viewgraph shows one concept where 9 layers of material are metered from a tension storage rack through locating rolls and through a multi-needle stitching machine to stitch the 9 layers together. In this case, the needles are 1-in. apart and a light weight polyester or nylon thread is used. This stitching neither adds or subtracts from later laminate mechanical properties, but is merely to secure the 9 layers together so they can be later processed and handled as a single ply.

After light density stitching, the 9-layer material is stored on a large diameter roll and is ready for the next manufacturing step.

NINE-PLY BASIC STITCHED SUB-ELEMENT

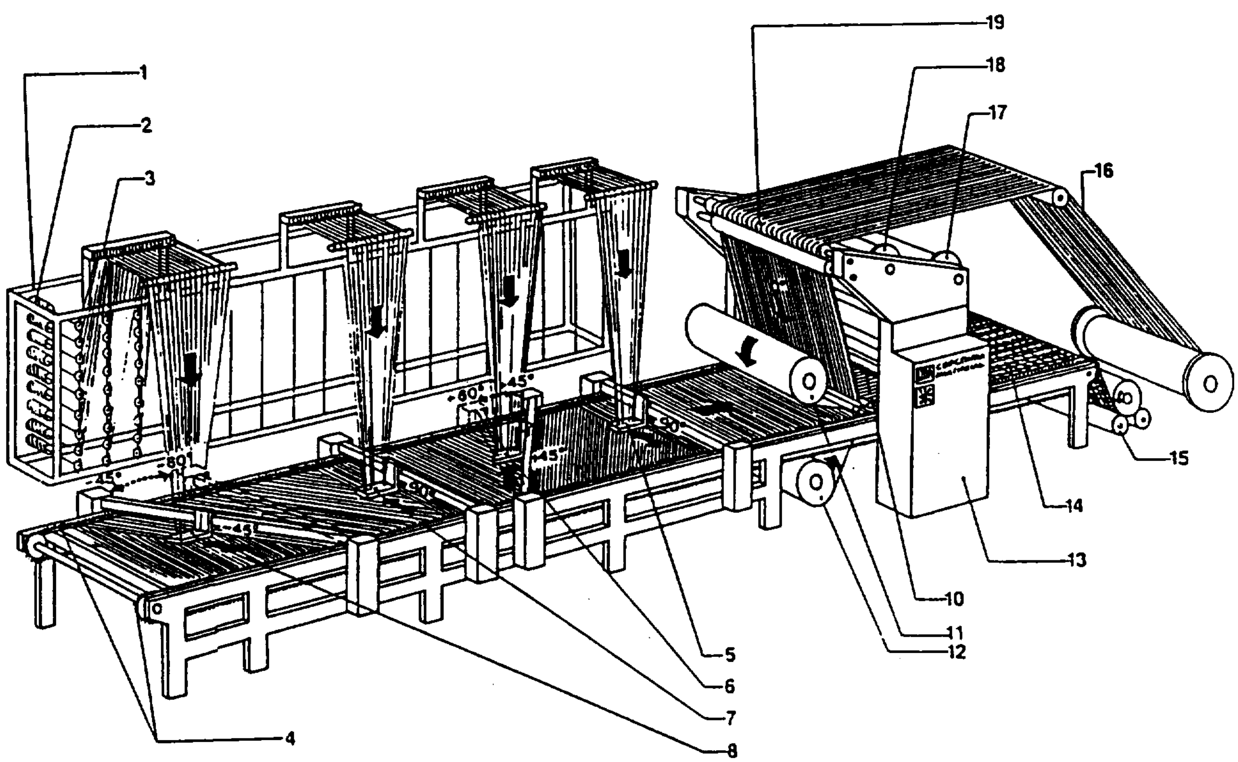
EXAMPLE



The Liba warp/knit machine is an alternate method of producing multi-oriented fiber layers of material. The 45° and the 90° layers are placed directly from the spools of fiber "tow" onto a moving belt with support pins along each edge. The 0° layer is first placed on a warp beam and then from the warp beam to the traveling belt with the 90° and 45° layers. The layers are all knit together immediately after the warp 0° layer insertion using a light polyester thread. As with the stitched uniwoven layers, the knit thread merely secures the layers of material together for later processing and adds nothing to the mechanical properties.

If the fiber pattern is acceptable, the Liba process of warp/knit is lower cost than an equivalent amount of uniwoven/stitched layered material and is a more desirable material form.

FUNCTIONAL DESCRIPTION OF THE LIBA-MULTI-AXIAL SYSTEM COPCENTRA MULTI-AXIAL, Version 5 (4 weft insertion systems):

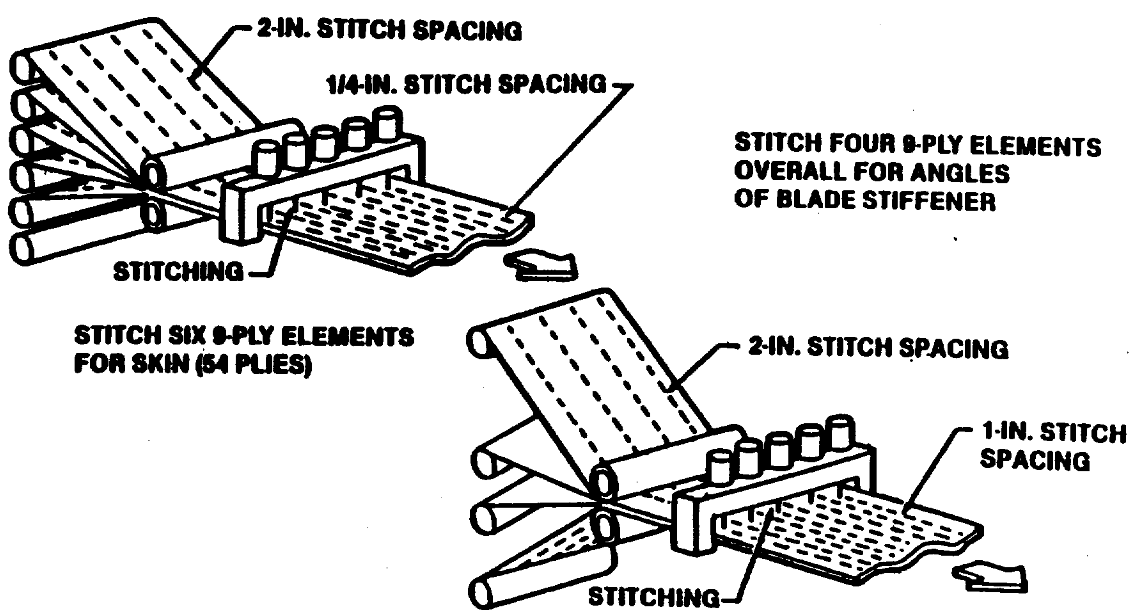


The figure on the left shows six 9-ply lightly stitched (54-ply total) layers of material passing from the tension storage rack through the multi-needle stitching machine. The stitch pattern is 0° rows, 1/4-in. apart as shown, and a high strength fiberglass or Kevlar thread is used. The resultant through thickness threads add damage tolerance to the finished panel.

The figure on the right shows four 9-ply lightly stitched (36-ply total) layers being stitched together on the multi-needle machine. This second step in preparation of panel blade stiffeners uses light-weight nylon or polyester thread merely to secure the 36 layers together but adds no mechanical properties to the final stiffeners.

STITCHING CONCEPT FOR BLADE-STIFFENED PANEL

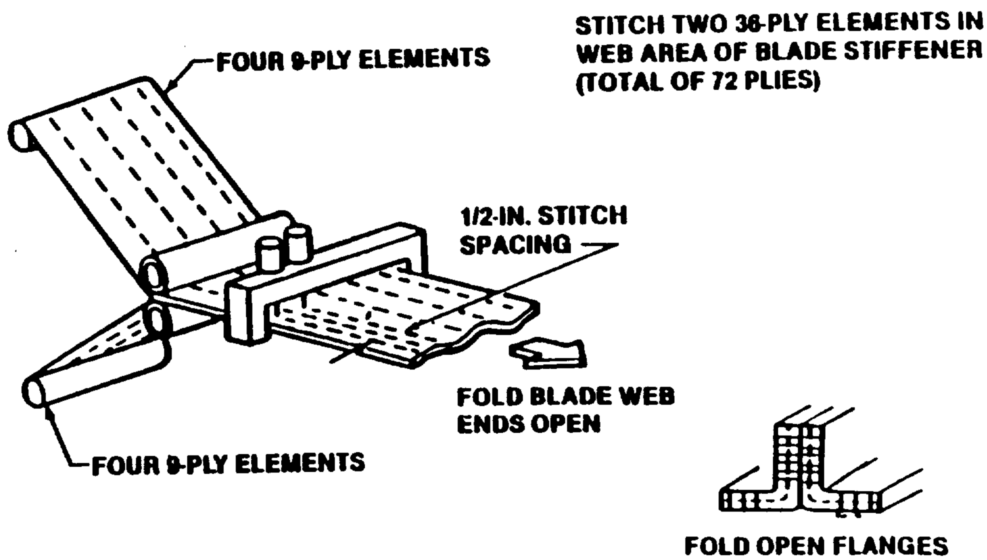
EXAMPLE



Two layers of 36-ply lightly stitched material are shown passing through the multi-needle machine. In this case the web area of the stiffener is the only area stitched (72-ply) using high density stitching and fiberglass or Kevlar thread to gain damage tolerance in the stiffener web area. As shown on the right, the stiffeners are cut to the desired width and flanges folded left and right to make the blade stiffeners.

STITCHING CONCEPT FOR BLADE-STIFFENED PANEL

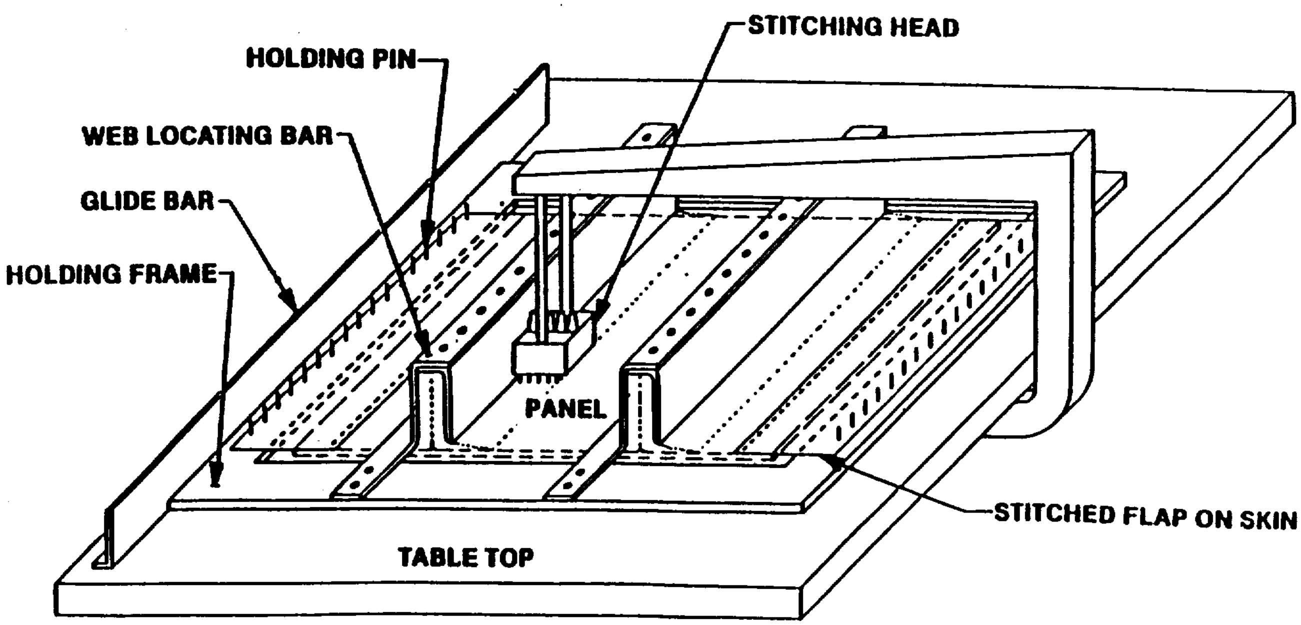
EXAMPLE



The 54-ply stitched skin is shown mounted in a holding picture frame in a long arm single needle stitching machine. The stiffener webs are secured in individual holding frames and the stiffener frames pin located to the skin picture frame. This viewgraph shows the stiffener flange being stitched to the skin using a 4-needle sewing head. In this case, the sewing head is in a fixed location and the work passed by the stitching head with stitching location achieved by the edge of the skin picture frame and a guide bar on the stitching machine.

STITCHING "T" FLANGE TO SKIN

EXAMPLE



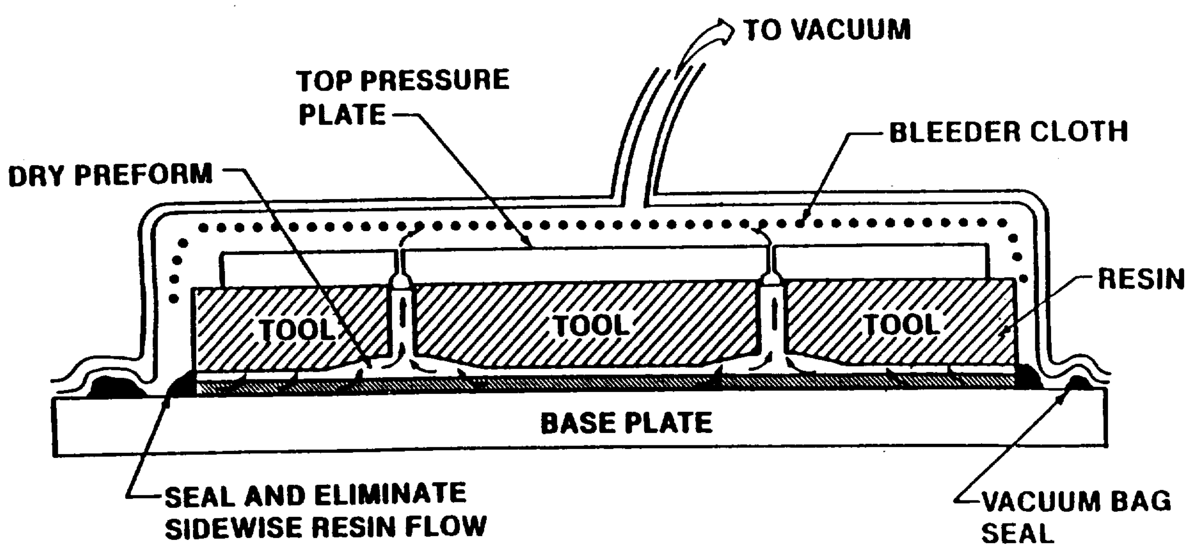
The stitched preform is next trimmed to final size to fit the tooling and weighed. A "B" stage resin slab is weighed and cast to the size of the skin and will give approximately 34 percent panel resin content.

The cast film of resin is located on the tool base plate. The preform is located over the resin. The tooling mandrels (3 shown) are located in position between the stiffener webs. The top pressure plate is located over the tool mandrels. The tooling mandrels are pin located through the top pressure plate to assure accurate location and thickness of the final cured stiffeners.

Bleeder cloth and a vacuum bag are placed over the entire assembly. The assembly is then placed in an autoclave for final heat and pressure cycle to impregnate the preform and cure the panel.

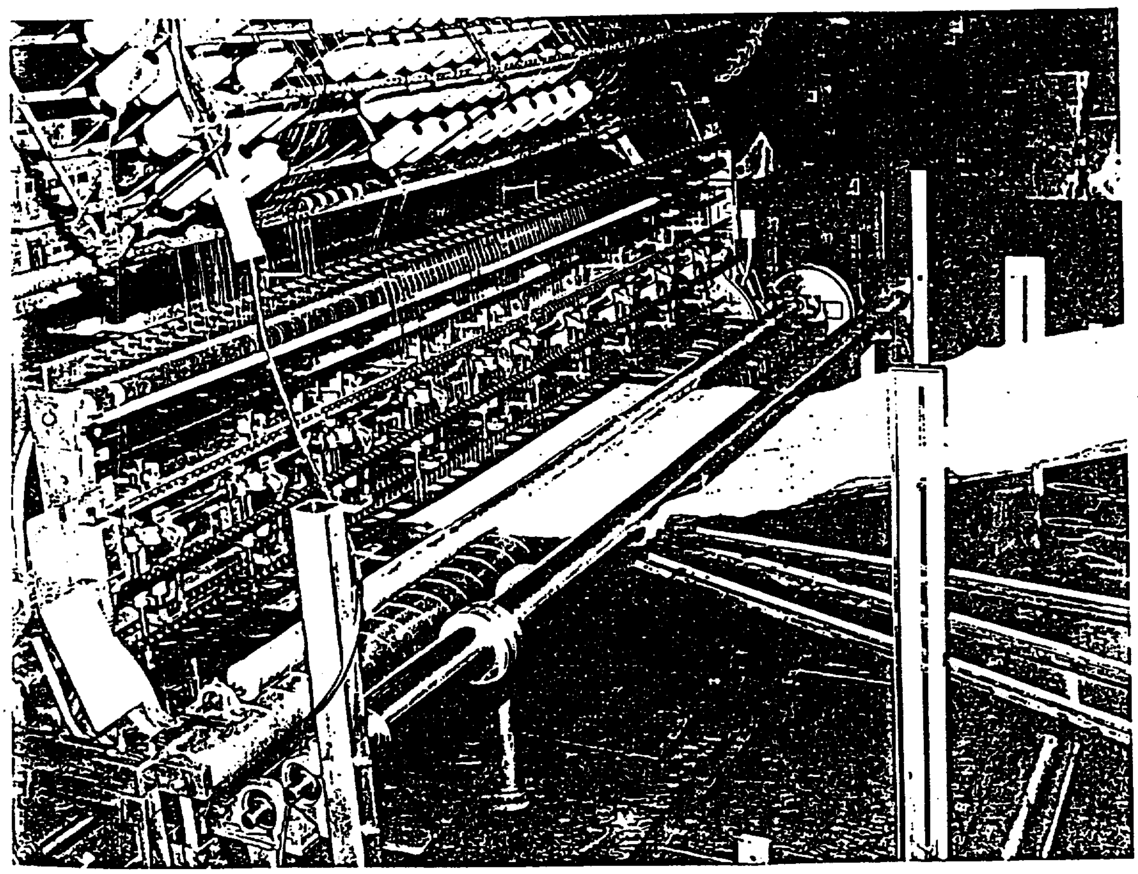
With heat, the resin becomes very thin and impregnates up through the skin, the blade stiffeners, and into the upper bleeder cloth. At this time the resin gels, and the cure cycle is completed with a standard autoclave curing cycle.

VACUUM IMPREGNATION OF STIFFENED PANEL



This is a photograph of the DAC multi-needle stitching machine, 128" wide, that was made by Pathe, Inc., Irvington, New Jersey. The machine has 128 needles and is shown with material feeding into the machine from a 12 roll tension storage rack. The stitching rate can be controlled from a few penetrations per minute to over 200 penetrations per minute - depending on the number of layers or thickness of material being stitched.

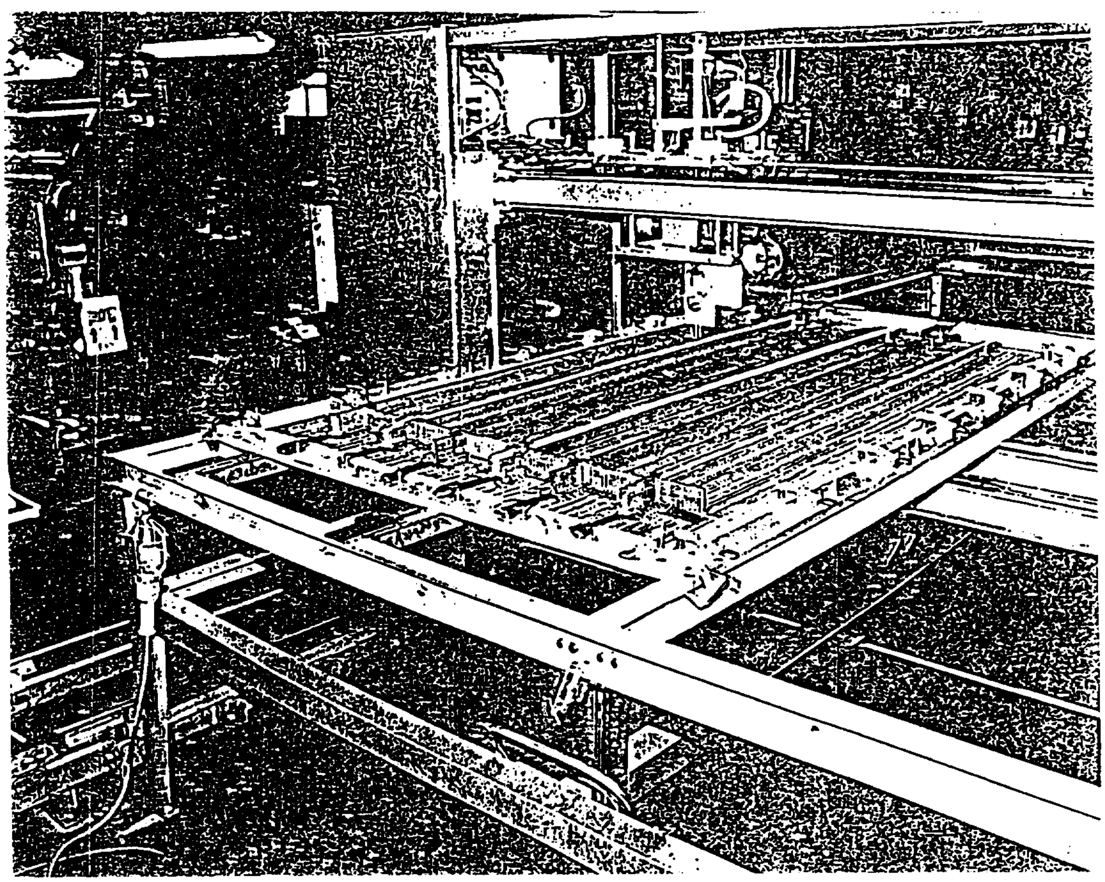
MULTI-NEEDLE STITCHING MACHINE 128" WIDE



(Original photo unavailable)

This is a photograph of the DAC single needle stitching machine, fully computer controlled for x-y-motion over an area 9-ft. wide and 15-ft. long. The material to be stitched is mounted in a holding frame, in a fixed location, and the stitching head moves to create the desired stitch pattern. A "lock" type stitch is used to allow stitching in any direction. This machine will stitch at rates between a few penetrations per minute, and over 400 penetrations per minute, depending on the thickness of the material being stitched.

COMPUTER CONTROL SINGLE NEEDLE STITCHING MACHINE 9' X 15' AREA



(Original photo unavailable)

A cost estimate has been made to produce an arbitrary size panel 8-ft. by 20-ft. in size with 12 lengthwise stiffeners and 10 rows of 13, each 90° intercostals (total 130).

Assumptions used in the preparation of cost estimates for material and stitching tape layup and tow placement labor are shown. Special note should be taken that all stitching tape layup and tow placement estimates are loaded with a 50 percent efficiency factor. Note also, that all equipment is assumed to be in working order and that repeat part fabrication is occurring so that no individual set up time is included and that no Quality Control (QC) costs are included in any of the estimates.

The estimates include all layup, bagging, and curing but no time is included for NDI, part trim, part finish, or part assembly.

COST ESTIMATE

THIS IS NOT A REAL PANEL. SIZE, THICKNESS AND PLY PATTERN ARE ARBITRARY.

COST ASSUMPTIONS:

MATERIAL	3K 35M FIBER	= $32 / POUND
	12K 35M FIBER	= $18 / POUND
	UNIWOVEN FABRIC	= 2 x FIBER
	"B" STAGE FABRIC	= 2 x FABRIC
	WARP/KNIT	= 2 x FIBER
	RESIN	= $20 / POUND

STITCHING	SET-UP TIME	= NOT ESTIMATED
	ALL STITCHING	= 50% EFFICIENCY
	THIN STITCHING	= 112 ppm
	THICK STITCHING	= 32 ppm
	MULTI-NEEDLE MACHINE	= 100 NEEDLES 1" APART
	SINGLE NEEDLE MACHINE	AUTOMATED CONTROL
	AUTOMATED TAPE LAYUP/TOW PLACEMENT	= 50% EFFICIENCY

LABOR	$50 / HOUR

This viewgraph shows the methodology used to estimate the cost of 100-in. wide unidirectional fabric (not yet purchased). All material costs followed this format.

Dry fiber 3k carbon cost of \$32/lb was supplied by the fiber manufacturer. Cost of 2x fiber cost for uniwoven fabric was considered proper by the fiber supplier/weaver at \$64/lb. The weight of 100-in. wide fabric, at 145 gm/m^2 fiber areal weight was calculated at 0.76 lbs/linear yard x \$64/lb = \$48.64/linear yard.

IM-7, higher modulus fiber at \$47/lb, for example uniwoven fabric with the same fiber areal weight would cost 0.76 lbs/linear yard x \$94/lb = 71.44/linear yard.

MATERIAL COST ESTIMATE

EXAMPLE

Dry Fiber Uniwoven 100" Wide

Assume:	**3K 35M Modulus fiber**	**\$32.00/lb**
	Uniwoven fabric 0°,45°,90°	**145 gm/m^2**
	Fabric width	**100 inches**
	Fabric cost = 2 x fiber cost	**\$64.00/lb**

Weight: **100" Wide fabric at 145 gm/m^2**

$$145\ \frac{\text{gm}}{\text{m}^2} \times \frac{1\text{m} \times 1\text{m}}{39" \times 39"} \times \frac{100" \times 36"}{\text{linear yard}} \times \frac{1\ \text{lb}}{454\ \text{gm}} = 0.76\ \text{lb/lin yd}$$

$$\text{Cost/linear Yard} = 0.76\ \frac{\text{lb}}{\text{lin yd}} \times \frac{\$64.00}{\text{lb}} = \$48.64/\text{lin yd}$$

"B" Stage Layup - 8-ft. by 20-ft. Panel

This is a summary of material and labor costs for an 8-ft. by 20-ft. 12 stiffener, 130 intercostal stiffened panel made by the "B" stage layup autoclave curing process. It includes individual breakdown of cost for skin, stiffeners, intercostals, and bonding costs for stiffeners and intercostals to skin.

The cost of materials was estimated following the procedure identified in viewgraph 25 and 26, cost estimate, and material cost estimate. An estimate of total square feet of "B" stage material was made and then reduced to pounds of carbon fiber. This weight was multiplied by 4 times the cost of fiber per pound to estimate total "B" stage material cost.

The labor cost for the 60-ply skin was 2 man hours estimate per ply, 8-ft. by 20-ft. x 60 plies. In addition, densification was estimated as vacuum for 1 hour, each 10 layers. The skin was estimated to be cured in a separate cure cycle. All 12 stiffeners and 130 intercostals were laid up, densified every 10 plies, and cured in separate cure cycles. The stiffeners and intercostals were trimmed to specification and secondary bonded to the skin.

Note that the cost estimate of $166,053 is the highest cost of all processes estimated and is over 5 times the cost estimate for equivalent aluminum structure. It should also be noted that most current composite production is by this method. We believe that great savings in composite fabrication cost is right around the corner.

The fiber/resin system used for this estimate was a 35m modulus fiber (Hercules AS-4 3k and a general purpose brittle resin, Hercules 3501-6 in "B" stage uniwoven form.

A toughened resin system such as Hercules 8551-2 resin, to achieve some damage tolerance would add about 50% to the material cost but no change in the labor cost.

MATERIAL AND LABOR COST ESTIMATE

"B" STAGE LAYUP
8' x 20' DRY FIBER PREFORM

	MATERIAL	LABOR	TOTAL
60-PLY 8' x 20' SKIN	28,493	7,200	35,693
12 STIFFENERS	14,246	25,200	39,446
130 INTERCOSTALS	2,714	45,500	48,214
	$45,453	$77,900	$123,353

\+ ROM ESTIMATE: BAG AND CURE PARTS
TRIM PARTS
SECONDARY BOND PARTS

TO COMPLETE SKIN: 854 HOURS x $50/HR = $42,700

TOTAL STIFFENED SKIN = $166,053

Dry Fiber Uniwoven/RIP 8-ft. by 20-ft. Panel

This is a summary of material and labor costs for an 8-ft. by 20-ft. 12 stiffener, 130 intercostal stiffened panel made by the dry fiber uniwoven/stitched preform Resin Infusion Process (RIP). It includes an individual breakdown of cost for skin, stiffeners, intercostals, bagging, and curing the complete assembly.

The uniwoven fabric material cost was estimated by first determining the number of square feet of material required, reduce this to pounds of fiber and then multiplying the pounds of fiber by 2x the fiber cost. This equaled the total cost of 9-ply material. The stitching cost was estimated for skin and stiffeners using a 100-needle machine, operating at 112 penetrations per minute plus a 50 percent efficiency factor. The intercostals and the stitching of stiffener and intercostal flanges to the skin were estimated using an automated single needle machine operating at 32 penetrations per minute plus a 50 percent efficiency factor. Total hours were multiplied by $50/hour to get the cost.

The dry preform has integral stitched skin, stiffeners, and intercostals that are all processed in one RIP cycle.

Note that the cost estimate of $66,722 is only 42 percent of the standard "B" stage uniwoven hand layup autoclave cure process, but still more than 2 times the cost of equivalent aluminum structure.

MATERIAL AND LABOR COST ESTIMATE

DRY FIBER UNIWOVEN - 100" WIDE
8' x 20' DRY FIBER PREFORM

	MATERIAL	*LABOR*	*TOTAL*
LIGHT DENSITY STITCH 10-PLY MATERIAL		8,400	8,400
60 PLY 8' x 20' SKIN	24,093	1,950	26,043
12 STIFFENERS	12,046	975	13,021
130 INTERCOSTALS	2,008	1,450	3,458
STITCH 12 STIFFENERS TO SKIN		9,400	9,400
STITCH 130 INTERCOSTALS TO SKIN		1,800	1,800
RESIN COST	2,200		2,200
TOTAL:	$40,347	$23,975	$64,322

+ ROM ESTIMATE: WEIGH RESIN - LOCATE
BAG AND CURE PANEL
TRIM TO SIZE

TO COMPLETE SKIN: 48 HOURS x $50/HR = $2,400.00

TOTAL STIFFENED SKIN = $66,722

Dry Fiber Warp/Knit/RIP 8-ft. by 20-ft. Panel

This is a summary of material and labor costs for an 8-ft. by 20-ft. 12 stiffener, 130 intercostal stiffened panel made by the dry fiber warp/knit/stitched preform Resin Infusion Process (RIP). It includes an individual breakdown of cost for skin, stiffeners, intercostals, bagging, and curing the complete assembly by RIP. The dry fiber preform has integral stitched skin, stiffeners, and intercostals that are all processed in one RIP cycle. The warp/knit fabric material cost was estimated as 2x the total fiber cost. This is the same method of material cost estimate as uniwoven fabric. The big difference in lower cost for warp/knit is that 12k tow is used in warp/knit and 3k tow is used in uniwoven fabric.

The stitching costs were estimated, using the 100-needle machine operation at 112 penetration/minute plus a 50 percent efficiency factor for skins and stiffeners. The intercostals and the stitching of intercostals and stiffener flanges to skin were estimated using a 2-needle automated stitching machine at 32 penetrations per minute plus a 50 percent efficiency factor.

Note that the cost estimate of $30,915 is only 18 percent of the standard "B" stage uniwoven hand layup autoclave cure process, and in addition, is only 95 percent of the estimated cost of a similar aluminum structure.

MATERIAL AND LABOR COST ESTIMATE

DRY FIBER WARP/KNIT - 100" WIDE 12k
8' x 20' DRY FIBER PREFORM

ASSUME: 12k tow = $18/lb
Warp/knit 10-ply fabric - 100" wide
50 percent stitching efficiency - no set up time
100 multi-needle machine - operational
Two needle computer controlled machine - operational

	MATERIAL	LABOR	TOTAL
60 PLY 8' x 20' SKIN	11,415	950	12,365
12 STIFFENERS	5,708	475	6,183
130 INTERCOSTALS	1,087	1,050	2,137
STITCH 12 STIFFENERS TO SKIN		4,200	4,200
STITCH 130 INTERCOSTALS TO SKIN		1,430	1,430
RESIN COST	2,200		2,200
IMPREGNATE/CURE/TRIM		2,400	2,400
TOTAL:	20,410	10,505	30,915

TOTAL STIFFENED SKIN = $30,915

Automated Tape Layup - Standard 8-ft. by 20-ft. Panel

This is a summary of material and labor cost for an 8-ft. by 20-ft. 12 stiffener, 130 intercostal stiffened panel made by the standard automated "B" stage tape layup/autoclave cure procedure. Six inch wide tape is laid up at a rate of 10 ft. per minute. It includes an individual breakdown of cost for skin, stiffeners, intercostals, bagging, and curing the complete assembly in the autoclave. This process assumes no stitching. The skin is automated tape layup. The stiffeners and intercostals have automated tape layup 4-ply material that is then hand laid up and densified on the mandrels. All individual "B" stage layups are then assembled on the "B" stage tape layup skin and cured together in one autoclave cure cycle.

Note that the cost estimate of $46,242 is only 35 percent of the standard "B" stage uniwoven hand layup autoclave cure process but is still 43 percent higher than the cost of equivalent aluminum structure.

MATERIAL AND LABOR COST ESTIMATE

AUTOMATED TAPE LAYUP - STANDARD

12 STIFFENER 8' x 20' PANEL

ASSUME: 12k tow = $18/lb - "B" Stage x 2 = $36/lb
Tape width = 6 in.
10 ft. per minute layup speed, 10-second turn time
Apply 50 percent efficiency
Co-cure skin/stiffeners/intercostals

	MATERIAL	LABOR	TOTAL
60 PLY 8' x 20' SKIN	19,051	3,500	22,551
12 60-PLY STIFFENERS	9,525	5,350	14,875
130 20-PLY INTERCOSTALS	1,116	4,650	5,716
ASSEMBLE ON TOOL		1,500	1,500
BAG, CURE, TRIM		1,600	1,600
TOTAL:	$29,692	$16,600	$46,242

TOTAL STIFFENED SKIN = $46,242

Automated Tape Layup - Advanced 8-ft. by 12-ft. Panel

This is a summary of material and labor cost for an 8-ft. by 20-ft. 12 stiffener, 130 intercostal stiffened panel made by the advanced automated "B" stage tape layup/autoclave cure process. Six inch wide tape is laid up at a rate of 30 ft. per minute. It includes an individual breakdown of cost for skin, stiffeners, intercostals, bagging, and curing the complete assembly in the autoclave. This process assumes no stitching. The skin is automated tape layup. The stiffeners and intercostals have automated tape layup 4-ply material that is then hand laid up and densified on the mandrels. All individual "B" stage layups are then assembled and cured together in one autoclave cure cycle.

Note that the cost estimate of $42,748 is only 26 percent of the standard "B" stage uniwoven hand layup autoclave cure process but it is still 32 percent higher than the cost of equivalent aluminum structure.

MATERIAL AND LABOR COST ESTIMATE

AUTOMATED TAPE LAYUP - ADVANCED
12 STIFFENER 8' x 20' PANEL

ASSUME: 12k tow = $18/lb - "B" Stage x 2 = $36/lb
Tape width = 6 in.
30 ft. per minute layup speed, 3-second turn time
Apply 50 percent efficiency
Co-cure skin/stiffeners/intercostals

	MATERIAL	LABOR	TOTAL
60 PLY 8' x 20' SKIN	19,051	1,383	20,434
12 60-PLY STIFFENERS	9,525	4,123	13,648
130 20-PLY INTERCOSTALS	1,116	4,450	5,566
ASSEMBLE ON TOOL		1,500	1,500
BAG, CURE, TRIM		1,600	1,600
TOTAL:	$29,692	$13,056	$42,748

TOTAL STIFFENED SKIN = $42,748

Automated "B" Stage Tow Placement 8-ft. by 20-ft. Panel

This is a summary of material and labor cost for an 8-ft. by 20-ft. 12 stiffener, 130 intercostal stiffened panel made by the automated "B" stage tow placement autoclave cure process. It assumes a band of tow 6 in. wide and tow lay down at 30 ft. per minute. It includes an individual breakdown of cost for skin, stiffeners, intercostals, bagging, and curing the complete assembly in the autoclave. This process assumes no stitching. The skin is made by automated tow placement, the stiffeners and intercostals made from 4-ply tow placement "sheets" that are then hand laid up and densified on individual mold mandrels. All individual details of skin, stiffeners, intercostals, and tooling mandrels are assembled and cured together in one autoclave curing process.

Note that the cost estimate of $42,795 is only 25 percent of the standard "B" stage uniwoven hand layup autoclave cure process but it is still 29 percent higher than the cost of equivalent aluminum structure.

MATERIAL AND LABOR COST ESTIMATE

TOW PLACEMENT
12 STIFFENER 8' x 20' PANEL

ASSUME: 12k tow = $18/lb - "B" Stage x 2 = $36/lb
Note: Use of multiple spools of tow to equal 6-in. band makes tow cost and 6-in. tape cost the same.
30 ft. per minute layup speed, 3 second turn time
Apply 50 percent efficiency
Co-cure skin/stiffeners/intercostals

	MATERIAL	LABOR	TOTAL
60 PLY 8' x 20' SKIN	19,051	1,383	20,434
12 60-PLY STIFFENERS	9,525	4,170	13,695
130 20-PLY INTERCOSTALS	1,116	4,450	5,566
ASSEMBLE ON TOOL		1,500	1,500
BAG, CURE, TRIM		1,600	1,600
TOTAL:	$29,692	$13,103	$42,795

TOTAL STIFFENED SKIN = $42,795

Aluminum Riveted Stiffener 8-ft. by 20-ft. Panel

This is a summary of material and labor cost for an 8-ft. by 20-ft. 12 stiffener, 130 intercostal all aluminum riveted stiffened panel. It includes an individual breakdown of cost for skin, stiffeners, intercostals, drilling and riveting of stiffeners and intercostals to the skin.

This estimate was prepared by a professional metal component production estimator and was sized to be equal structural performance to the composite panels.

The skin cost was extrapolated from real production cost of similar size/thickness metal structure. The machined-to-thickness skin was estimated as a purchase part that includes material and outside machining labor cost (all reported as material cost).

The stiffeners were priced as extruded aluminum details, extrapolated from existing production records plus final machine to size and flange taper labor.

The intercostals estimate was extrapolated from similar "purchased part records and included both material and outside source labor all as material cost.

The drilling and riveting of stiffeners and intercostals to skin assumes automated drivematic production rates for 90 percent of the rivets. The remaining 10 percent are estimated for hand drill and set of rivets.

Note that these are all production quantity estimates for the aluminum stiffened panel and there is *not* a 50 percent efficiency factor placed on any of the fabrication operations.

Note also that the cost estimate of $32,392 is only 20 percent of the standard "B" stage uniwoven hand layup autoclave cure process.

Equal performance aluminum structure cost is the target cost for our innovative composite materials, design, fabrication program at DAC.

MATERIAL AND LABOR COST ESTIMATE

ALUMINUM
12 STIFFENER 8' x 20' PANEL

	MATERIAL	LABOR	TOTAL
8' x 20 SKIN	4,156		4,156
12 STIFFENERS	3,870	8,966	12,836
130 INTERCOSTALS	13,000		13,000
ASSEMBLY		2,400	2,400
TOTAL	$21,056	$11,366	$32,392

ASSUMES: SKIN - PURCHASED
STIFFENERS - MATERIAL AND MACHINING
INTERCOSTALS - PURCHASED
ASSEMBLY - 90% AUTOMATED DRILL/RIVET (DRIVEMATIC)

Estimated Cost Summary - Material + Labor

8-ft. by 20-ft. Stiffened Panel

The viewgraph presents a materials and labor cost summary for fabrication of an aluminum 8-ft. by 20-ft. stiffened panel with six different methods of fabrication of similar composite structure.

These estimated cost numbers do not reflect the total panel cost - only materials and fabrication labor cost are included. There is no consideration for design, analysis, tooling, equipment, individual process set up time, quality control, or finishing costs included in this estimate.

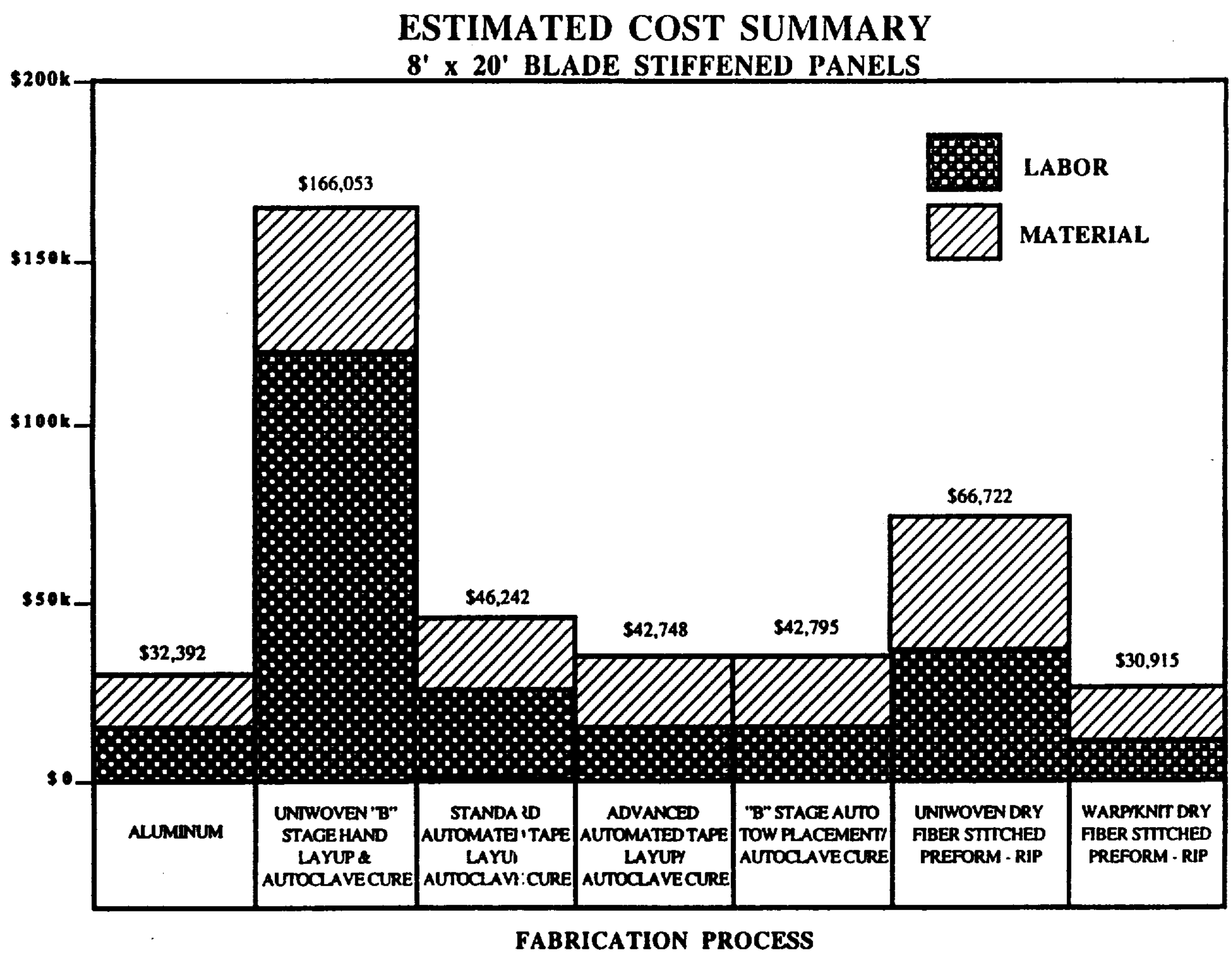
Note that the aluminum structure is estimated from large-scale production records. The composite cost estimates are conservative as shown by the use of automated equipment - 50 percent efficiency factors added to the processing time.

Note that the warp/knit/stitched dry fiber preform/RIP concept is the only composite materials and process method that was cost competitive with the aluminum structure.

8' x 20' BLADE STIFFENED PANELS

FABRICATION METHOD	MATERIAL	LABOR	TOTAL	% OF ALUMINUM
ALUMINUM	21,056	11,366	$32,392	100%
UNIWOVEN "B" STAGE HAND LAYUP & AUTOCLAVE CURE	45,453	120,600	$166,053	512%
STANDARD AUTOMATED TAPE LAYUP/AUTOCLAVE CURE - 12k TOW	29,692	16,550	$46,242	143%
ADVANCED AUTOMATED TAPE LAYUP/AUTOCLAVE CURE - 12k TOW	29,692	13,056	$42,748	132%
"B" STAGE AUTOMATED TOW PLACEMENT/AUTOCLAVE CURE - 12k TOW	29,692	13,103	$42,795	132%
UNIWOVEN DRY FIBER STITCHED PREFORM RIP IMPREGNATION AND CURE - 3k TOW	40,347	26,375	$66,722	205%
WARP/KNIT DRY FIBER STITCHED PREFORM RIP IMPREGNATION AND CURE - 12k TOW	20,410	10,505	$30,915	95%

This viewgraph is a bar chart showing all of the data presented in the previous viewgraph.



This review covered the following topics:

- There was a detailed discussion of the benefits that composites can bring to the aircraft
- Current applications of composites were identified for the McDonnell Douglas series of aircraft
- Critical issues of marginal composite damage tolerance and high cost were identified
- A DAC fabrication process for composite structure with affordable cost potential was described
- A series of cost estimates for different automated composite processing methods to fabricate an 8-ft. by 20-ft. stiffened panel were presented and compared to similar aluminum structure
- We feel we can meet the target of 100 percent improvement in composite damage tolerant structure
- We feel we can meet the challenge of composite structure cost near to similar metal structure cost

SUMMARY

DISCUSSED BENEFITS OF COMPOSITES - INCLUDES OTHER THAN WEIGHT SAVINGS

- **DISCUSSED AIRCRAFT APPLICATIONS - PRESENT AND FUTURE**
- **IDENTIFIED CRITICAL ISSUES - COST, DAMAGE TOLERANCE**
- **PRESENTED DAC *LOW COST FABRICATION* APPROACH**
- **PRESENTED COMPARATIVE COST ESTIMATES**

WE CAN MEET THE CHALLENGE OF EQUAL TO METAL COST.

WE CAN MEET THE CHALLENGE FOR 100% IMPROVED DAMAGE TOLERANCE.