NASA CR-
147704

MCDONNELL DOUGLAS TECHNICAL SERVICES CO.
HOUSTON ASTRONAUTICS DIVISION

SPACE SHUTTLE ENGINEERING AND OPERATIONS SUPPORT

DESIGN NOTE NO. 1.2-DN-B0104-03

ADVANCED COMPOSITES-FABRICATION PROCESSES FOR SELECTED RESIN MATRIX MATERIALS

ENGINEERING SYSTEMS ANALYSIS

31 March 1976

This Design Note is Submitted to NASA Under Task Order No. B0404 in Partial Fulfillment of Contract NAS 9-13970

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(NASA-CR-147704) ADVANCED COMPOSITES:
FABRICATION PROCESSES FOR SELECTED RESIN
MATRIX MATERIALS Space Shuttle Engineering
and Operations Support (McDonnell-Douglas
Technical Services) 49 p HC $4.00 CSCL 11D G3/24
N76-23360
Unclas
41456

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PREFACE

This report is the third of three reports planned to summarize the technology state-of-the-art for graphite and boron reinforced epoxy and polyimide matrix materials. Titles of the reports are as follows:

1.2-DN-B0104-1
"Advanced Composites - Mechanical Properties, and Hardware Programs for Selected Resin Matrix Materials"

1.2-DN-B0104-2
"Advanced Composites - Environment Effects on Selected Resin Matrix Materials"

1.2-DN-B0104-3
"Advanced Composites - Fabrication Processes for Selected Resin Matrix Materials"

The information presented herein on fabrication methods, tooling materials and methods, tolerances, machining, quality control and costs is given as a general aid to obtain a better designed composite structure. Raw material processing and component fabrication can strongly influence the mechanical and physical properties of the end item.

SUMMARY

This Design Note is based on present state of the art for epoxy and polyimide matrix composite fabrication technology. Boron/epoxy and polyimide and graphite/epoxy and polyimide structural parts can be successfully fabricated. Fabrication cycles for polyimide matrix composites have been shortened to near epoxy cycle times. Nondestructive testing (ultrasonics and radiography) has proven useful in detecting defects and anomalies in composite structural elements.

Fabrication methods and tooling materials are discussed along with the advantages and disadvantages of different tooling materials. Types of honeycomb core, material costs and fabrication methods are shown in table form for comparison. Fabrication limits based on tooling size, pressure capabilities and various machining operations are also discussed. Conventional cutting materials are not adequate for composites and, therefore, diamond tools are required. Prepreg forms and relative costs are listed; although these are constantly changing as composite fabrication technology continues to improve.

INTRODUCTION

This Design Note deals with basic processing procedures being utilized for fabricating graphite and boron epoxy and polyimide resin matrix composites. The combination of partially cured (B stage) thermoset resins and high strength fibers is called a prepreg. The binder or matrix is usually a thermosetting resin applied as a syrup to reinforcements which are usually in the form of strands. The prepreg is laid over a mold and cured with heat and pressure to form the desired shape. This basic method is the most widely used in the aerospace industry because of the following general characteristics:

(1) Ease of obtaining thickness variation and fiber orientation

(2) Ease of formation into large and/or complex shapes

(3) Adaptability to integral design and fabrication

Thermosetting resins such as epoxies are most often used as matrix materials in advanced composites. The reason epoxy resins are used is because they are 100% solids in the prepreg form and therefore, do not shrink upon curing. They also are among the strongest matrix materials available and allow the fullest development of the fiber properties. However, it is expected that some thermoplastics, particularly those whose glass-transition temperatures are very close to their melting temperatures, may also soon find use in composite applications. Such materials (polysulfones and benzimidizoles) appear to be excellent candidates. They offer attractive possibilities for rapid bonding and lamination because neither cure nor postcure is required.

Polyimide matrix composite parts can be fabricated using high pressure vacuum bag or moderate pressure autoclave/vacuum bag methods or by filament winding. All these methods require cure schedules at various temperatures and for varying periods of time. In the case of composites based on condensation-type polyimides, steps must be incorporated to permit the release of volatiles so that voids can be minimized.

The specific processing method for epoxy and polyimide prepregs must tolerate slight variables in raw material chemical and physical properties. These properties not only change from batch to batch but can also vary from roll to roll within the same batch. Most fabrication methods presented in this report can tolerate normal variables and produce finished material having acceptable mechanical and physical properties.

DISCUSSION

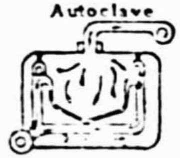
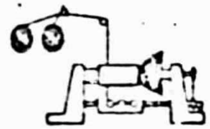
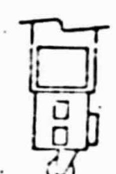
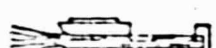
Fabrication Methods

Structural shapes are produced by applying prepreg, one layer at a time, to a mold of the desired shape until the proper thickness is obtained. The layup is then cured by the application of controlled heat and pressure. The resin in the various layers of the prepreg flow together and polymerize to bind the graphite or boron fibers into a strong non-homogeneous material. The thickness of the prepreg layup will normally decrease about 25% during cure because of resin flow and reinforcement compaction. Detail processing information contained in Appendix A are actual processes used to make specific aerospace hardware. The processes described are typically used by many companies and are considered practicable for fabricating highly loaded components. The processes cover graphite/epoxy, boron/epoxy, graphite/polyimide (HT-S/710) and graphite/polyimide (PI-2501).

Of the five processes shown in Table I only the autoclave and matched metal die methods form and cure the composite materials at the same time. The other three processes shown in Table I are methods for orienting or forming the fiber/matrix material. An additional operation is required with these processes to complete the curing process. The autoclave and filament winding processes currently are the most used processes.

The advantages of the autoclave process are: (1) Capability to orient fibers in specified directions, (2) Application of uniform fluid pressure to laminate, (3) Industry experience for fabricating fiberglass

TABLE I
COMPOSITE FABRICATION METHODS

Process	Usage	Configuration	Advantages	Constraints
Autoclave 	General laminating	Flat to complex contours Variable thickness laminate Bonded honeycomb structures	Oriented fibers Uniform pressure Compatible with adhesive and prepreg resins Industry experience Method lends itself to co-curing	Potential material movement in cure Autoclave size Autoclave characteristics • Heat-up rate • Pressurization rate • Temperature and pressure limit
Matched Metal Dies 	High production High dimensional control (all surfaces) Minimum machining required	Complex shapes	Oriented plies • Fibers can be oriented as required	Expensive tooling Hand layup of preform required
			Chopped fibers • High production potential	Expensive tooling Precise die filling required Lower strengths than autoclave process
Filament Winding 	Mandrel wound laminates	Surfaces of revolution	More rapid surface coverage Hoop tension loading of fibers Fiber tension uniformity	Difficult to maintain filament angle for variable contoured parts Poor protection for boron
Tape Laying 	High production General laminating	Flat to simple contours with constant or variable thickness	Low layup labor cost Low raw material scrap factor	Limited compound contour capabilities Setup labor required
Pultrusion 	Structural shapes	Angles, channels, rods, etc., of constant cross-section	Automated layup and cure continuous process	Straight or curved lengths of section Limited to shapes of constant cross-section

reinforced plastics is directly applicable, and (4) Allows cocuring of adhesives and prepregs together in a bonded composite structure.

Layup Tooling Methods

Tooling methods used for composites are not unlike those used for conventional fiberglass layups. In most cases curing pressure is applied to the composite layup by exhausting the air between the mold and a flexible impermeable covering (known as a vacuum bag) over the laminate. Ambient air pressure applied by the autoclave, thus exerts a uniform fluid force to the surface of the laminate and forces the layers of prepreg together. A coarse breather cloth laid between the vacuum bag and the laminate serves as a path for evacuation of air and volatiles released by the resin during cure.

Matched die laminates (when both surfaces are tool surfaces) generally require costly tooling and more sensitive fabrication procedures. When matched die tooling is perfected and processing temperature, time and pressure developed, parts can be produced in a rapid and economical manner. It still remains that the vacuum bag autoclave method is the most effective from a tooling development and cost standpoint.

The filament winding process produces mandrel wound laminates. The part configuration must be somewhat symmetrical and adaptable to revolving during the application of filaments. Filament winding easily lends itself to producing high strength cylinders, tanks and bottles. The advantages specifically are rapid surface coverage, hoop tension loading of fibers and fiber tension uniformity. A difficulty that can be encountered is maintaining filament angle for variable contoured parts.

The pultrusion method is a rapid, economical method of producing structural shapes. Preimpregnated fibers (bundles) are drawn through hot dies to form angles, channels, rods, etc., of constant cross section. The method essentially is an automated layup and continuous cure process. Constraints are that only straight or curved lengths can be made and extrusions are limited to shapes of constant cross section.

It appears that aluminum molds are the best for overall usage and cost. The only disadvantage of using aluminum is the high thermal coefficient of expansion compared to boron and graphite composites. This difference can cause graphite/epoxy parts to grow approximately 3/8 of an inch in 15 feet of longitudinal direction. This disadvantage can be avoided by proper consideration in tool design. Where costs prohibit the use of aluminum for compound curvature molds, chopped fiberglass/polyimide sprayed-up and cured as a master mold is the next best tooling approach. Table II lists the advantages and disadvantages of aluminum, fiberglass cloth/epoxy, chopped fiberglass/polyimide spray-up, and chopped fiberglass/epoxy spray-up tools. Table III shows the thermal coefficients of expansion for composite materials and metals that might be used for tooling.

TABLE II

ADVANCED COMPOSITE LAYUP TOOLING MATERIALS

TOOL TYPES	ADVANTAGES	DISADVANTAGES
ALUMINUM	LOW MATERIAL COST EASE OF FABRICATION EXCELLENT FOR SMALL DETAILS GOOD HEAT TRANSFER	EXPANSION RATE IN/IN/°F IS APPROXIMATELY 10 TIMES THAT OF EPOXY/GRAPHITE WHICH CAUSES DETAILS TO GROW APPROX. 3/8 IN. IN 15 FT
FIBERGLASS CLOTH EPOXY LAYUP	CAN BE FABRICATED COMPLETE ON A MASTER MOLD. EXCELLENT FOR COMPOUND SHAPES BETTER THAN ALUMINUM FOR EXPANSION	POOR HEAT TRANSMISSION DIFFICULT TO SEAL FOR VACUUM LEAKS DOES NOT RETAIN CONTOUR ON PRODUCTION RUNS AT BONDING TEMPERATURE
CHOPPED FIBERGLASS POLYIMIDE SPRAY-UP	SPRAYED AND CURED AS MASTER MOLD RETAINS CONTOUR AFTER REPEATED CYCLES NOT AFFECTED BY CURE TEMPERATURE MINIMUM MAINTENANCE REQUIRED	HIGH INITIAL COST FOR MATERIAL MORE MANHOURS TO FABRICATE
CHOPPED FIBERGLASS EPOXY SPRAY-UP	SPRAYED AND CURED AS MASTER MOLD EXCELLENT FOR COMPOUND SHAPES LESS EXPENSIVE THAN LAYUP TECHNIQUE NO VACUUM LEAKAGE	POOR HEAT TRANSMISSION TENDS TO CHANGE SHAPE AFTER REPEATED RUNS AT BONDING TEMPERATURE

TABLE III

THERMAL COEFFICIENT OF EXPANSION FOR SELECTED COMPOSITE AND METALLIC MATERIALS

Material	Thermal Coefficient of Expansion, (μin./in. °F)	
	Longitudinal	Transverse
Boron/epoxy [0]	2.3	10.7
Boron/epoxy [0_2/±45]	2.4	7.7
Graphite/epoxy [0]	0.3	14.4
Graphite/epoxy [0/±45/90]	1.9	1.9
E glass/epoxy [0]	4.8	---
E glass (181 style weave)/epoxy	5.5	6.7
KEVLAR-49/epoxy [0]	-6.0	---
E glass (181 style weave) / polyimide	0.74	---
Aluminum	13	
Low Carbon Steel	7	
Titanium	5.6	

Honeycomb Sandwich

General

Honeycomb sandwich structures are composed of thin, high strength facings bonded with a structural adhesive to a lightweight honeycomb core material. When rigidly bonded together, the three individual elements of a sandwich structure produce a single structure of far greater strength and rigidity than the sum of its individual parts. Overall panel design is normally simple even including edge details and fittings. Methods will be discussed in later paragraphs covering design procedures employing such details. Proper attention to fabrication details of bonded sandwich constructions can minimize costs and save weight.

Pressure, heat, and time are the basic bonding requirements for the production of honeycomb sandwich structural bonds. These parameters are inter-dependent and will be considered in that sequence in the following paragraphs. The bonding procedure consists generally of a preheat period, a "breathing" sequence for non face-to-core bonds, and the final cure period.

Pressure can be furnished by autoclave, press, diaphragm, or special fixtures. The proper pressure must be exerted on the bond line. Existence of sufficient pressure around the article being bonded is not evidence in itself that there is correct pressure at the bondline. Most bonds can be made with 100 psi. In some isolated designs, less pressure could be used to obtain good bonds. On the other hand, heavy sections (thick sheet material bonded to a forging) usually require pressures higher than 100 psi to bring the parts into satisfactory contact. For sandwich constructions, the maximum allowable bonding pressure is limited to the allowable compressive strength of the core material.

Heat is required at the bond line and can be ascertained by the use of thermocouples inserted in the adhesive. Tools and fixtures must be so designed that heat goes into the adhesive from both sides and not merely pass through the adhesive with exchange of heat taking place between two parts of the bonding fixture. This latter condition usually is a cause of unsatisfactory bonds.

The time to heat the tools and fixtures is most often a large part of the bonding cycle. Heat-up time, and likewise cool-down time, can be shortened by auxiliary means without affecting significantly the properties of the adhesive. However, any change in time to reach curing temperature does alter adhesive characteristics. Extremely long heat-up times can be detrimental to the final product. Just the opposite is true for time at cure temperature. When compared to a short cycle, a long curing cycle produces bonds with higher fatigue strengths, better creep characteristics, and higher strengths at elevated temperatures.

A comparison of honeycomb core types shown in Table IV shows that the so called "Flex Core" materials can conform to contours and do not require machining or hot forming to conform to single or compound curvatures. At service temperatures above 350°F the use of polyimide/fiberglass or titanium core should be considered.

TABLE IV

TYPICAL HONEYCOMB CORE CHARACTERISTICS

MATERIAL \ CHARACTERISTIC	CORE TYPE	COMPOUND CURVATURE FORMABILITY	MATERIAL COST * ($/FT^2)	GENERAL COMMENTS
5056-H39 AL	HEX	MUST BE MACHINED	0.70	ADDED MACHINING COST EXCESSIVE MATERIAL REQUIRED CELLS NOT NORMAL TO CONTOUR
5056-H39 AL	FLEX	CONFORMS TO CONTOUR	1.50	HIGHER STRENGTH TO DENSITY RATIO THAN FIBERGLASS LOWER COST THAN FLEX FIBERGLASS
MIL-T-9046 TYPE I TITANIUM	HEX	MUST BE MACHINED	--	ADDED MACHINING COST EXCESSIVE MATERIAL REQUIRED CELLS NOT NORMAL TO CONTOUR
FIBERGLASS/PHENOLIC	HEX	MUST BE MACHINED	2.50	ADDED MACHINING COST EXCESSIVE MATERIAL REQUIRED CELLS NOT NORMAL TO CONTOUR
FIBERGLASS/PHENOLIC	HEX-HEAT FORMED	HEAT FORMED BETWEEN MATCHED DIES	--	ADDED FORMING COST BUT CHEAPER THAN MACHINING WHEN REQUIRED IN QUANTITY
FIBERGLASS/PHENOLIC (HEXCEL-TSB120)	FLEX	CONFORMS TO CONTOUR	13.00	LOWER STRENGTH TO DENSITY RATIO THAN ALUMINUM VERY EXPENSIVE
POLYAMIDE PAPER/ PHENOLIC (HEXCEL-HRH-10)	FLEX	CONFORMS TO CONTOUR	18.00	LOWER STRENGTH TO DENSITY RATIO THAN ALUMINUM VERY EXPENSIVE

* MATERIAL COST IS BASED ON 4500 SQ. FT. OF LOW DENSITY, APPROX. 0.40 INCH THICK MATERIAL. COSTS SHOWN SHOULD ONLY BE USED FOR RELATIVE COST COMPARISONS.

Edge closeouts and attachments

From a fabrication viewpoint the design concept of honeycomb sandwich construction requires that edge closeouts attachments and panel splice details are an integral detail of the concept.

Closeout members along the edges of composite faced honeycomb panels are necessary to seal the edges, reduce potential damage to facesheets and the core, and provide required strength to the structure.

Typical edge closeouts and attachment members are shown in Figure 1. The following information supplements the information shown in the figure.

A. Filler Material - For this concept a filler material is used to fill all open honeycomb core cells around the edges of the panel. This material is generally trowelled to a smooth surface flush with the edges of the facesheets after the core is recessed. For panel attachment either bonded in precast plugs (installed during the initial panel layup), or bonded in metallic inserts (installed after the panels are cured) are required for each fastener.

B. Zee Closeout - As shown, the closeout member is shaped in the form of a "Z". For the composite panel the zee and, therefore, closeout is an integral part of the panel.

C. Tapered Core - For this concept the inner facesheet is joggled to conform to a tapered core. Composite facesheets are shaped by draping during the layup operation.

D. Channel Closeout - As in the previous concept a separate part in

EDGE CLOSEOUTS AND ATTACH CONCEPTS

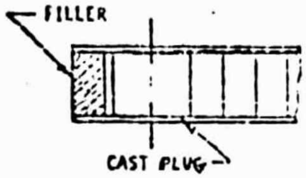
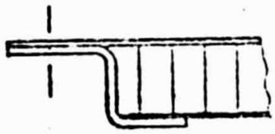
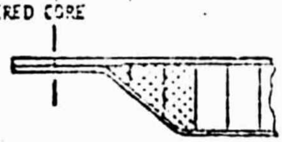
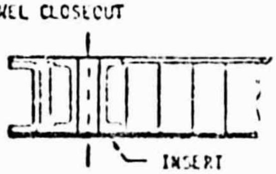
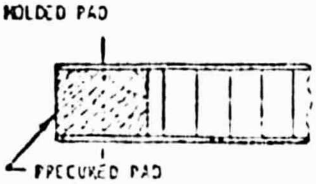
CONCEPTS	ADVANTAGES	DISADVANTAGES
FILLER MATERIAL 	PANELS CAN BE SIZED AFTER AUTOCLAVE PROCESSING	ADDING FILLER STRIP REQUIRES POST PROCESSING STRESS CONCENTRATIONS MAY REQUIRE LOCAL SKIN THICKNESS INCREASE INDIVIDUAL PLUGS OR INSERTS REQUIRE ADDITIONAL FABRICATION TIME
ZEE CLOSEOUT 	LIGHT WEIGHT PANELS CAN BE TRIMMED AFTER AUTOCLAVE PROCESSING ZEE SECTION CAN BE LAYED UP AND CURED WITH COMPOSITE PANEL	MAY REQUIRE LOCAL SKIN THICKNESS INCREASE REQUIRES CLOSE TOLERANCE CORE THICKNESS AT Z CLOSEOUT DIFFICULT CORE SHEAR TIE
TAPERED CORE 	PANELS CAN BE SIZED AFTER AUTOCLAVE PROCESSING NO SEPARATE CLOSEOUT MEMBERS REQUIRED	COMPLICATES PANEL LAYUP KICK LOADS IN LOWER SKIN TAPERED CORE MUST BE CRUSH FORMED OR MACHINED AND STABILIZED MAY REQUIRE LOCAL SKIN THICKNESS INCREASE
CHANNEL CLOSEOUT 	PANELS CAN BE SIZED AFTER AUTOCLAVE PROCESSING	EACH FASTENER LOCATION REQUIRES INSERTS PROCESSING TIME CONSUMING STRESS CONCENTRATIONS MAY REQUIRE LOCAL SKIN THICKNESS INCREASE REQUIRES FABRICATING SEPARATE CLOSEOUT MEMBERS
MOLDED PAD 	PANELS CAN BE SIZED AFTER AUTOCLAVE PROCESSING DAMAGED HOLES CAN BE PLUGGED AND REDRILLED	SOLID PAD PLUS ADJACENT ADHESIVE CREATES LOCAL WEIGHT PENALTY REQUIRES FABRICATING SEPARATE CLOSEOUT MEMBERS

FIGURE 1

the form of a channel is used as the closeout member. Metallic inserts are installed after curing.

E. Molded Pad - For the last concept illustrated, a low density reinforced plastic laminate is precured, conforming to shape and thickness requirements. During the panel layup operation this part is located between the facesheets adjacent to the trimmed honeycomb core and is simultaneously bonded to both facesheets and core. Panel attach holes are machined, thru pad and facesheets after panel cure.

Panel to panel splices

When a design is too large to consider fabricating as a single honeycomb panel sandwich, smaller panels are used which must then be joined together in some way to make a large panel.

Some panel-to-panel splice concepts in current use are shown in Figure 2. As illustrated in the figure, some splice concepts employ panel closeout concepts discussed in preceding paragraphs. There are, of course, many variations to these basic splice concepts that may be used depending upon application.

A. Panel Butt Splice - This concept configures the mating edges of the panels per the tapered core closeout member approach shown in Figure 1. Each panel is attached to a "T" stiffening member, which may coincide with existing substructure bulkheads and/or longeron locations.

PANEL-TO-PANEL SPLICE CONCEPTS

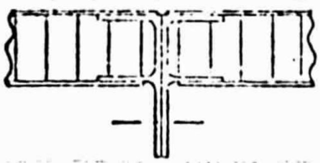
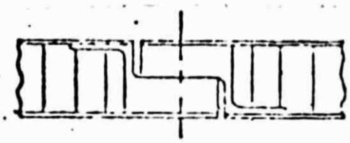
CONCEPT	ADVANTAGES	DISADVANTAGES
PANEL BUTT SPLICE 	• NO SEPARATE CLOSEOUT MEMBER REQUIRED. • LIGHT WEIGHT IF PANELS ARE SIZED TO COINCIDE WITH SUBSTRUCTURE (BULKHEADS AND LONGERONS)	• COMPLICATES PANEL LAYUP • REQUIRES USE OF DOUBLE ROW OF FASTENERS • TAPERED CORE MUST BE CRUSHED FORMED OR MACHINED AND STABILIZED • MAY REQUIRE LOCAL SKIN THICKNESS INCREASE
OVERLAP BOND 	• SIMPLE PANEL LAYUP	• PROTRUDING SPLICE REQUIRES STEP IN SUBSTRUCTURE
CHANNEL CLOSURE BUTT SPLICE 	• NO HOLES REQUIRED IN PANEL	• COMPLICATES PANEL LAYUP • DIFFICULT TO SIZE-NO TRIM PROVISIONS FOR ADJUSTMENT AFTER CURE • DIFFICULT TO OBTAIN ADEQUATE PRESSURE DURING CURE CYCLE
OFFSET CLOSURE OVERLAP 	• REQUIRES SINGLE ROW OF FASTENERS.	• COMPLICATES PANEL LAYUP • SOLID CROSS SECTION ADDS WEIGHT • REQUIRES POST CURE MACHINING TO CONTROL MOLDLINE MISMATCH
ZEE CLOSURE BUTT SPLICE 	• MIN. WEIGHT IF PANELS ARE SIZED TO COINCIDE WITH STRUCTURE. • ZEE SECTION CAN BE LAYED UP AND CURED WITH COMPOSITE PANEL.	• MAY REQUIRE LOCAL SKIN THICKNESS INCREASE • REQUIRES USE OF DOUBLE ROW OF FASTENERS

FIGURE 2

B. Overlap Bond - In this appraoch, doublers spanning both the upper and lower gaps are bonded to the facesheets. As illustrated these doublers are external to the facesheet surfaces, however, by machining or providing steps in the core material, flush external surfaces can be provided, if required, but would add complications to fabrication procedures.

C. Channel Closure Butt Splice - The channel shaped closure member not only permits using a single row of fasteners for assembling the panels but also keeps the external surface smooth by using fasteners at internally located flanges.

D. Offset Closure Overlap - This concept, also designed for a single row of fasteners, uses a stepped overlapping machined closure member for the all honeycomb panel design. For the composite approach this edge member would be layed up as an integral part during the panel fabrication procedure. As shown this splice is not dependent upon additional support members and therefore, has less imposed location restrictions except for attachment bolt location.

E. Zee Closure Butt Splice - For this concept closure members are identical to those described earlier and employs two rows of fasteners for attaching panels to "T shaped" stiffeners.

Lamina Thickness Tolerances

Advanced composite organic matrix laminates are thickness-critical, with filament volume fraction and composite laminate mechanical properties directly related to final ply thickness. Table V contains

TABLE V

TYPICAL UNCURED AND CURED LAMINA PARAMETERS

Lamina Material	Uncured Ply Thickness t_u * (mils/ply)	Cured Ply Thickness t_c (mils/ply)	Bulk Factor t_u/t_c	Cured Laminate Filament Volume Fraction V_f
Unsupported boron/epoxy (4.0 mil)	4.8-5.0	4.7 max 4.5 nom 4.3 min	1.15	60-65% boron
Supported boron/epoxy (4.0 mil)	6.7-6.9	5.4 max 5.2 nom 5.1 min	1.31	52% boron 8% fiberglass
Supported boron/epoxy (5.6 mil)	8.8-9.1	6.9 max 6.8 nom 6.7 min	1.31	52% boron 8% fiberglass
Graphite/epoxy (oriented ply layups of all fiber types)	8.1-8.7	6.4 max* 6.0 nom 5.6 min*	1.37	60% graphite

* Most prevalent range. Where required, uncured ply thicknesses may be obtained to allow designs as thin as 0.002 inch or as thick as 0.010 inch per ply.

thicknesses, bulk factor and cured laminate filament volume fraction data for typical uncured and cured lamina.

Fabrication Limits

Table VI shows the maximum size of equipment currently being used by, or available to the aerospace industry. Size limitations for advanced composite structures are related to equipment capabilities and the ability to provide the required time-temperature-pressure cure cycle established for the particular matrix of the composite system. Feasibility of manufacturing large, complex laminates and sandwich structures utilizing advanced composites has been demonstrated by major airframe manufacturers. Structures such as the F-111 aft fuselage component, C-5A wing leading edge slat, and the F-100 wing skin are indicative of size and structural complexity achieved by present technology. These should not be considered as the maximum size possible.

For autoclave, platen press, or integrally heated tool cured structure, the pressure and temperature requirements for epoxy matrix composites generally used is approximately 100 psi and 350°F. Matched die epoxy matrix shortfiber molding requires approximately 2,000 psi and 380°F. As stated previously, there is a cure pressure restriction for sandwich panels having light weight honeycomb core. Table VII shows the maximum pressure allowed for different honeycomb core materials.

TABLE VI

AVAILABLE EQUIPMENT SIZE LIMITATIONS
FOR RESIN MATRIX COMPOSITES

Equipment Type	Length (ft)	Width (ft)
Autoclave	60	22 (dia)
Platen press	20	10
Tape-laying machine	36	6
Integrally heated tooling	No limit	No limit
Filament winding	40	20 (dia)

TABLE VII

MAXIMUM CURE PRESSURES FOR 5052 H-39 H/C CORE

CELL SIZE (inch)	FOIL THICKNESS (inch)	PRESSURE (PSI)
3/16	.0015	42 ± 3
3/16	.001	34 ± 3
1/4	.001	24 ± 3

Machining

The information in Table VIII is taken from Reference 7 and shows the type of tools that should be used for various machining operations and the tolerances, limitations, and finishes that can be expected.

Graphite and boron/epoxy or polyimide composites present a challenge in the area of machining. The matrix is relatively easy to machine, but its comparative weakness to the reinforcement, in some cases, does not provide adequate support to filaments to prevent fiber break-out. The boron filament presents other difficulties because its hardness approaches 9.5 on the Mohs scale. Conventional cutting materials, such as tungsten carbide, aluminum oxide, silicon carbide, and steel, are softer than boron. Diamond tools, with a value of 10 on the Mohs scale, have been shown to be effective for machining graphite or boron/epoxy or polyimide composites, including those which may contain metallic interleaves.

TABLE VIII

COMPOSITE MACHINING METHOD CHARACTERISTICS

Operation	Graphite or Boron/Epoxy or Polyimide		
	Tooling	Limitations	Nominal Finish
Drilling	Diamond core drill *	Hole tolerance is ± 0.002 inch and requires flood coolant and backup.	100 to 200 μ inch (AA)
Reaming	Diamond reamer	Hole tolerance is ± 0.002 inch and requires flood coolant and backup.	100 to 200 μ inch (AA)
Countersinking	Diamond tools	None	Good
Routing	Diamond-slotted tools	None	Good
Milling	Diamond cup wheel	Not suited to internal pockets	100 μ inch (AA)

* Graphite composites may be drilled with carbide drills.

Quality Control

In general the quality control procedures associated with fiberglass reinforced laminates can be expected with boron or graphite reinforced composites. The structural composites will need to be inspected as a final completed component or assembly in addition to performing numerous "in process" control operations. Laminate areas can have the following defects: delaminations or debonds between plies; porosity; resin variations due to incomplete or no cure; fiber orientation non-conforming to drawing; mislocated thickness variations; and fiber defects. In addition bonded composite assemblies, either integral or secondary bonded, can have core damage or inclusions occur during the bonding operation. As is done in the case of fiberglass reinforced structures, nondestructive inspection methods are normally used to find the defects. The nondestructive test methods most generally used are radiography, ultrasonics, sonics, and thermography. Specific methods used to detect various defects are shown in Table IX.

Prepreg Forms

Advanced composite raw materials are available in many standard and special - application forms. The most commonly used epoxy resin prepreg forms are shown in Table X. Boron prepreg tapes generally include a supporting carrier of woven fiberglass fabric. Graphite fiber prepreg forms are the same for the three principal types of filament: high-modulus, high-strength, and intermediate-strength.

TABLE IX

NDT DEFECT DETECTION CAPABILITY

Defect / NDT Method	Delaminations	Debonds	Porosity	Resin Variations	Fiber Orientation	Sandwich Assemblies: Core Damage	Sandwich Assemblies: Inclusions	Mislocated Details	Fiber Defects
Radiography	C	C	B	B	B	A	A	A	B
Ultrasonics	A	A	A	B/A	B	B	B	B	B
Sonics	A	A	C	C	D	B	B	C	D
Thermography	B	B	B	C	C	C	B	A	C

Legend: A - Good capability C - Poor capability
B - Fair capability D - No capability

TABLE X

PREPREG FORMS FOR EPOXY MATRIX COMPOSITES

Prepreg Form	Boron Filament: 4.0 mil dia	Boron Filament: 5.6 mil dia	Graphite Fiber: Short Staple	Graphite Fiber: Continuous
3-inch tape	Continuous roll	Continuous roll		Continuous roll
Narrow and wide tape	Special order	Special order		Special order 1/8 to 14 in.
Sheet			12 x 45 inches standard size	
Broadgoods (drum wrapped)	Special order	Special order		Special order

Various thickness graphite prepregs are available, with cured thickness specified at 0.0056 to 0.0064 inch per ply being the most common. Broadgoods length and width dimensions are a function of the individual prepreger's facility equipment.

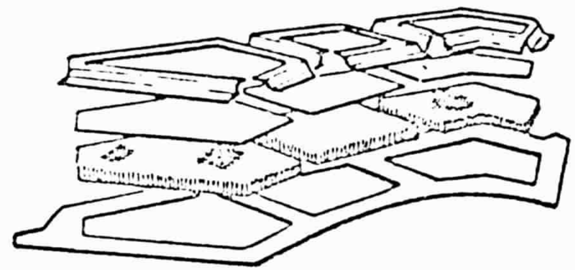
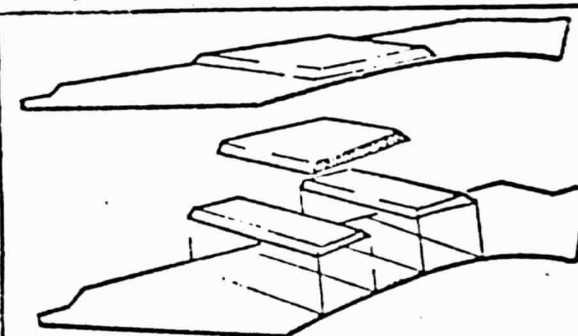
Costs

The cost of a finished product is greatly affected by the cost of the raw material, and advanced composites are no exception. Depending upon the tooling, number of items to be made, complexity of part and other factors, it is possible for advanced composite parts to be made cheaper than other types of construction. Table XI contains data from a study done by McDonnell Douglas (Reference 9) which points this out. Although this table shows the composite raw material cost to be greater than fabrication labor, it can usually be shown that labor costs are a higher percentage of finished part cost than raw material.

Figure 3, taken from Reference 11, shows how the future costs of composite raw materials should continue to drop. There are several reasons why the cost reductions are imminent: (1) As production of fibers increase it is almost certain that fibers and, therefore, prepreg costs will come down dramatically; (2) It is being found that some reinforcing fibers actually perform better in large diameters which , usually, are less expensive to produce; (3) Efforts are going forward to produce graphite fibers from a pitch precursor. Starting with a petroleum by-product, producers are talking, hopefully, of ultimately arriving at a $5 per pound price for graphite in just a few years.

TABLE XI

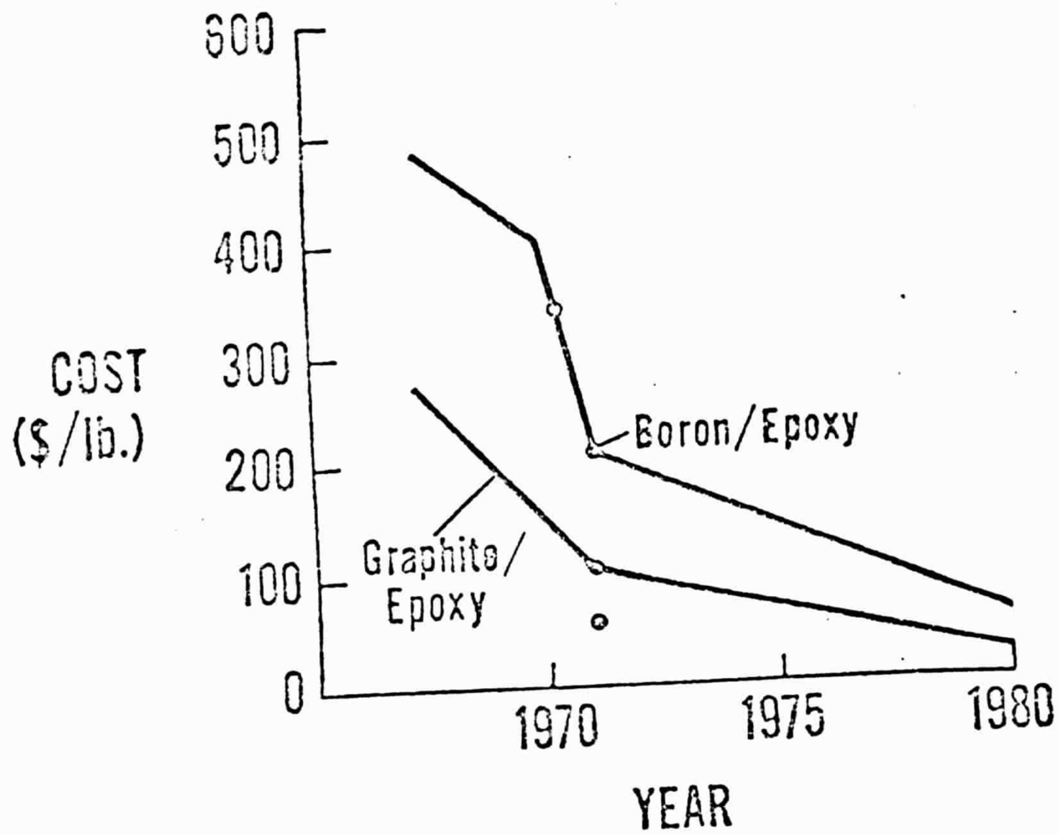
COST COMPARISON FOR METALLIC AND COMPOSITE STRUCTURE

	 ALUMINUM SANDWICH 12% Rejection Typical	 GRAPHITE SANDWICH 2.5% Rejection Actual
Number of Operations	103	19
Number of Hours	31.60	7.6
Labor Costs @$13.00/hr	410.80	98.80
Material Cost	12.50	258.00
TOTAL	$423.30	$356.80
TOTAL NUMBER MANUFACTURED		200
TOTAL NUMBER ORDERED		1000

Reference No. 9

FIGURE 3

PROJECTED RAW MATERIAL COSTS



Reference No. 11

The cost of tungsten, which is the conventional substrate upon which boron is vapor deposited, continues to show a cost increase trend. However, carbon monofilament, produced from coal tar pitch, has been shown to be suitable for replacing the tungsten at a cost reduction of 60 to 70% per pound of boron produced. In addition to the material cost advantage, the manufacturer, according to Reference 12, has determined that the carbon filament will run faster than the tungsten wire in their boron converters which could also result in a reduction in labor cost and an approximately 40% decrease in their capital equipment depreciation cost. Partial production of this product is expected in 1978, with estimated overall cost reduction of 30% for boron/epoxy prepreg made by this method.

A further indication that labor costs are becoming a significant portion of finished part cost is the emphasis being placed on development of low cost fabrication techniques. Specifics of several studies along this line are presented in the following subsection.

Advanced Fabrication Technology

Manufacturing technology has considerable potential for reducing the end product cost of composites and thus, considerable research is directed in this area. The use of rubber tooling, for example, can cut the cost of forming laminate composites by increasing processing speed and eliminating the need for expensive autoclaves. Silicone rubber, tailored to provide a controlled expansion upon heating, is used in place of conventional female dies. Expansion upon heating forms reinforced laminates over a male die at much higher pressures than can be applied in autoclaves. For example, Reference 3 indicates that in an Air Force Materials Laboratory program, pressures on the order of 500 psi were developed to form both boron/epoxy and graphite/epoxy laminates. At NASA (Langley), pressures of 600-1000 psi are routinely developed to form graphite/polyimide laminates. Unlike matched metal dies,the rubber can apply side pressures on flanges and other design details. For epoxy matrix composites, which are cured at about 350°F, the rubber is reusable many times. The silicone rubber degrades sooner when forming polyimide matrix composites because of the higher cure temperatures used.

NASA (Langley) is evaluating the hot forming characteristics of fully imidized (cured) graphite/polyimide laminates formed to shape with integral stiffeners at elevated temperatures and moderate pressures with simultaneous post curing without cross-linking. Such thermoforming overcomes the problem of adhesive bonding detail parts to form integral structures and may prove to be an efficient means of processing polyimide composites. At present, the work reported by John Vaccari in Reference 3 indicates that the resulting mechanical properties are low.

Cost reduction is also the principal aim of several AFML programs to develop better techniques for making improved preimpregnated tapes and tape-laying machines. Also in development is a new technique for producing multi-ply, oriented broadgoods (nonwoven fabrics) in widths up to 8 ft. In addition, radio frequency and internal resistance heating techniques are being explored for curing composites. The latter program promises a sixfold reduction in curing time over conventional autoclave methods. To reduce the cost of boron fiber, AFML is establishing production techniques for depositing the metal on carbon monofilaments rather than on tungsten wire substrates. Work on this area is being done by AVCO Corporation under Air Force Contract F33615-71-C-1334.

CONCLUSIONS

1. Shape limitations for advanced composite structures are similar to the limitations for fiberglass-reinforced structures.

2. Advanced composite raw materials are available in many standard and special application forms.

3. Both boron and graphite/polyimide composite structural parts can be successfully fabricated by the vacuum bag/autoclave method.

4. Unless high quantities or production rates are required vacuum bag/autoclave fabrication methods are today the most feasible methods of production because of:

   (a) Ease of obtaining thickness variation
   (b) Ease of laying-up desired fiber orientation
   (c) Ease of formation into large and/or complex shapes
   (d) Adaptability to integral design and fabrication

5. The curing pressure and temperature requirements for epoxy and polyimide matrix composites can be generally stated as 100 psi and 350°F. Polyimide composites, however, require post curing to temperatures up to 700°F.

6. Nondestructive testing methods have been proven useful in detecting defects and anomalies in composite structures.

7. Diamond type cutting tools are required for machining boron and graphite resin matrix composites. Graphite/resin composites however can be drilled with carbide tools.

8. Cost reduction for materials and fabrication is currently the principal aim of industry.

REFERENCES

1. "Prototype Rigid Polyimide Components" Final Report SD75-SH-0068, Rockwell International - Space Division, March 1975

2. "Development of Design Data for Graphite Reinforced Epoxy and Polyimide Composites" Contract NAS 8-26198, Report No. GDC-DBG70-005, General Dynamics Convair Aerospace Division, May 1974

3. "What's Ahead in Aircraft and Aerospace Materials" John A. Vaccari, Special Report from Materials Engineering, November 1974

4. "Development of Lightweight Graphite/Polyimide Honeycomb, Phase I - Materials Selection" Contract NAS 9-11368, Report H400-12-1-8, Hercules Incorporated Systems Group, Bacchus Works, Magna, Utah, July 1971

5. "Fabrication and Acceptance of Boron/Epoxy Structural Skins" Process Specification No. 14130, McDonnell Douglas Corp. St. Louis, September 1971

6. "Fabrication and Acceptance of Solid Laminates Graphite/Epoxy Parts" Process Specification No. 14242, McDonnell Douglas Corp., St. Louis, June 1975

7. "Develop Fabrication/Processing Techniques for High Temperature Advanced Composites for Use in Aircraft Structures" AFML TR-72-91 McDonnell Douglas Corp. St. Louis, July 1972

8. "Advanced Fiber-Resin Composites" Machine Design, K. R. Berg, F. J. Fillippi, Whittaker Corp., April 1, 1971

9. "Air Force Recent Progress in Composites" George P. Peterson, Conference Presentation from Advanced Composites, An Assessment of the Future, 11-12 June, 1975

10. "Advanced Composites - An Assessment of the Future", Leonard A. Harris, Astronautics and Aeronautics, March 1976

11. "Advanced Composite Materials" Presentation by General Dynamics, Convair Aerospace Division at Review of Advanced Composite Materials, Johnson Space Flight Center, March 1974

12. "Evaluation of Boron/Epoxy Prepreg Produced from Boron Deposited on Carbon Fibers", Report No. MDC A3492, R. E. Walter, McDonnell Aircraft Company, 8 October 1975

APPENDIX A

TYPICAL FABRICATION METHODS

INTRODUCTION

This appendix contains processes for fabricating four typical, but different composite materials. The epoxy processes chosen are based on currently employed procedures for the most popular materials in use. The high temperature material processes cover a typical addition type polyimide and a typical condensation type polyimide.

The two epoxy matrix fabrication details (graphite and boron) are condensed from advanced procedures used by McDonnell Douglas Corporation. The first graphite/polyimide process specification was condensed from a General Dynamics Convair Aerospace Division Report and the second graphite/polyimide process specification was condensed from a McDonnell Douglas Corporation report. The four different processes are given as examples of fabrication procedures used for making composite parts.

The reason for including example procedures is to allow Design Engineers and other affected disciplines to familiarize themselves with procedures and problems entailed in manufacturing composite structures. Without knowledge of detail fabrication procedures, a component cannot be designed cost effectively.

FABRICATION OF GRAPHITE/EPOXY STRUCTURES

1.0 PREPARATION OF CURING TOOLS

Coat tooling molds as follows:

Clean the tool surfaces with an MEK dampened cheesecloth and apply a mold release compound to the contact surfaces in accordance with the manufacturer's instructions. Apply a thin coat of release coating to the glossy finished mold and bake the mold for one hour at 380 ± 10°F (mold temperature). Cool the mold to room temperature, then buff the mold surface to a smooth, non-sticky surface. This procedure is performed only once and need not be repeated for the duration of mold usage. Before each lay-up operation, spray the surface of the mold with release coating with the mold surface at room temperature.

2.0 PREPARATION FOR COMPOSITE LAY-UP

Templates may be used to facilitate prepreg lay-up. Fabricate templates to the dimensions specified on the applicable engineering drawing.

Keep the prepreg material in the plastic bag a minimum of 2.0 hours after removal from the freezer. Orient the prepreg tape and/or broad-goods in the designated direction and cut to the dimensions of the templates or as specified on the engineering drawing.

3.0 LAY-UP AND BAGGING PROCEDURE

Use cork or silicone rubber dams around the composite lay-up unless a metal dam is an integral part of the lay-up tool. Use metal dams the same thickness +0.020-0.000 inch as the thickness of the expected cured laminate. The cut edges of the prepreg shall be within 0.10 inch of the inside of the dam.

Apply each individual ply of prepreg to the tool, using a Mylar covered Teflon squeegee and (if required) hot air gun, tacking table or hot iron to momentarily soften the resin while working it with the squeegee. Using prepreg plies collate the composite laminate as specified on the applicable engineering drawing or applicable document. A peel ply is required if the laminate surface is ultimately going to be bonded. Add one layer of porous release film over the laminate surface and within the area formed by the dams.

4.0 LEAK CHECKING

Pull a 27-29 inch mercury vacuum source and record the vacuum reading. Take a pressure reading 2 minutes after isolation of the system. The maximum allowable leakage is 0.5 inches of mercury per minute.

Maintain at least 10 inches of mercury vacuum on the bagged lay-up and place in the autoclave. Connect the required plumbing (thermocouples, and vacuum and static lines) and repeat the leak check of the previous paragraph. At the conclusion of the second leak check close the autoclave door while maintaining 27-29 inches of mercury vacuum and proceed with the cure schedule.

5.0 CURE SCHEDULE

Pressurize the autoclave to 10 psig and hold 27-29 inches of mercury vacuum on the bagged layup. Raise the part temperature to 350 ± 10°F in 60 to 140 minutes while holding 10 psig autoclave pressure and 20 inches minimum of mercury vacuum. Set the autoclave free air temperature no more than 150°F above the lowest reading part thermocouple during the heat up. During the part heat up, when any one of the part thermocouples reaches 140°F, pressurize the autoclave to 125 psig. Maintain the 20 inches minimum of mercury vacuum.

Hold at 350 ± 10°F for 120 ± 10 minutes under 125 ± 5 psig autoclave pressure and 20 inches minimum of mercury vacuum.

Cool to 150°F or less using a cool down rate of 2-5°F/min. while maintaining 8 psig minimum autoclave pressure and 24 inches minimum of mercury vacuum.

6.0 POST CURE SCHEDULE

Do not post cure parts which are subsequently bonded in a honeycomb sandwich assembly. Do not post cure the associated process control laminates. Post cure other graphite/epoxy parts and their process control laminates as follows:

(a) If the part has just completed an autoclave cure operation, it may be autoclave post cured or be removed from the autoclave at 150°F or less and placed in the oven.

(b) Place parts to be post cured in the bonding tool or other suitable support fixture. Vacuum bag restraint is not required during post cure.

(c) Connect the thermocouples (minimum of four) to the part and the process control laminates.

(d) Heat the assembly to 350 $\pm$ 10°F at 1.5 - 5.0°F/min. and post cure 4 $\pm$ 1/2 hours at 350 $\pm$ 10°F. Cool to 220°F maximum in a minimum of 30 minutes prior to removal from the oven.

FABRICATION OF BORON/EPOXY STRUCTURES

1.0 PREPARATION FOR COMPOSITE LAY-UP

Determine the dimensions and orientation of each ply in the composite laminate from the Engineering drawing and fabricate Mylar templates accordingly.

Cut one layer of 120 dry glass cloth for every five plys of pre-preg material. When peel ply is required on the laminate surface(s), each peel ply layer will substitute for one layer of 120 dry glass bleeder cloth.

2.0 LAY-UP AND BAGGING PROCEDURE

Keep the pre-preg material in the plastic bag a minimum of two hours after removal from the freezer.

If Mylar templates are used, orient the pre-preg tape and/or broadgoods in the designated direction and cut to the dimensions of the Mylar templates. Place the steel dams on top of the release film with the outside edge of the dam directly over the outside edge of the release material. Tape the dams to the release material. Coat the inside of the dams with release material.

Lay-up the composite laminate by collating the pre-preg plys on the fixture from the Mylar templates. Lay in place, being most careful about correct location of the edge that terminates inside the lay-up. If a peel ply is required on the tool side of the laminate, lay-up the tool side ply of pre-preg on the peel ply and then transfer this lay-up combination to the tooling. Attach thermocouples to the lay-up.

Place the top release material over the completed lay-up and attach to the steel dams. Place the correct number of 120 glass cloth layers over the release material. Place one layer of Mylar Type A or nylon film on top of the 120 glass cloth and extend to the outside edges of the dam and/or splice skin. Make two 2 inch long slits at 90° to each other in each corner of the film. Attach the film to the tool with Tape. Place 2-5 layers of Style 1000 dry glass cloth over the top of the entire lay-up.

Place a single layer of Mylar Type A or nylon film on top of the entire lay-up and extend several inches beyond the dams and/or splice skin assembly. The vacuum and static lines shall be located adjacent to the lay-ups and inside the sealed edge of the vacuum bag. Seal the vacuum bag to the fixture with Sealing Tape.

See Figure 1 for typical laminate lay-up cross section.

3.0 LEAK CHECKING

Pull a 20-29 inch mercury vacuum on the bagged lay-up and close off the vacuum source.

Take a pressure reading 2 minutes after isolation of the system. The maximum allowable leakage rate is 0.5 inches of mercury per minute. Maintain at least 10 inches of mercury vacuum on the bagged lay-up and place in the autoclave. Connect the required plumbing (thermocouples, and vacuum and static lines) and repeat the leak check of the previous paragraph. At the conclusion of the second leak check, close the autoclave door while maintaining 20-29 inches of mercury vacuum on the bagged lay-up. Apply autoclave pressure. When the positive applied pressure on the outside of the vacuum bag exceeds 15 psig close off the vacuum bag to the atmosphere.

4.0 CURE SCHEDULE

Pressurize the autoclave to 85 ± 5 psig with nitrogen. When achieved, heat the lay-up to 350°F at a rate of 1.5-4.0°F/min. Hold at 350 ± 10°F for 2 hours. If the part will be post cured in an oven, cool to at least 150°F, while maintaining at least 7.5 psig, in a minimum of 2 hours. Record the autoclave temperature and pressure throughout the cure cycle. Also, record pressure build-up in the vacuum bag at least once every fifteen minutes.

5.0 POST CURING

Post curing of the bagged composite part may be performed in either the autoclave or an oven.

5.1 Autoclave - At the conclusion of the regular cure schedule, raise the part temperature from 350 ± 10°F to 370 ± 10°F while maintaining autoclave pressure. Post cure for 4 hours at 370 ± 10°F and then cool to at least 150°F in a minimum of 30 minutes. During the post cure (4 hours at 370 ± 10°F) and subsequent cooling down period, it is not necessary to continuously recharge the autoclave with fresh nitrogen. A positive pressure of at least 7.5 psig nitrogen, however, must be maintained on the part at all times.

5.2 Oven

Remove the bagged lay-up from the autoclave. Make certain the part temperature does not exceed 150°F at time of removal. Place the bagged lay-up in an oven and connect the thermocouples already attached to the lay-up. Subsequently, pull a minimum of

15 inches mercury vacuum on the lay-up. The oven may be at temperature not to exceed 150°F during this operation. Heat the lay-up to 370 ± 10°F at a rate of 1/2 to 10°F/minute. Maintain this temperature for 4 hours after the coolest thermocouple has reached this temperature. Cool under vacuum to at least 150°F in a minimum of 30 minutes.

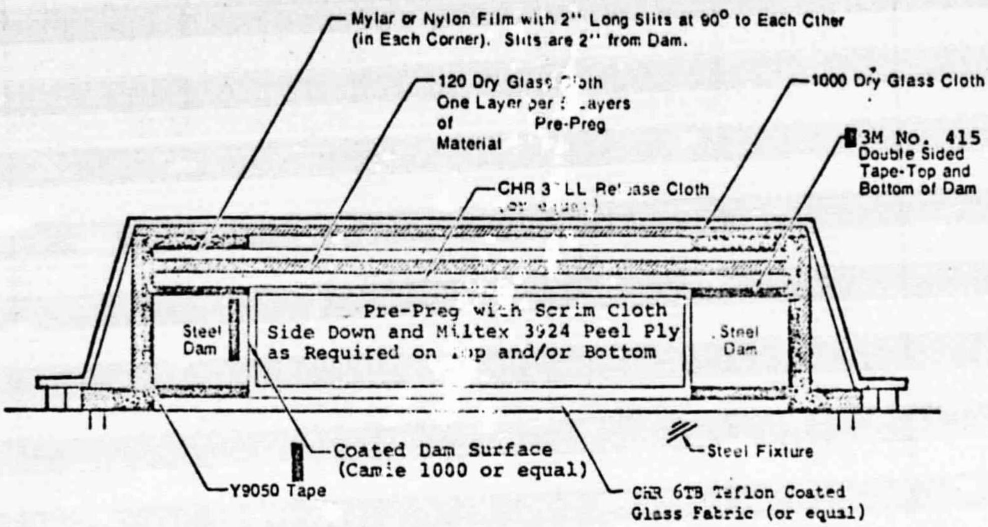


FIGURE 1 - LAY-UP CROSS SECTION OF LAMINATE WITHOUT EDGE MEMBERS

FABRICATION OF GRAPHITE/POLYIMIDE (HT-S/710) STRUCTURES

The following process, is completely adaptable to vacuum-bag, press, or autoclave-curing techniques simply by modification of the pressure. It is insensitive to heating rates up to 10°F/minute.

1.0 PREPARATION OF CURING TOOLS

Clean the tool surfaces with an MEK dampened cheese cloth and apply a mold release compound to the contact surfaces in accordance with the manufacturer's instructions. Apply a thin coat of release coating to the glossy finished mold and bake the mold for one hour at 350°F. Cool the mold to room temperature, then buff the mold surface to a smooth, non-sticky surface.

Before each lay-up operation, lightly wipe the mold surface with release coating with the mold surface at least at 80°F.

2.0 PREPARATION FOR COMPOSITE LAY-UP

Identify the orientation of each ply in the composite shape by marking Mylar templates.

Keep the pre-preg material in the plastic bag a minimum of 2 hours after removal from the freezer. However, do not allow the pre-preg material to be exposed to temperatures over 0 ± 10°F unnecessarily. Record the cumulative time out of 0 ± 10°F storage for each pre-preg container.

3.0 LAY-UP AND BAGGING PROCEDURE

Lay up a separator cloth and the correct number of plies of 120 glass cloth bleeder on both sides of the graphite/polyimide layup. Seal top and bottom with suitable film material such as Mylar or Teflon. Vacuum bag the layup, apply full vacuum 760 mm (29 in.) Hg, and place the part in the autoclave. Heat the part to 175°F at a rate of 3 to 5°F/minute, apply 25 psi, hold 15 minutes, and cool to 75°F. Remove bleeder and initiate the cure cycle layup.

4.0 CURE CYCLE

Place 3 plies of 104 glass cloth on each side of the pre-compacted layup and complete the bagging and venting material as shown in Figure 2.

Apply full vacuum 29 in. Hg, heat to 175°F at a rate of 3 to 5°F/minute, hold 30 minutes, heat to 260°F at a rate of 3 to 5°F/minute, hold 25 minutes, apply 100 psi, heat to 350°F at a rate of 3 to 5°F/minute, hold 2 hours, cool under pressure to 175°F or lower at a rate no greater than 3°F/minute.

5.0 POSTCURE CYCLE

Place the part, unrestrained, in a room-temperature oven, heat to 350°F, and initiate the following postcure cycle:

2 hours at 350°F	2 hours at 550°F
2 hours at 400°F	2 hours at 600°F
2 hours at 450°F	4 hours at 650°F
2 hours at 500°F	8 hours at 700°F

Cool to 175°F or lower and remove from the oven.

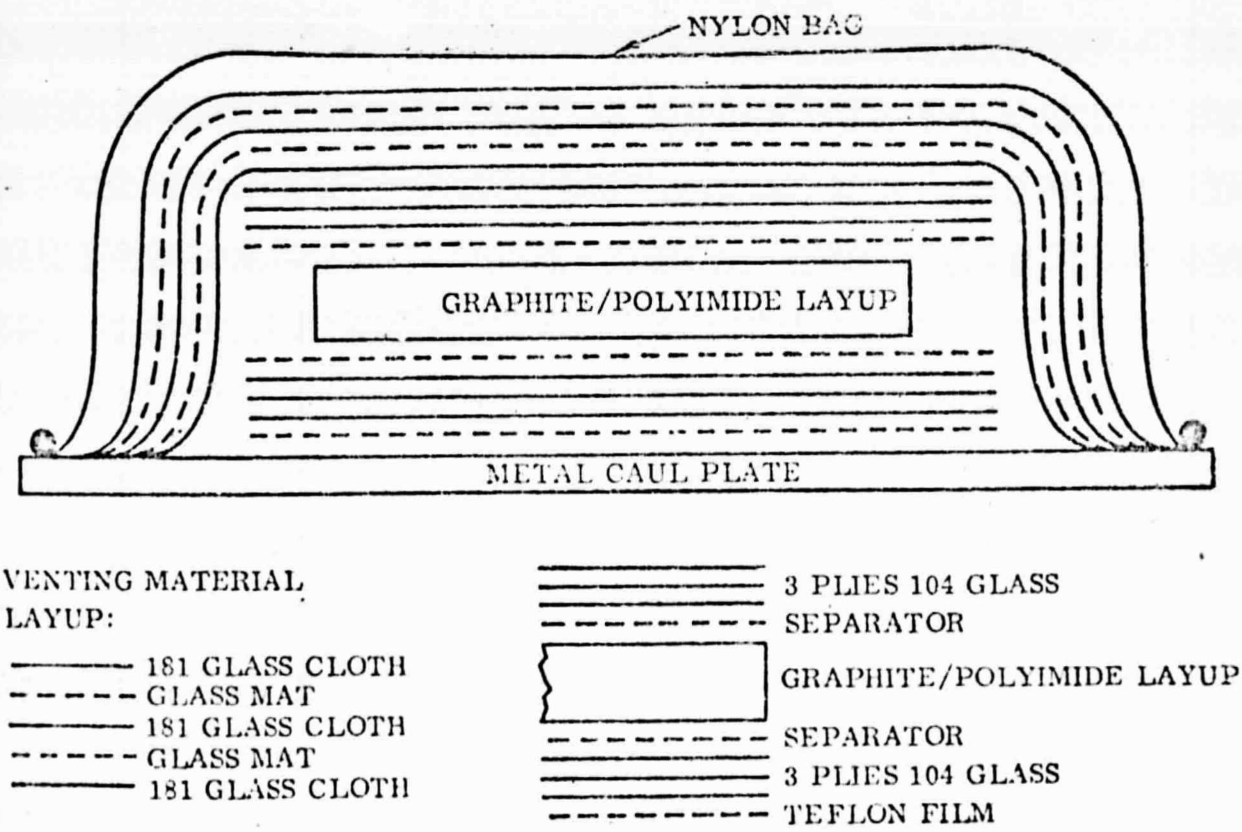


Figure 2. Schematic of Graphite/Polyimide Cure Layup

FABRICATION OF GRAPHITE/POLYIMIDE (PI-2501) STRUCTURE

DuPont's PI 2501 polyimide resin is a condensation polymer that is dissolved in 12-15 percent n-methyl pyrrolidone (NMP) solvent. The high boiling point of NMP (395°F), combined with the condensation (H_2O) byproduct of the polyimide resin during polymerization, characterizes this system as one requiring close process control. In addition, the polyimide resin, in the B-stage condition, is hygroscopic and requires stringent control during storage and processing. Therefore, it is apparent that extreme care and accurate fabrication techniques must be exercised. For this reason, a specific area for polyimide layup must be maintained under clean room conditions with temperature/humidity controls.

The need of resin content control to achieve the non-flammability of PI glass laminates requires a unique approach to the fabrication of laminated components. A technique that assures minimum resin content variability must be used.

1.0 TOOL PREPARATION

Clean the tool with an MEK dampended cheese cloth, and apply a mold release compound to the contact surfaces in accordance with the manufacturer's instructions. Apply a thin coat of release coating to the glossy finished mold and bake the mold for one hour at 350°F. Cool the mold to room temperature, then buff the mold surface to a smooth, non-sticky surface.

Before each lay-up operation, lightly wipe the mold surface with release coating with the mold surface at least at 80°F.

2.0 LAYUP

One ply of fluorocarbon parting fabric shall be placed on the platen or mold surface and allowed to overhang the layup by 1/2 inch, minimum. Eight plies of bleeder fabric shall be applied on the parting fabric. All edges shall be sealed with a fluorocarbon tape and pigtailed to the layup to provide uniform vacuum distribution. Orient the pre-preg broadgoods in the designated direction and cut to the dimensions of the Mylar templates if applicable.

3.0 CURE CYCLE

Apply an initial vacuum of 26 inches of mercury, minimum, as measured from the vacuum static line and insert assembly in autoclave. Pressurize autoclave to obtain a pressure differential of 45 $\pm$ 10 lb/in.^{2}. Heat the laminate to 295°F in 90-240 minutes while maintaining a pressure differential of 45 ± 10 lb/in.^{2}. When part temperature reaches 295°F, increase temperature and pressure differential at a gradual rate to 330°F and 85-100 lb/in.^{2} in a period of 55 ± 10 minutes. Hold at 340 ± 10°F and 85-100 lb/in.^{2} pressure differential for 180 minutes, minimum. Cool to 200°F under a pressure differential of 85-100 lb/in.^{2} Cool to 150°F or less under a pressure differential of at least 10 lb/in.^{2}

4.0 POST-CURE

Laminates shall be post-cured per the following cycle:

2 hours at 250 ± 10°F	2 hours at 500 ± 10°F
2 hours at 300 ± 10°F	8 hours at 600 ± 10°F
2 hours at 400 ± 10°F	Cool to 150°F or less in oven