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Development of Stitched / RTM Primary Structures for Transporl Aircraft

Arthur V. Hawley

McDonnell Douglas Aerospace -
Transport Aircraft
Long Beach Ca. 90846

Contract NAS1-18862

July 1993

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ABSTRACT

This report covers work accomplished in the Innovative Composite Aircraft Primary Structure (ICAPS) program, NASA Contract NAS1-18862. An account is given of the design criteria and philosophy that guides the development of composite primary components. Wing and fuselage components used as a baseline for development are described. The major thrust of the program is to achieve a major cost breakthrough through development of stitched dry preforms and resin transfer molding (RTM), and progress on these processes is reported. A full description is provided on the fabrication of stitched RTM wing panels. Test data are presented.

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FOREWORD

This report was prepared by the McDonnell Douglas Aerospace - Transport Aircraft Company, under NASA Contract NAS1-18862, "Innovative Composite Aircraft Primary Structure" (ICAPS), with the National Aeronautics and Space Administration. The contract is managed by the Langley Research Center (NASA-LaRC), Hampton, Virginia with Mr. Marvin B. Dow as the Technical Program Manager. The contract is part of the Aircraft Composite Technology (ACT) development being undertaken by NASA under the leadership of Dr. John G. Davis, Jr.

The identification of commercial products in this report does not constitute official endorsement of such products, either expressed or implied, by the National Aeronautics and Space Administration.

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ILLUSTRATIONS
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SECTION 1
INTRODUCTION

It has been widely recognized that cost is the major obstacle to the adoption of composite primary structural components for production commercial transport aircraft. Douglas (now McDonnell Douglas Aerospace - Transport Aircraft) identified the stitching of dry preforms and resin transfer molding (RTM), and automated tow placement (ATP), as having the potential to overcome this obstacle.

Stitching and RTM provide the focus for wing development in this program and the intention is to make a head-to-head comparison with a state-of-the-art (tough resin) in-house wing development effort. Essentially identical wing structural boxes are being developed in each program. A comparison will also be made between the cost of composite and metal wing box components. In the fuselage development effort, both the stitching/RTM approach and fiber placement have been investigated. A fuselage fabrication process will be selected on the basis of cost and performance.

Although RTM is not a new process, a considerable development program effort is being conducted to scale up the process to large components. Specially developed resin systems have been evaluated for this purpose. Stitching parameters also have been intensively researched with respect to processing, performance, and cost. Two specially developed stitching machines have been purchased to aid in the development of low-cost automated techniques.

Two methods of RTM are under development, using vacuum impregnation and pressure injection respectively. A variation, distinctly different from the normal RTM process, has been adopted to make the wing panels described in Section 7. In this method, known as Resin Film Infusion (RFI), the resin is cast as a solid film and placed in the tool beneath the preform. Successful fabrication of ATP fuselage panels has also been accomplished. Lessons have been learned and incorporated into the tooling, processing, and fabrication of stiffened wing test panels. A manufacturing cost model is being used to develop a cost data base and to provide design guidance. Further work performed under this contract will be described in subsequent reports.

SECTION 2
DESIGN CRITERIA AND PHILOSOPHY

The requirements specified in Federal Aviation Regulations, Part 25 (FAR 25) are used as the basis for design. Additional guidance is provided by FAA Advisory Circulars AC20-107A "Composite Aircraft Structure" and AC25.571-1A "Damage Tolerance and Fatigue Evaluation of Structure." The various steps necessary before receiving full FAA structural certification are complex as illustrated by the typical flow path diagram shown in Figure 1.

2.1 Damage Tolerance Rationale

All safety-critical structure, including primary wing and fuselage components, must meet FAA damage tolerance requirements. The manner in which it is validated that an adequate level of damage tolerance has been provided is particularly important, since this can play a significant role in influencing structural weight. The intent of the rules defined by FAR 25.571 is clear enough, and that is that no catastrophic structural failure shall occur throughout the operational life of the aircraft. Yet, the actual means for showing compliance with the rules must still be formulated by the aircraft manufacturer and submitted, in the Certification Plan, for approval by the FAA. A rationale for a damage tolerance certification approach is required which:

a) is simple enough to be practical,
b) is severe enough to satisfy the FAA,
c) ensures that the structure is repairable, and
d) reduces the cost of quality assurance and in-service inspection.

A successful damage tolerance approach has to:

1) identify structural components that are designed to be damage tolerant,
2) incorporate design features to assist in attaining a damage tolerant structure,
3) formulate an inspection plan to ensure detection of specified levels of damage,
4) define the extent of damage and the associated loading conditions, and
5) conduct structural tests and analyses to substantiate that the design objectives have been met.

Safety of damage-tolerant structure can be enhanced by providing multiple redundant load paths and by incorporating positive means of arresting the growth of damage in service. It must be emphasized that since damage is always assumed to be present, it is the allowable gross working stress level of the damaged structure, rather than the ultimate undamaged strength of the material, that drives the weight of the structural component.

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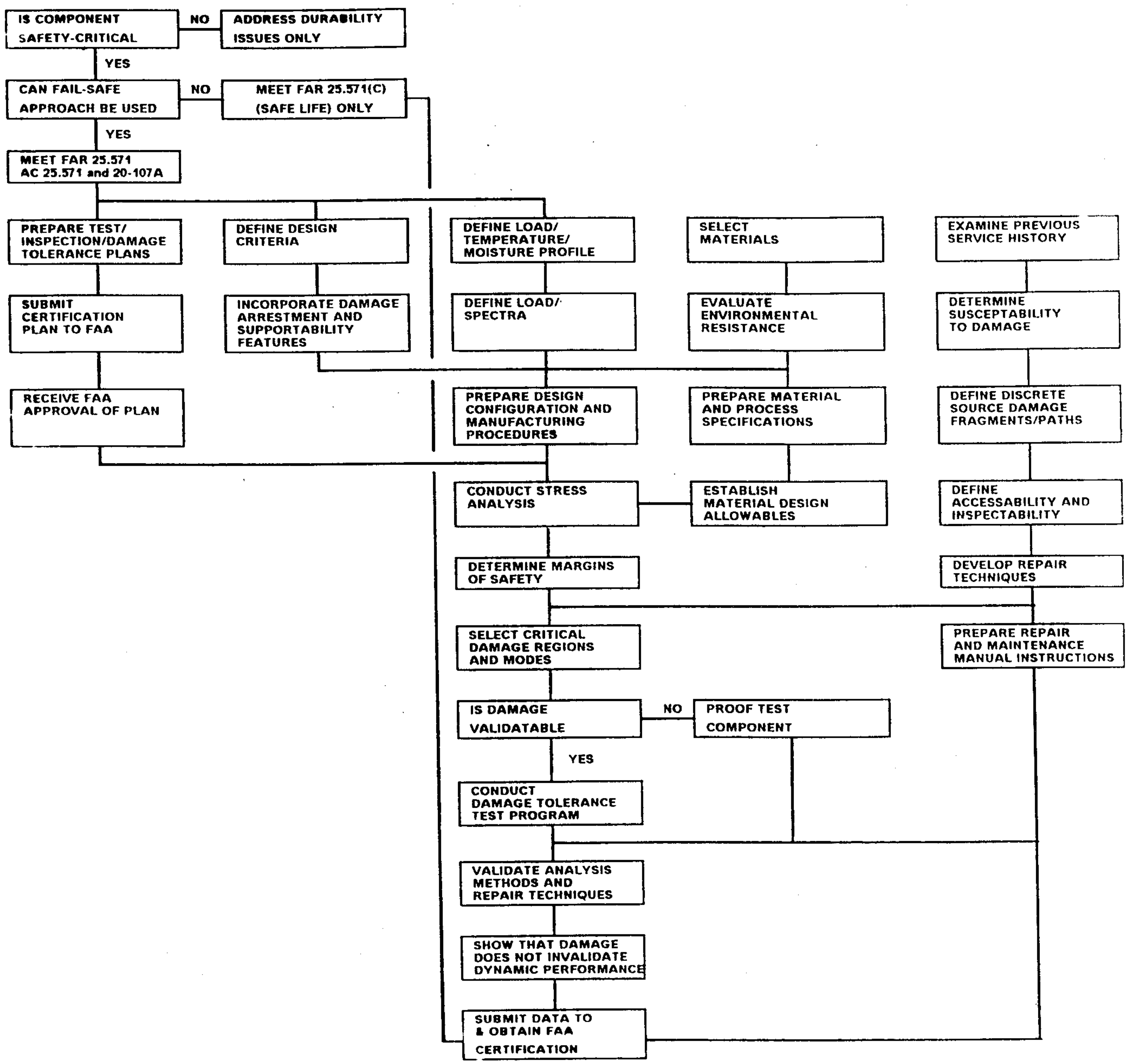


Figure 1. The Path to FAA Structural Certification

The two types of damage to be considered are manufacturing flaws and in-service damage. The presence and extent of damage must be determined through specified inspection procedures, and measured against defined accept/reject criteria (as defined, for example, in existing Douglas Process specifications). Damage rejected by the inspector may be "bought off" by the engineer if it is judged not to be critical, or it may be repaired to avoid scrapping the part.

Damage may arise from a variety of causes such as badly drilled holes, scratches, gouges, and impact. Impact damage is considered to be the most serious for structural elements loaded in compression (see Figure 2, extracted from References 1 and 2), because there can be delaminations around the impact site from which further growth can occur when compressive static or fatigue loading is applied. Note, however, that where delamination failure is suppressed (for example by stitching), then other types of damage may become more dominant and these should also be considered.

In tension, the concern is the catastrophic growth of through-the-thickness cracks. These usually originate at regions of high stress concentrations around defects, holes, and cutouts. Propagation of such damage in metal structures is normally slow enough to be monitored, at specified inspection intervals, and the aircraft is allowed to continue in service until the damage approaches a previously determined critical length. In composites, damage growth is usually so rapid that no reasonable inspection interval can be set. For this reason, it is normal practice to design to strain levels low enough for damage not to propagate. This is the "no growth" philosophy. However, growth is acceptable if a positive damage arrestment device can be shown to be effective. In this case, the design strain level can be increased to approach a value where the arrestment feature no longer prevents catastrophic propagation of damage.

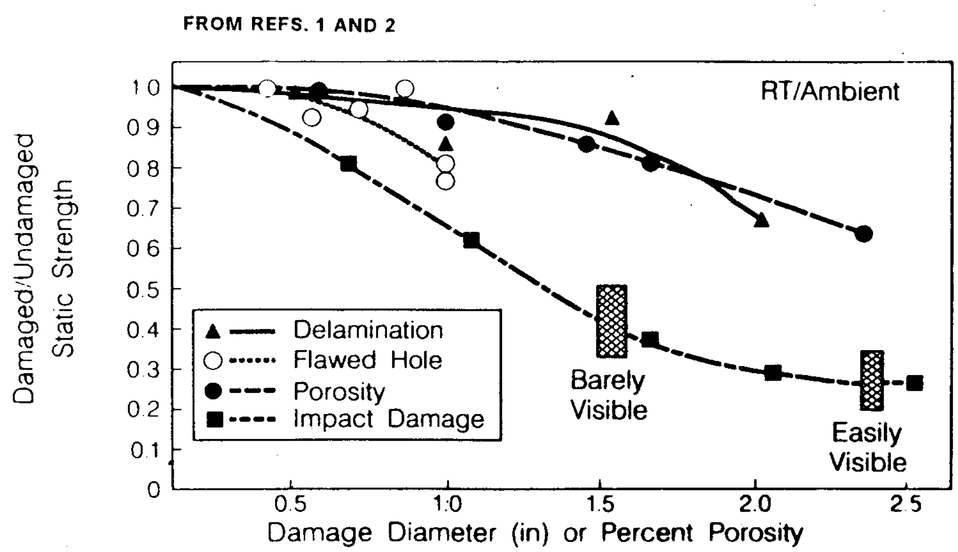


Figure 2. Degradation of Compression Strength

The allowable gross working stress level of structural components loaded in tension shall take account of all types of stress concentrations. Of these, the loaded fastener is considered to be the most serious, because the stress concentration due to the stress field running past the hole must be combined with the stress concentration due to bearing in the hole. For all structure where mechanical fasteners are required, either for permanent attachments or for repair, good load transfer behavior shall be ensured by selecting laminate patterns within the Douglas-recommended limits shown in Figure 3. However for other types of damage, where damage area is large, these can become the more critical defects and must also be considered in determining design allowable stress values.

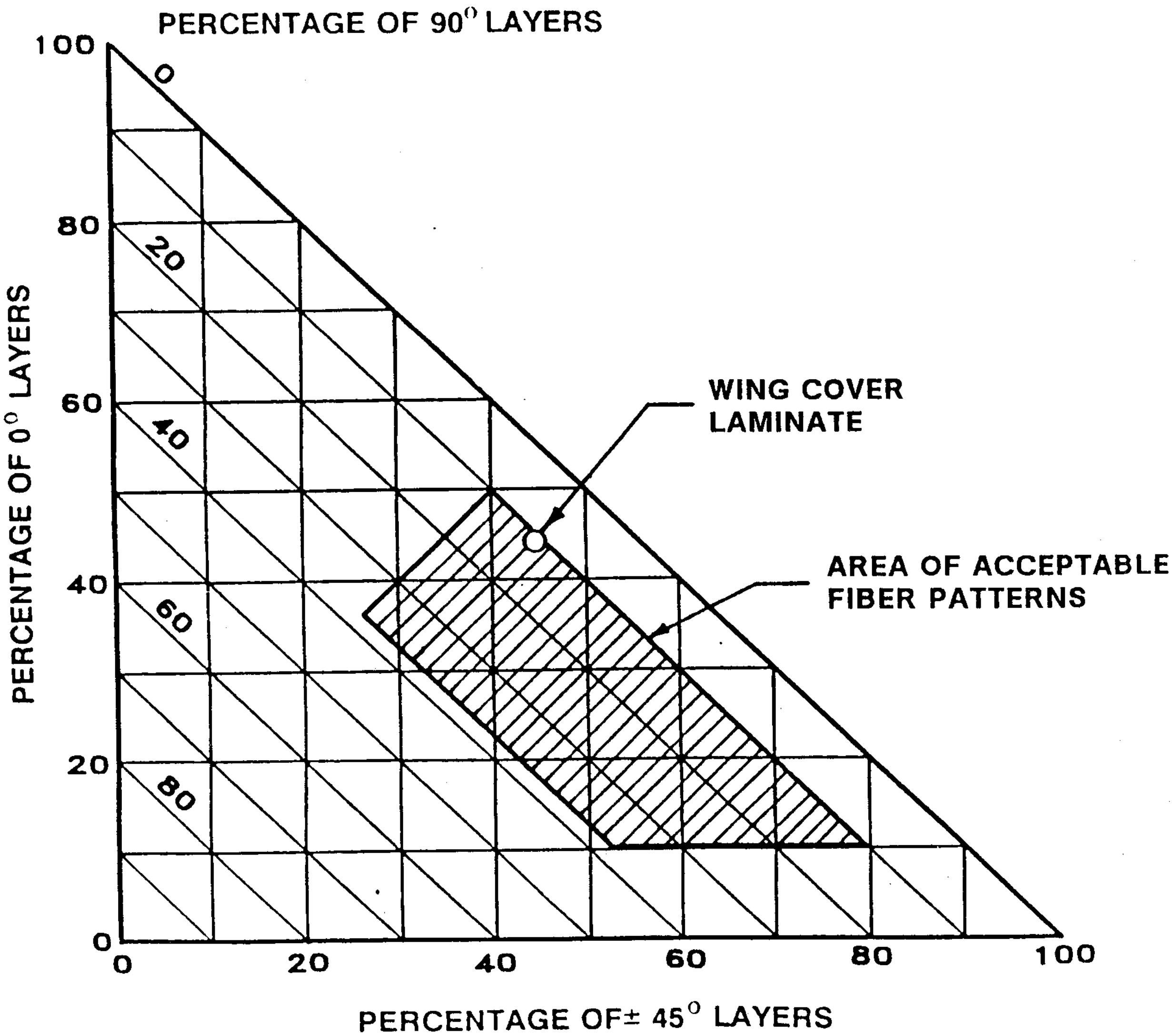


Figure 3. Recommended Laminate Pattern Range

2.2 Design Conditions

FAA requirements are not specific in defining types of damage, impact energies, or the impactor size and shape to be considered in design. They do, however, make a distinction between non-detectable, detectable, and discrete-source damage, and give a level of loads for each. The levels are illustrated, for symmetric flight conditions only, in the maneuver and gust envelopes shown in Figure 4.

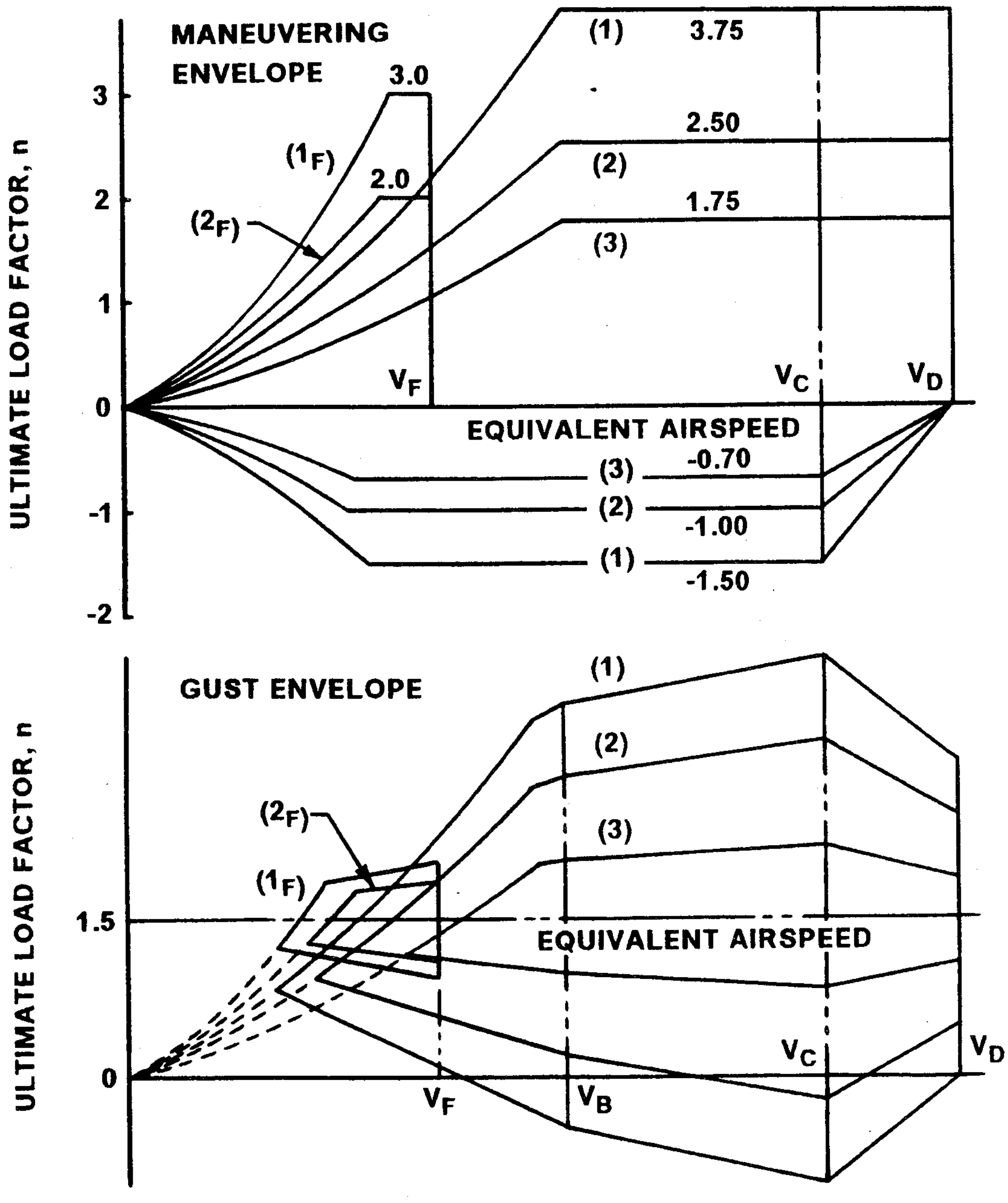
All safety-critical structure must be designed to meet the full range of FAA requirements (Level (1), Figure 4) while containing damage which may not be detected or repaired during the entire life of the aircraft. The detectability of damage will depend on its location and on the prescribed inspection procedures. For structure that receives no more than cursory pre-flight walk-around inspections, an arbitrary 0.10-inch indentation is proposed as the threshold of detectability.

Ironically, when damage is large enough to be detected, the extent and magnitude of load conditions is reduced (Level (2), Figure 4). The assumption here, of course, is that such damage will be repaired within a reasonable period of time and for this limited time of exposure, the probability of meeting high loads is reduced. After repair, the structure may continue in this condition for the life of the aircraft, and the full range of the requirements must be met.

All types of possible discrete source damage, such as engine fragments taking out one stringer and the two adjacent skin bays, shall be identified. The regulations allow, in these cases, that reduced load levels likely to be met while getting the aircraft back on the ground be used as an ultimate condition [FAR 25.571(e)]. Following the recommendations of FAA Advisory Circular AC 25.571-1A, the load levels are selected to be not less than 70 percent limit flight maneuver loads and, separately, 40 percent of the limit gust velocity (Level (3), Figure 4).

Discrete-source damage is regarded as an unlikely emergency event. Repair of such damage may impose a severe penalty on the allowable gross working stress levels, and hence the weight, of the structure. Such a repair would be conducted by the manufacturer, or at least with the manufacturer's approval. If it is determined that the residual strength, after repair, is lower than that for which structural approval was obtained, it might be necessary to scrap or re-skin the part. While this is a serious cost consideration it might, for the types of damage that are extremely rare, be preferable to carrying a weight penalty on a whole fleet of aircraft.

The fourth pair of flight envelopes shown in Figure 4 are for flaps-down conditions ($(1)_F$, $(2)_F$). These again illustrate the principle that design conditions can be reduced because of the lesser probability of meeting high loads during the limited exposure time. They are also of interest because they are associated with the high structural temperatures that exist immediately after takeoff. In fact, the aircraft cools so rapidly once it is in the air, that by the time the flaps are retracted, high temperature will not be a significant design factor, at least for subsonic aircraft. It is evident that degraded properties at temperature



(1) Nondetectable Damage Envelope
(2) Detectable Damage Envelope
(3) Discrete - Source Damage Envelope
Subscript F Denotes Flaps Down
V_B = Design Speed for Maximum Gust Intensity
V_C = Design Cruising Speed
V_D = Design Dive Speed
V_F = Design Flap Speed
n = Acceleration Factor (Multiple 1.0g)

Figure 4. Ultimate Symmetric Flight Envelopes

(such as for hot/wet compression) will not result in a serious weight penalty for subsonic structures unless the drop in properties exceeds 20-percent for temperatures in the 165°F range.

Further issues associated with impact damage are illustrated in Figure 5. For a given laminate thickness, impact energy can be increased until the threshold of detectability is reached. A further increase in impact energy will eventually result in penetration, requiring immediate repair. Discrete-source damage is in general large area damage which, if not obvious to the pilot in flight, will be detected before the next flight. Also plotted are different levels of impact, the lowest being large-area impacts such as might be the case with hail. While these are small in magnitude (say 6 ft.lb) damage from closely-spaced impacts can coalesce and form large-area damage. This type of damage can be serious for thin structure for which a special inspection should be specified after each such event.

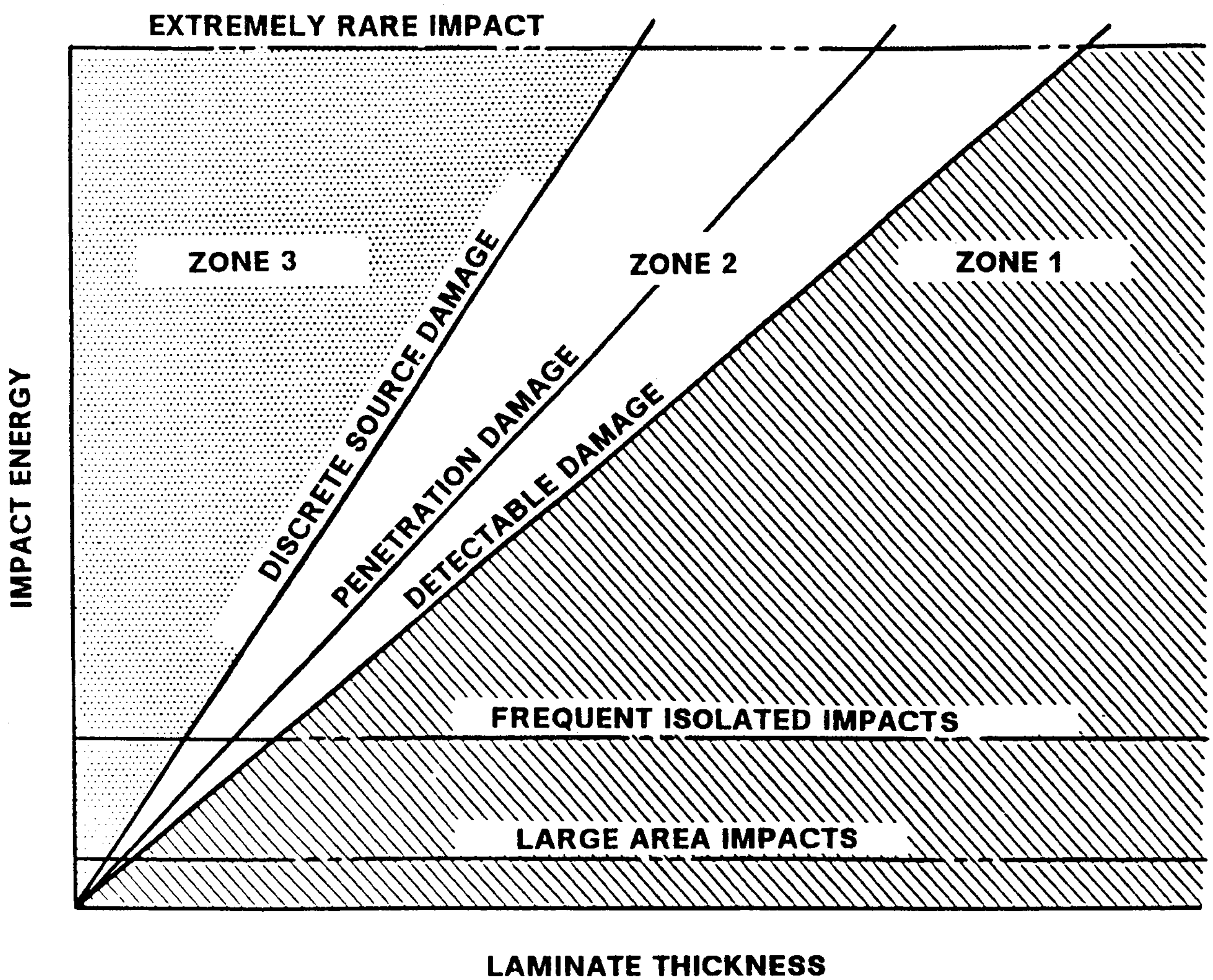


Figure 5. Impact Energy Levels

As damage impact size increases (to say 20 ft.lb for general exterior exposed structure) it is reasonable to suppose that such damage can exist at multiple discrete sites in the same structure. In areas especially prone to damage, such as around loading doors, special inspection procedures should be required. Damage from ground equipment, where impact energies can be measured in tens of thousands of ft.lb, is likely to be obvious or should be reported so that it can be assessed and corrected before the next flight, as necessary.

When impact energy reaches a level that can be regarded as an extremely unlikely event, that value can be used as a design cutoff. It is proposed that 100 ft.lb be used as a maximum for general exterior exposed surfaces and 20 ft.lb be used elsewhere. Impacts larger than these are covered by the requirements for discrete-source damage. These magnitudes of damage need only be considered at a single impact site within a given component.

2.3 Repair Rationale

The determination of whether the damage is to be repaired or not will depend on the observed visibility of the damage and the level of inspection to be expected in service. For example, a thick wing skin impacted by 100 ft.lb of energy will not have significant visible damage on the exterior surface. Even if the impact results in delaminations on the interior surface, the inspection requirements in this case may not allow an examination of this surface for perhaps 12,000 hours of flight service. For such a circumstance it may not be unreasonable to apply the full FAA load requirements to ensure the safety of the aircraft. Any in-service damage resulting in a surface indentation less than one-tenth of an inch deep is defined as non-detectable damage, even though such damage may be subsequently detected and repaired. However, any damage that results in a leak path, in either a wing fuel tank or a fuselage pressure shell, may be obvious and will then require immediate repair action.

For thinner structure, procedures are in place for repair by adhesively bonded patches. However, the possibility exists, particularly for field repairs, that mechanically-fastened patches might be used. For thicker structure, adhesively bonded lap repairs cannot be used because there is a limit to the load that can be transferred through simple lap joints. Adhesively bonded scarf repairs can be designed for thick structure, but these are largely impractical because of the difficulty and the large amount of undamaged material that has to be removed to provide the necessary shallow scarf angle. The philosophy to be adopted must ensure that the structure will be fully supportable during its service life. For bolted structure, this philosophy can be simply stated as: "If at any time during the operational life of a structural component there is the possibility of a hole being drilled for installation of a mechanical fastener, for repairs or for any other reason, then that structure shall be designed to accommodate the stress concentrations associated with loaded fastener holes." It is assumed that this possibility exists throughout the entire primary structure.

The philosophy really implies that the localized pad-ups associated with fixed joints and attachments have to be extended over the entire structural part. However the payoff is high since the repairability of the component in service has been assured.

It should be noted that it is not necessary for the repair to restore the strength of the undamaged structure as long as the repaired component can carry the full range of design loads. This requirement generally imposes a weight penalty by reducing the allowable design strain values. Where the penalty due to the repair is greater than the penalty due to the damage being repaired, consideration may be given to the use of soft (low modulus) repair patches, or cosmetic repair only.

2.4 Inspection and Quality Assurance

Inspection costs can be a large part of the total fabrication cost of a part, and inspection requirements should not be unnecessarily severe. A part which is designed to be tolerant of damage is also going to be tolerant of all sorts of minor flaws. Since these flaws add no additional penalty or threat to the structure, there is little point in making heroic attempts to locate and characterize the damage. This is particularly relevant to airline day-by-day usage, when no more than cursory walk-around visual inspections will be made.

Once the first few parts have been thoroughly inspected, and process control is established in series production, it should be enough to conduct a visual inspection only. This will detect levels of damage which are rejectable or which can be corrected by repair action using validated repair procedures. Similarly, for in-service inspections, and when there are no special concerns, a visual inspection should be all that is required. In other words, the burden of proof has been assumed in the design and test validation process, and can be largely removed from the manufacturing and airline operational phase.

This simplified inspection approach shall be supplemented by special requirements for critical regions of the structure. Specified additional inspections shall apply both through the manufacturing phase and during regular maintenance checks. The airline inspection plan shall require a closer look at the exterior surfaces of the structure and allow inspection of the interior by removal of selected access panels and interior trim items where necessary, at specified major checks.

2.5 Validation of Structural Integrity

The integrity of a structural component must be validated either by analysis or by test. The FAA requires that, in the absence of experience with similar designs, structural development tests on components, subcomponents, and elements should be performed. Structural analysis may be used to demonstrate compliance with strength and deformation requirements only if the structure conforms to those for which experience has shown this method to be reliable.

It is still necessary to develop reliable analysis methods, both for design sizing and for the prediction of structural behavior under load. With the introduction of new composite features such as stitching and resin transfer molding, analysis methods must be developed and validated by test. A test program must be formulated to include the characterization of basic material properties, bolt transfer behavior, the effect of structural discontinuities and damage, and to investigate certain novel features of the design.

Typical of the analysis methods, required to implement production readiness for ICAPS technology, is the prediction of structural behavior of a damaged stitched component loaded in compression. This is an example where the complexity and variables associated with stitching make it extremely difficult to derive a reliable, rigorous, purely analytical method. For this reason it will probably be necessary to rely on a semi-empirical step-by-step approach.

SECTION 3
WING DEVELOPMENT

This section describes the baseline aircraft and wing component used for this contract, the wing structural configuration and the loads definition.

3.1 Baseline Aircraft

The development of composite primary wing structure has been under study at Douglas since 1975. Previous contractual work for NASA is described in (References 3 through 10). In 1984, Douglas began a company study to provide technology and production readiness for the proposed MD-94X advanced technology transport aircraft. One of the configurations under evaluation for this aircraft (Figure 6) was originally selected as the baseline for the present composite wing contract. The baseline was later changed to an aircraft with aft-mounted engines (Figure 7) since at that time it was believed that the airlines would favor the propfan type of engine. Apart from the mounting of the engines, the wings of both aircraft were essentially similar, and are representative of the whole family of advanced commercial transport aircraft with high aspect ratio wings.

The slenderness of the structural box is illustrated in Figure 8. Locations of spars, fuel bulkheads, and ribs are largely dictated by the need to accommodate the landing gear, the high-lift and control surfaces, and their operating systems. Large access panels are provided for servicing the inside of the fuel tanks. Major structural breaks at the side-of-body and aerobreak stations have been assumed in the study, to cover the possibility that these might be necessary on a future aircraft.

3.2 Structural and Test Loads

At the start of the design, an in-house computer program (D9EZ) was used to derive overall shear, bending moment, and torque data for the wing. Table I shows the load cases that were considered and also indicates how limit and ultimate factors are interpreted in accordance with FAR 25.571 and the design conditions specified in Section 2.2 of this report. Structural design conditions are summarized in Table II.

Limit loads are shown at three stations on the wing for twenty-one significant flight cases in Table III. Maximum values of shear, bending moment, and positive and negative torque are shown "boxed" at each station. Two test conditions have been selected for the in-house and ICAPS inner wing ground test units. A positive torque condition combines an envelope of worst conditions, which is slightly conservative. Condition 10011 is selected as the negative torque case.

For simplicity, it was decided to use only one load application point per test condition as shown in Figure 9. This means that only one spanwise wing station will have the correct match to the shear, moment and torque values provided by the external loads program. Wing Station 141 was chosen for this station since it lies in the center of the test box. This approach gives a close match with the true bending distribution, but shears and torques are higher than actuals at the outboard end of the box, and lower at the inboard end. This discrepancy will not seriously affect

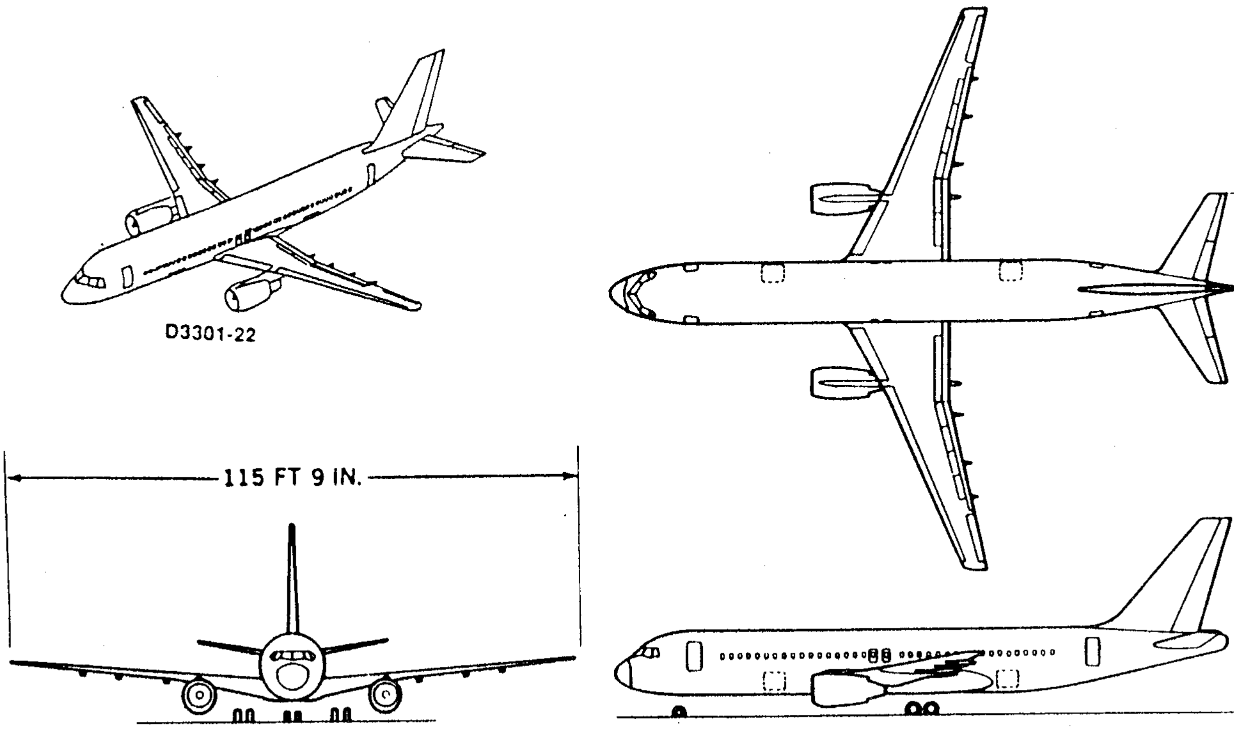


Figure 6. Original Baseline Aircraft for Wing Study

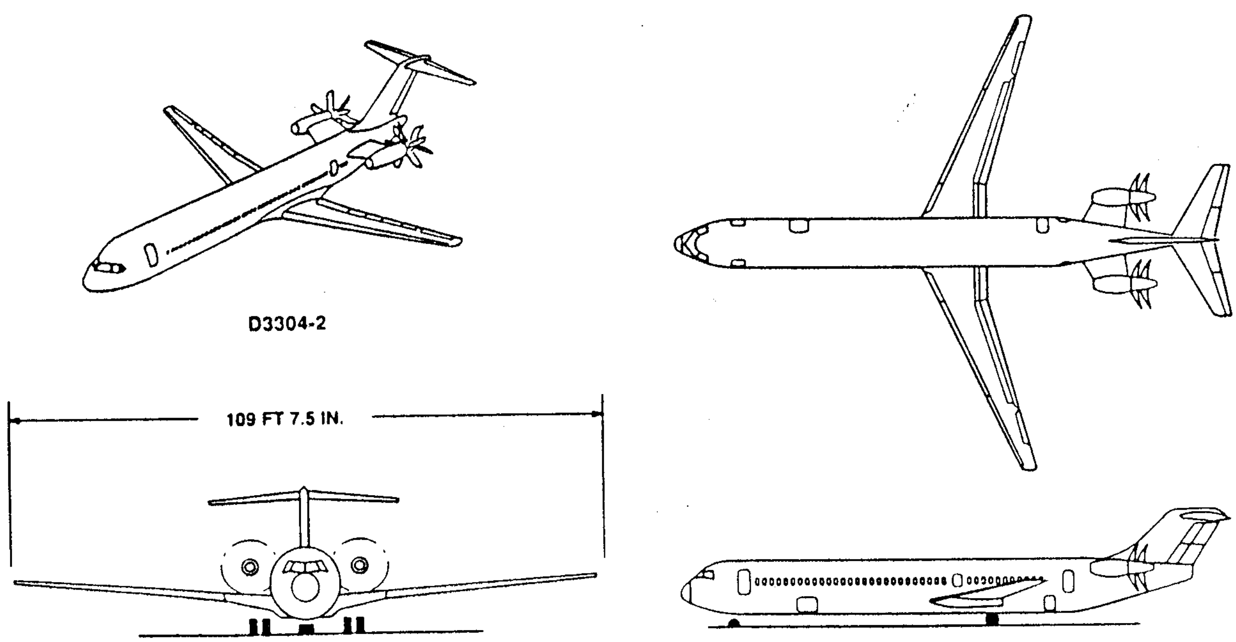


Figure 7. Revised Baseline Aircraft for Wing Study

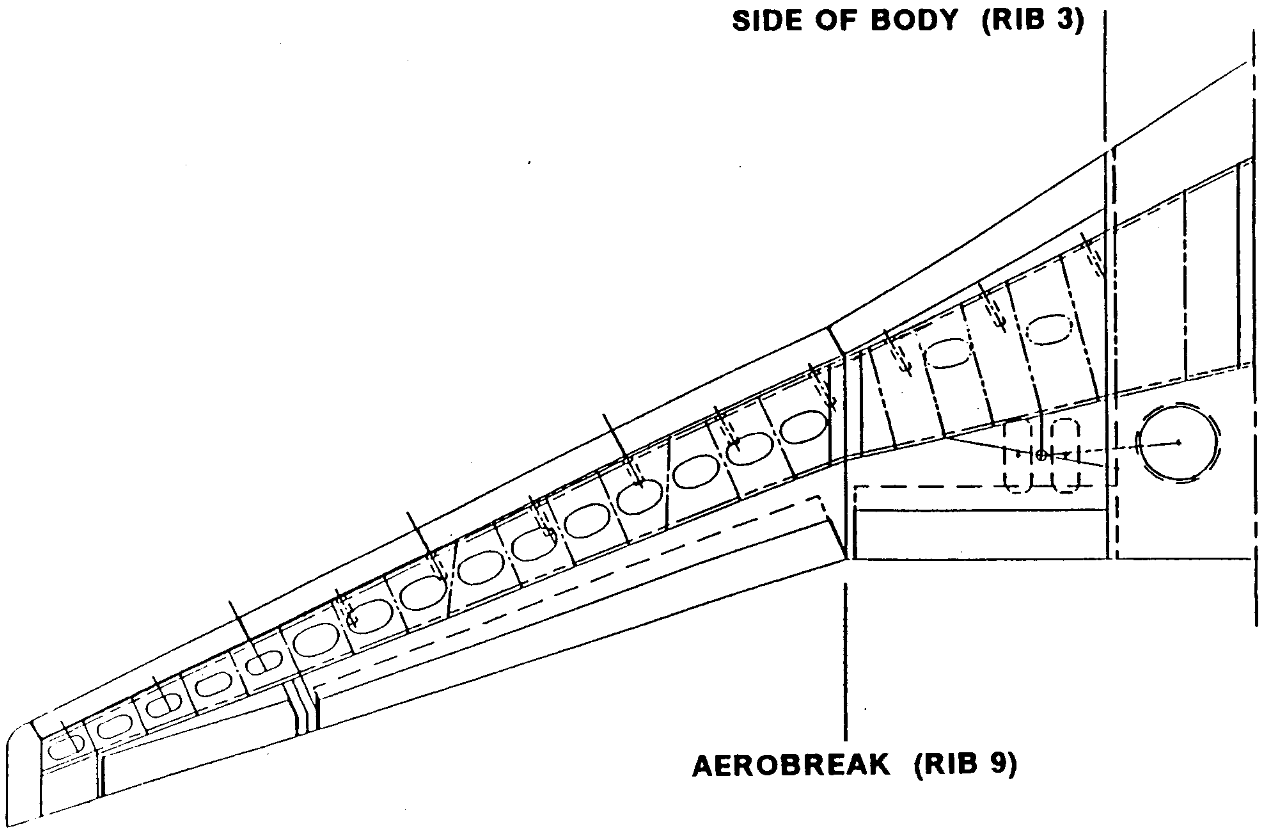


Figure 8. Wing Structural Box Geometry

the validity of the test program. Test results will be compared against analytical predictions based on the actual test conditions, and this will be satisfactory for demonstrating the adequacy of the design approach.

3.3 Wing Structural Configuration

A good data base was available for the baseline wing, including the sizing of a composite structural box that had been designed for strength, based on an allowable 4,500 microstrain. Aeroelastic studies indicated that this wing box was deficient in stiffness, and the optimum manner to add the required incremental stiffness was then deduced. This resulted in bending (EI) and torsional (GJ) stiffness distribution criteria which then provided a new basis for the sizing of the wing.

It is necessary to explain at this point the influence that a stiffness-critical situation has on the design of the wing box. Stiffness imposes a maximum limit on the design strain at which the structure can operate. Any attempt to exceed this limit will result in a structure which is deficient in stiffness. In bending, it is possible to explain this dependence of strain on stiffness by the simple engineering relationship:

$$\text{Strain} = Mc/(EI)$$

Table I. Load Conditions and Damage Factors

LOAD CONDITION NUMBER	TYPE OF CONDITION	ACCN FACTOR n	DESIGN SPEED	DAMAGE CATEGORY		
				A		B*
				LIMIT LOAD FACTOR	ULTIMATE LOAD FACTOR	ULTIMATE LOAD FACTOR
10001	LEVEL FLT	1.0	V_A	1.0	1.5	N/A
10002	SYM. GUST	2.84	V_B			1.0
10003		2.92	V_C			1.0
10004	SYM. MAN	2.50	V_A			N/A
10005		2.50	V_C			1.0
10006		2.50	V_D			N/A
10007	SYM. GUST	2.96	V_B			1.0
10008		3.03	V_C			1.0
10009	SYM. MAN	2.50	V_A			N/A
10010		2.50	V_C			1.0
10011		2.50	V_D			N/A
10102	SYM. GUST	2.84	V_B			1.0
10103		2.92	V_C			1.0
10104	SYM. MAN	2.50	V_A			N/A
10105		2.50	V_C			1.0
10106		2.50	V_D			N/A
10107	SYM. GUST	2.96	V_B			1.0
10108		3.03	V_C			1.0
10109	SYM. MAN.	2.50	V_A			N/A
10110		2.50	V_A			1.0
10111		2.50	V_D			N/A
201	WEIGHT ONLY	1.0	—			N/A
301	BRAKED ROLL					1.0
302	HIGH SINK LANDING					1.0
303	HIGH SPEED TURN					N/A
401	AILERON −30°					1.0
402	AILERON +30°					1.0
500	FUEL OVER PRESSURE	—	—			N/A

Category A is non-detectable damage.
Category B is detectable damage.
* There is no Limit Load criterion to be met for Category "B"

Table II. Inner Wing Design Condition Summary

COVER PANELS - MAXIMUM SPANWISE LOADING	N_x LB/IN. ULTIMATE [AVERAGED ACROSS BOX]	TENSION	23,200
		COMPRESSION	23,600
SPARS - MAXIMUM SHEAR FLOW	N_{xy} LB/IN. ULTIMATE [AVERAGED OVER SPAR DEPTH]		3,700
REQUIRED FLEXURAL STIFFNESS (ROOT)	(EI) 10^6 LB/IN.^{2}		261,300
REQUIRED TORSIONAL STIFFNESS (ROOT)	(GK) 10^6 LB/IN.^{2}		137,500
MAXIMUM FUEL PRESSURE	LB/IN.^{2} ULTIMATE	INNER WING	36.2
		OUTER WING	46.7

ALL LOADS ARE ULTIMATE

Table III. Inner Wing Structural and Test Loads

	Wing Station 72			Wing Station 141			Wing Station 212		
LOAD	S	M	T	S	M	T	S	M	T
10001	43.0	10.4	-.63	34.6	7.5	-.73	27.4	5.2	-.70
10002	93.6	26.5	-.34	82.5	20.1	-.76	69.9	14.3	-.92
10003	111.8	26.7	-.74	86.3	19.5	-1.15	69.8	13.7	-1.26
10004	102.0	25.7	.27	80.5	19.1	-.25	66.8	13.7	-.47
10005	102.7	24.7	-.36	79.5	18.1	-.84	64.6	12.7	-.98
10006	102.4	22.5	-1.30	77.1	16.0	-1.64	60.0	10.8	-1.63
10007	112.0	28.3	.04	90.5	21.0	-.46	75.5	14.8	-.69
10008	114.4	27.5	-.89	90.6	20.1	-1.27	73.9	13.8	-1.35
10009	99.0	24.8	-.14	79.6	18.3	-.56	66.1	12.8	-.74
10010	99.4	23.6	-.97	78.5	17.1	-1.27	63.7	11.8	-1.31
10011	99.4	21.9	-1.46	76.4	15.5	-1.74	60.1	10.3	-1.69
10102	91.5	25.2	.53	78.8	19.1	.02	65.5	13.7	-.25
10103	108.8	25.1	.69	81.0	18.3	.13	63.5	12.9	-.14
10104	100.7	24.9	.72	78.3	18.5	.14	64.1	13.2	-.13
10105	100.5	23.4	.64	75.5	17.1	.04	59.7	12.1	-.20
10106	98.7	20.9	.70	70.5	14.8	.15	52.4	10.3	-.06
10107	109.7	27.0	1.09	86.6	19.9	.47	70.9	14.1	.12
10108	111.5	26.0	.44	85.6	18.9	-.08	68.1	13.2	-.31
10109	97.0	23.6	.72	76.2	17.4	.19	62.0	12.2	-.08
10110	96.4	22.0	.46.	73.2	15.9	.00	57.4	11.0	-.20
10111	95.7	20.2	.54	69.9	14.3	.05	52.5	9.7	-.12

TEST CONDITIONS

	Wing Station 72			Wing Station 141			Wing Station 212		
1				90.6	21.0	.47			
2				76.4	15.5	-1.74			

Limit Load
S = Shear (xE3) lb.
M = Moment (xE6) in-lb
T = Torque (xE6) in-lb

GTU LOAD CASE	JACK NUMBER	ULTIMATE LOAD (LB)
1	1	135,900
2	2	114,600

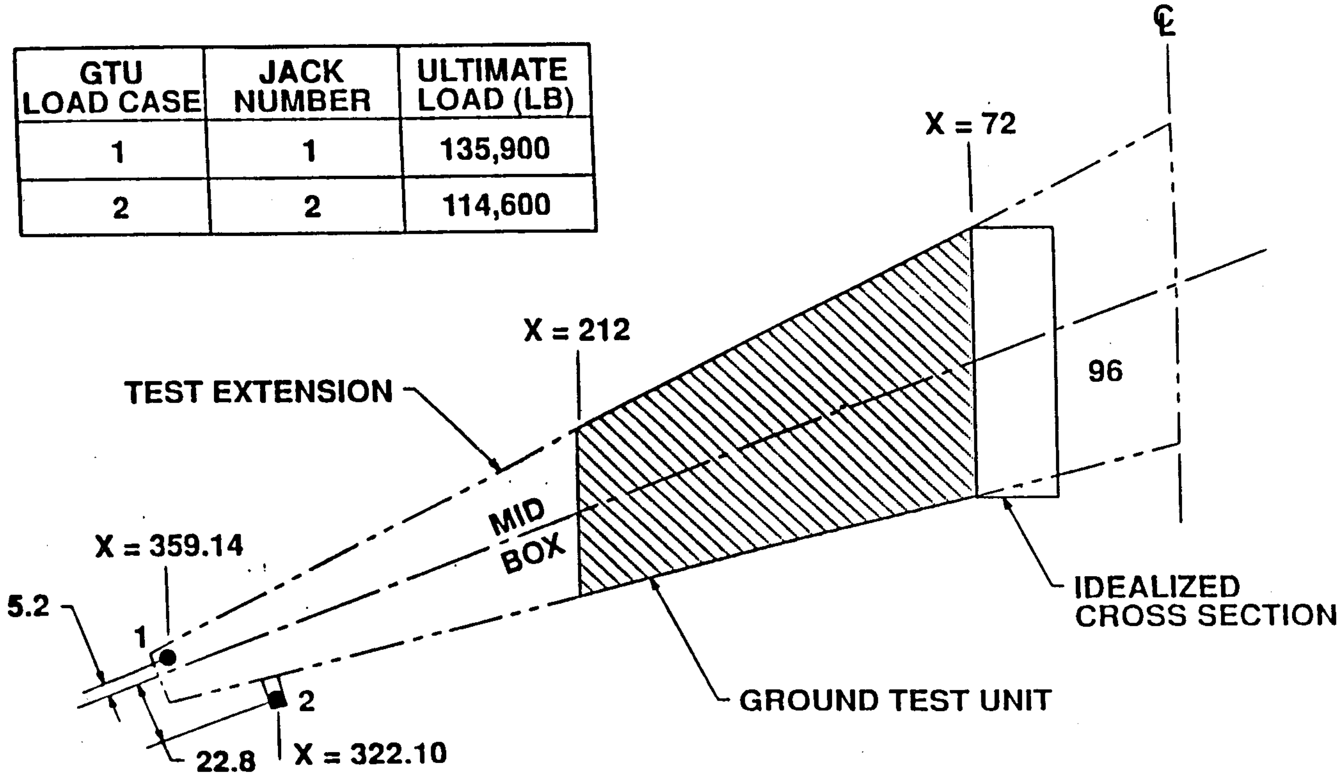


Figure 9. Ground Test Unit Load Conditions

No attempt was made to design a "high strain" wing. Even if the above expression had allowed it, such a wing would have been considered unacceptable. Its complexity would have put the cost out of reach and it would not have met the design repairability criterion. Furthermore, it was believed that a simple "hard skin" design, working in the region of 4,500 microstrain, could be just as weight-efficient as a "soft skin" design working at 6,000 microstrain. It must be emphasized that strain is merely a measure of structural deformation, not of structural efficiency.

In 1984, NASA awarded Douglas a contract to validate the hard skin approach and to show that it could provide a simple wing that was just as weight-efficient as the "high strain" configurations. During this program, stiffened compression panels were fabricated and tested, with damage, to high gross stress levels (Reference 6).

A second influence of the stiffness-critical requirement is that, for a given material modulus, cover panel weight is independent of the type of stringer used. This is because the spanwise bending rigidity (EA) of the cover panels at any station is a direct function of the required (EI) at that station. For this reason, it was possible to select the easiest stringer to fabricate, providing it could be shown to also satisfy strength and stability design requirements. A "blade" stringer configuration was found to satisfy these needs.

The shell of the box formed by the spars and the cover skins was first sized with a uniform shear stiffness (GT) to satisfy the (GJ) requirements. Laminate patterns were selected to give a preponderance of ±45° layers in the spars and of 0° layers in the cover skins. However, practical limits in this choice were imposed by two major design considerations, damage tolerance and repairability, as dictated by the criteria given in Section 2.1 of this report. The selected skin pattern was based on a 9-ply stack (0/45/0/-45/90/-45/0/45/0), with 44.4 percent plies in the 0° direction. This layup is within the Douglas guideline limits as can be seen from Figure 3.

The skins designed in this way were found to be compatible with good stringer design. The shell alone, sized by torsional stiffness, met only a part of the bending stiffness requirement. The stringers provided the remainder and also provided the rigidity needed for general stability of the box structure. The laminate patterns for the stringers and skin are identical, thereby eliminating the distortion and prestresses that follow each thermal cycle. Shear and extensional stiffness matches resulting from this approach are shown in Figures 10 and 11.

3.4 Wing Box Units

The ICAPS Wing Box Unit is a structural specimen representing a 12 foot section of the inner wing box. It is derived from a similar box developed in-house and benefits from that effort in the following ways:

1) the structural design is essentially similar
2) the rib members are identical and built with the in-house tools
3) the same assembly fixture will be used for both boxes
4) an in-house extension box will be adapted to introduce the test loads, and
5) the in-house test fixture would be used for both boxes.

Slight changes have arisen in the design of the ICAPS box due to the fabrication process under development. The stringer shape has been modified, while essentially retaining the same basic dimensions (Figure 12). The central stack of layers in the blade was eliminated by using thicker skin flanges which taper toward their edges. This change was made possible because the stitching that holds the stringers to the skin dispenses with the need for bolted attachments through the skin.

By similar reasoning, it was decided to incorporate separate spar caps into the skin preform as shown in Figure 13. This allows the spars, like the ribs, to be simple webs joining the two cover panels together. Hence the stitching is providing two additional benefits to the overall design:

1) it is allowing most, if not all, of the bolts through the cover panels to be eliminated, thereby saving cost and alleviating fuel leakage problems, and
2) it simplifies box assembly by avoiding fit-up problems and allows outer mold loft tooling to be used.

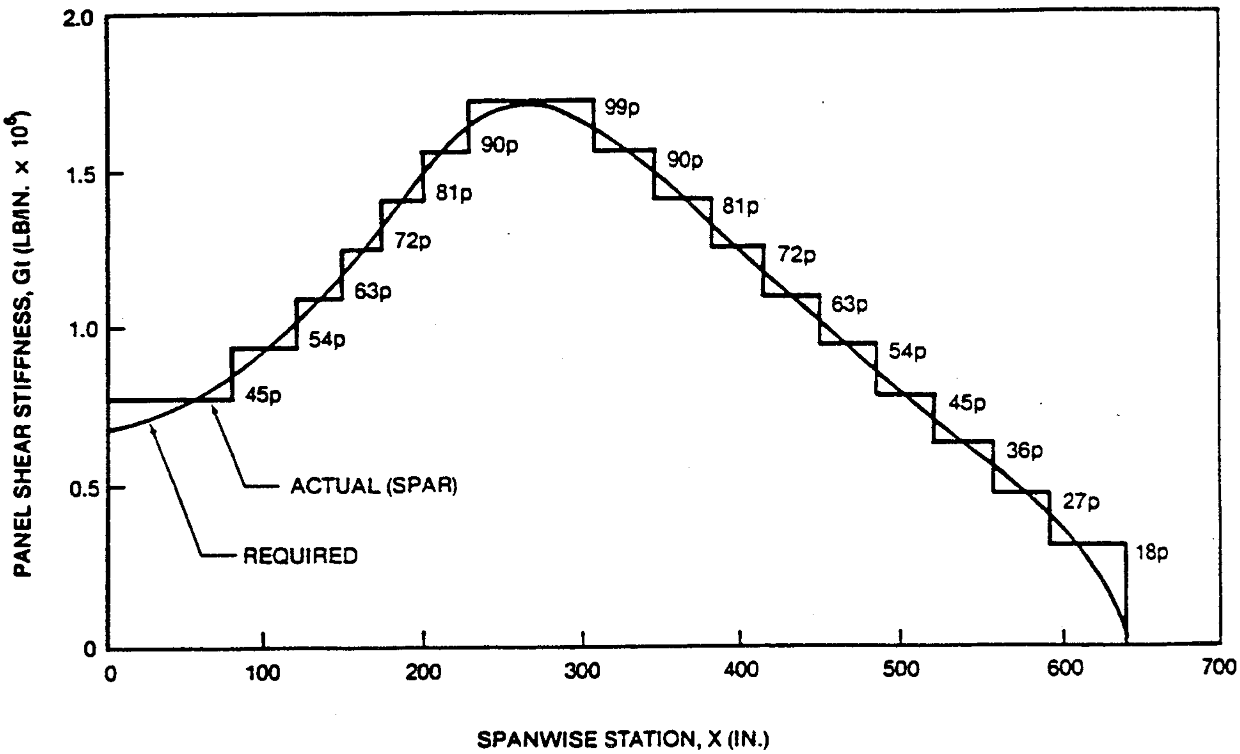


Figure 10. Shear Stiffness Match

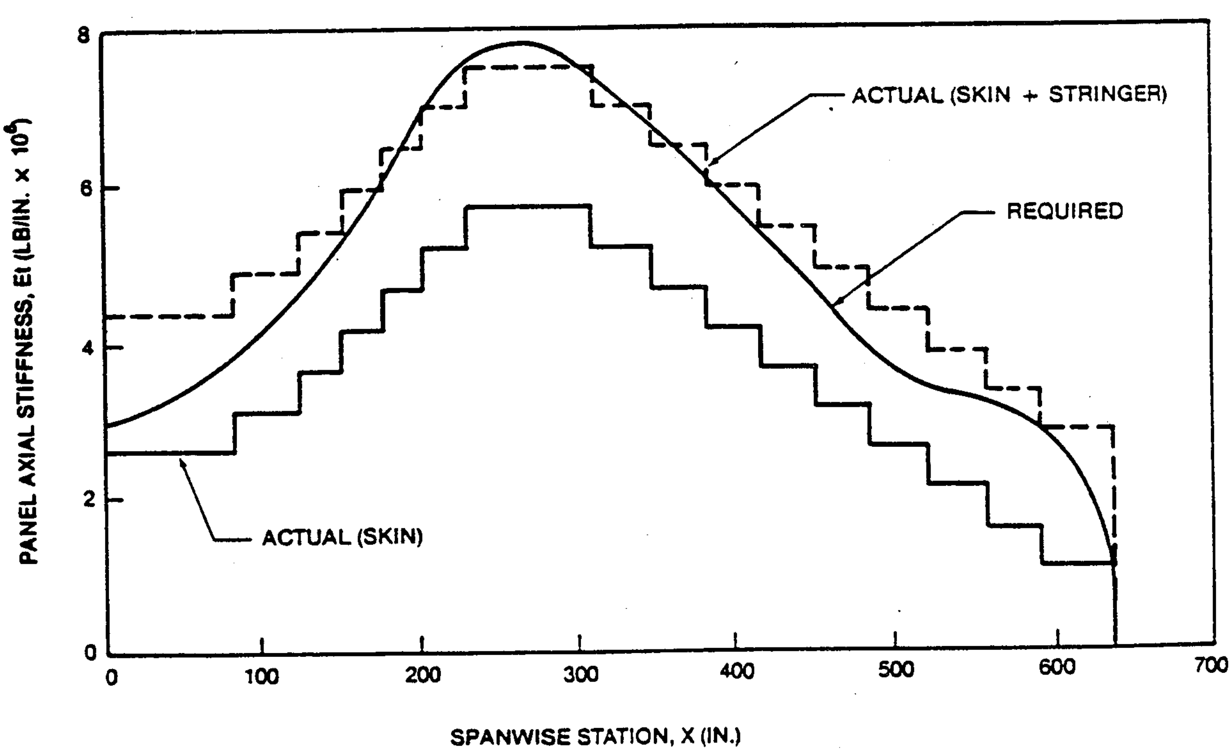


Figure 11. Extensional Stiffness Match

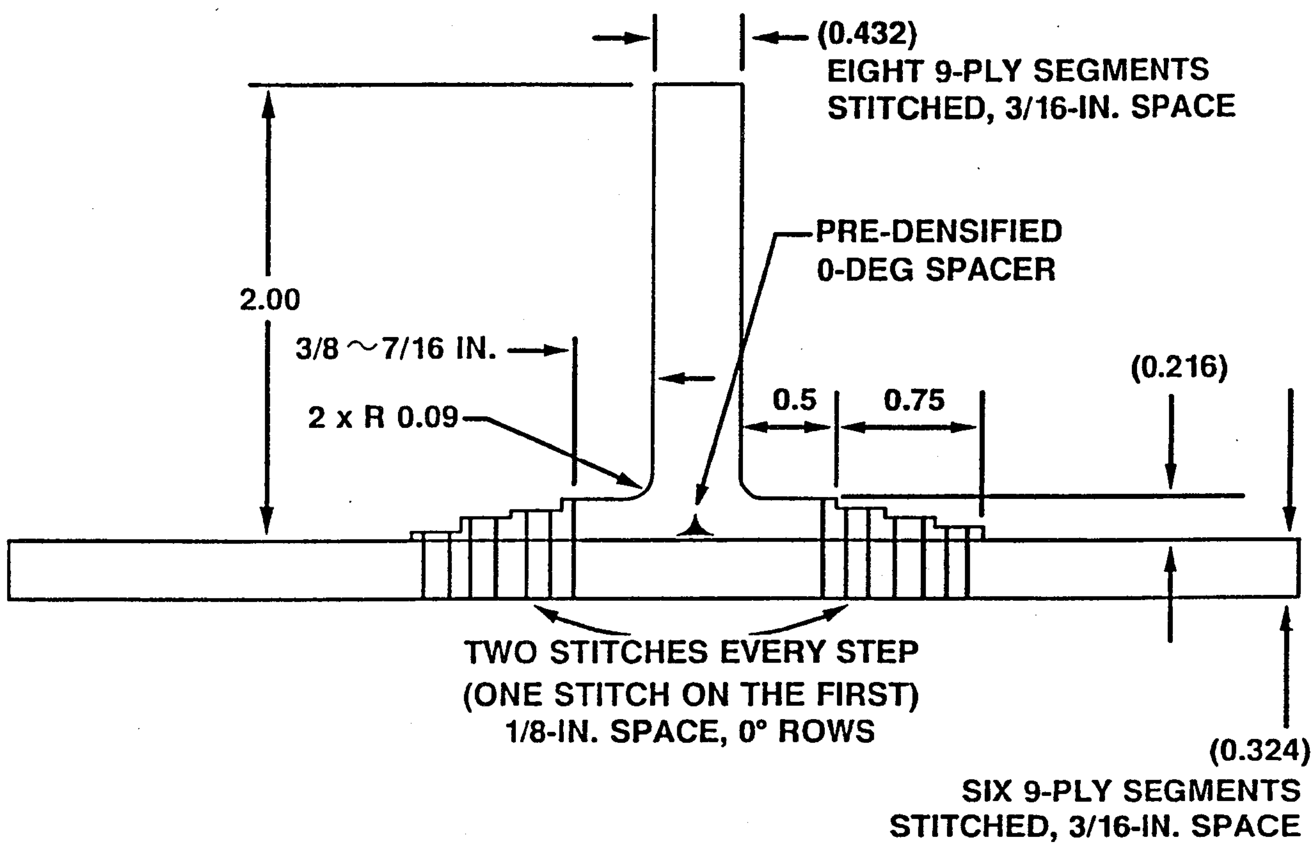


Figure 12. ICAPS Wing Stringer Configuration

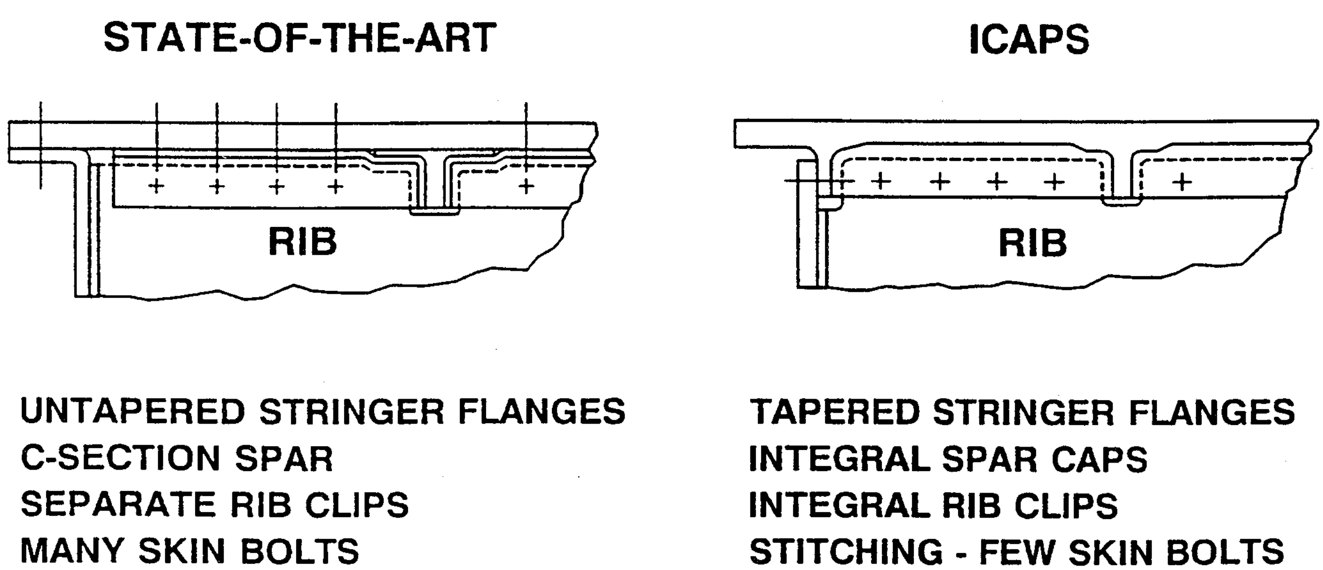


Figure 13. ICAPS Design Changes

3.5 Manufacturing Cost Model

It is extremely difficult for a designer to minimize the cost of a composite component. This difficulty arises because, unlike when designing for minimum weight where precise equations can be used, cost implications are usually presented as opinions rather than as hard facts. For this reason, a Composite Manufacturing Cost Model was developed as part of the in-house Composite Wing program.

The model was developed by a joint Douglas/Arthur Anderson team to help answer primary management questions. It evaluates a manufacturing facility to build composite parts and compares the cost, using this facility, to the cost of purchased metal components. The facility is conceptually designed using manufacturing cells to minimize part moves. The characteristics of the parts being fabricated drive the processing logic. Depending upon the part being processed, the model calculates equipment hours, tooling hours, labor hours, and materials required. These accumulate and provide the base to obtain the total facilities requirements. The resulting costs are then computed and represented as one-time and recurring costs.

The model is designed to be routinely updated as the design and manufacturing processes are refined. In this way, the model offers several benefits:

- Consistency - All evaluations of the facility are based on the same calculating criteria.
- Completeness - The model considers costs associated with building and running a remote undesignated ("greenfield") facility.
- Supporting Data - Levels of equipment, tooling, labor, and material usage are all recorded and easily accessible.
- Adaptability/Flexibility - The model can be used for other applications, such as the fuselage components, by making changes in table values and manufacturing processing assumptions.

Access to the model provides the designer with a valuable tool for decision making and allows rapid changes to be made in structural configurations to achieve a truly weight-efficient and cost-effective component. This model will be used to assess and store information derived from the ICAPS program. A wing box cost data projection is shown in Figure 14.

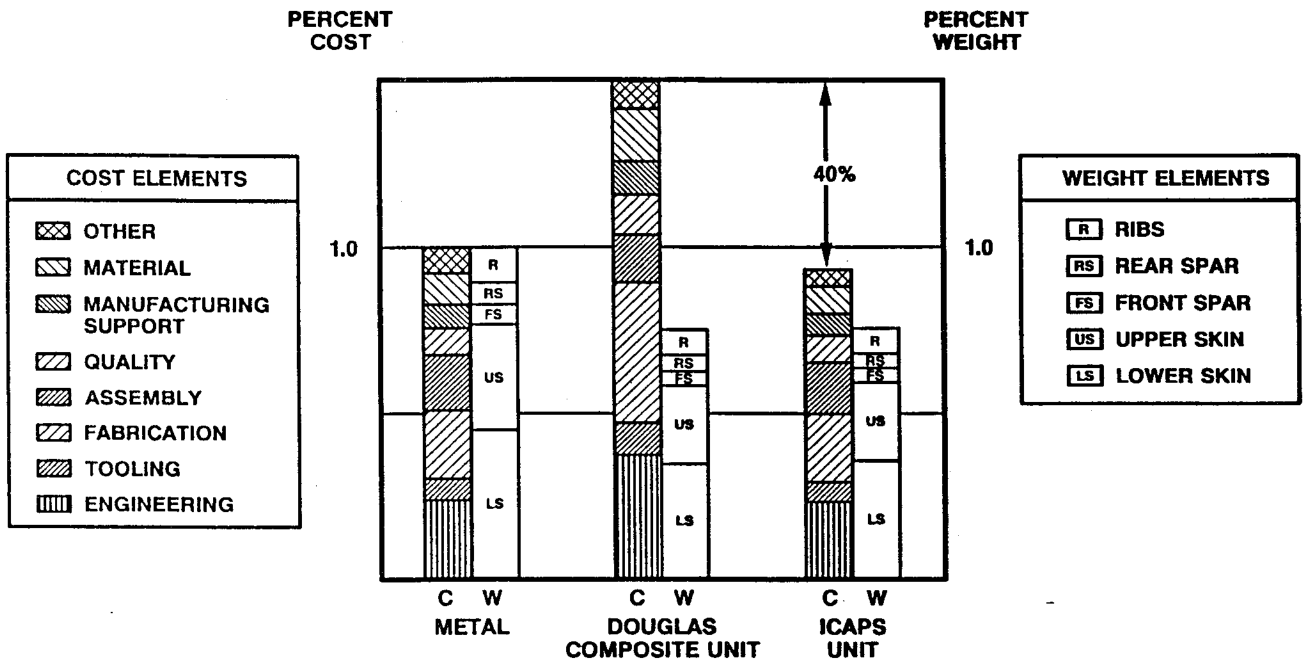


Figure 14. Wing Box Cost Projection

SECTION 4

FUSELAGE DEVELOPMENT

A study on the utilization of advanced composites in fuselage structures of commercial aircraft was conducted by Douglas under NASA contract (Reference 8). That work was followed by a second NASA contract on the development of composites technology for joints and cutouts (References 9 and 10). These two programs developed the structural configurations that are being used in the ICAPS program. The baseline aircraft selected for these earlier studies was a proposed derivative of the DC-10, namely the MD-100 (Figure 15). The component selected was the forward fuselage barrel just ahead of the wing. This barrel is 364 inches long and has a constant 118.5 inch radius. The section contains two 42"x76" passenger doors, a 104"x58" cargo door, 26 windows, 17 full frames, 19 floor beams, and 103 longerons (Figure 16). The barrel section is joined to the nose and aft fuselage sections by mechanically fastened circumferential skin and longeron joints.

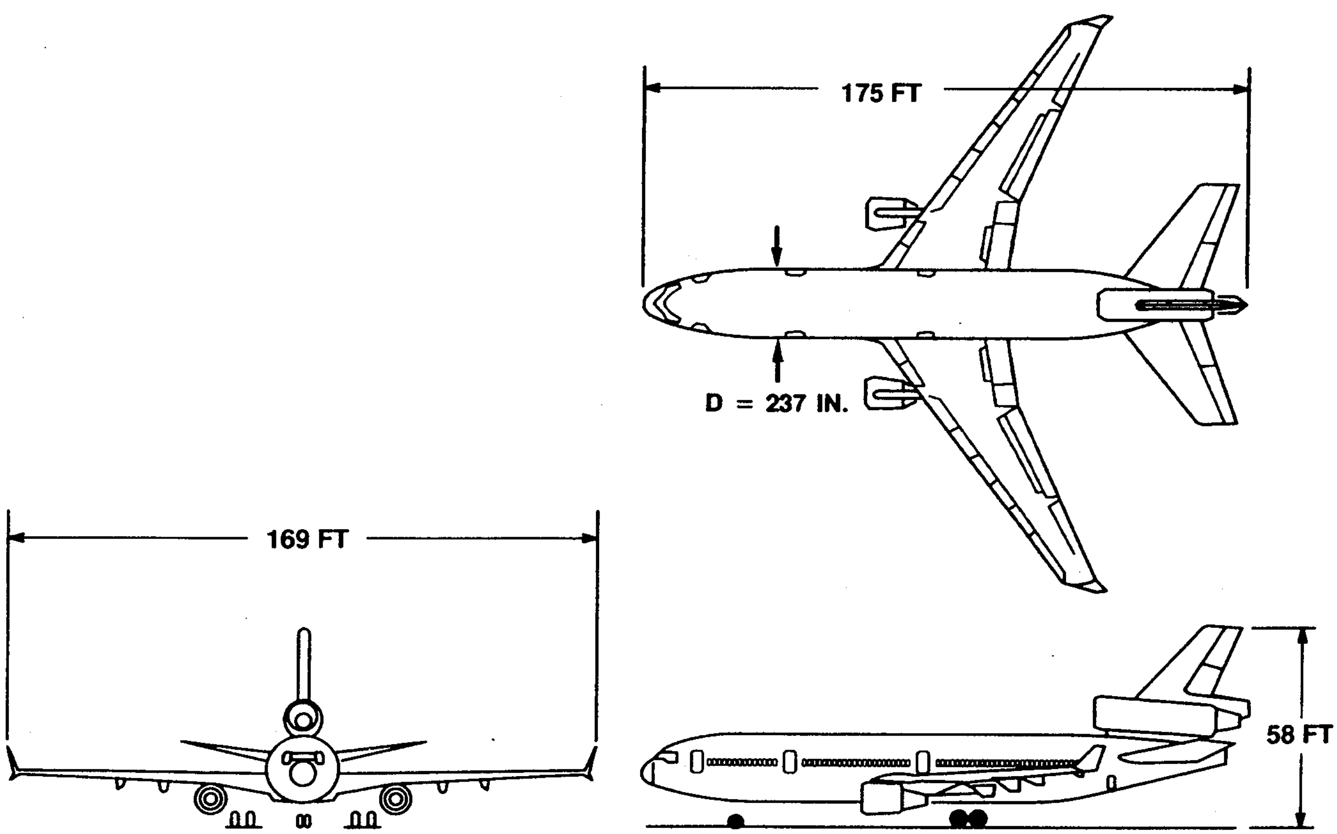


Figure 15. Baseline Aircraft for Fuselage Study

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4.1 Design Criteria and Loads

The fuselage structure is designed to meet the load conditions specified in Federal Aviation Regulations Part 25 (FAR 25). All critical ground and flight conditions are included. Maneuver and gust envelopes will be met with and without the addition of an internal limit cabin pressure (P) equal to: 8.6 + 0.5 (valve tolerance) psi. An ultimate load condition of 2P acting alone is also specified.

Fatigue, durability, and damage tolerance requirements will be fully met. One fatigue lifetime is 60,000 flight hours. The structure will not be allowed to buckle while standing on the ground or in 1.0g flight. Provisions will be made in the design for both bonded and bolted repairs to be affected during the service life of the aircraft. Fasteners through the skin will be flush and the plain (uncountersunk) portion of the hole shall not be less than 0.010 inches in thickness. Minimum diameter for threaded fasteners is 3/16 inch.

Maximum and minimum values for vertical shear and bending moment for the baseline aircraft are shown in Figures 17 and 18. Shell longitudinal and shear loadings are shown around the circumference for fuselage station 1109 in Figure 19. These loads are summarized in Table IV for each barrel segment.

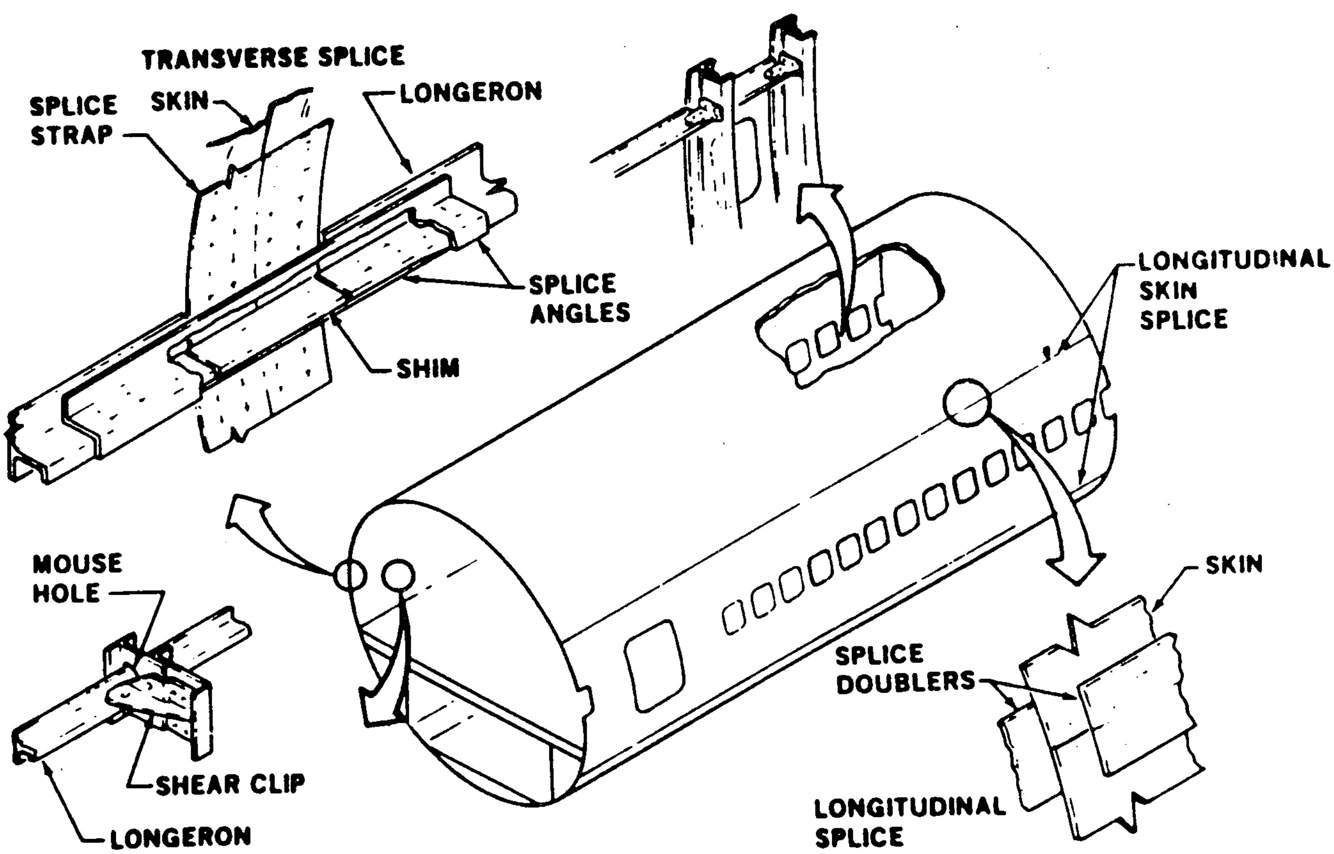


Figure 16. Fuselage Barrel Component

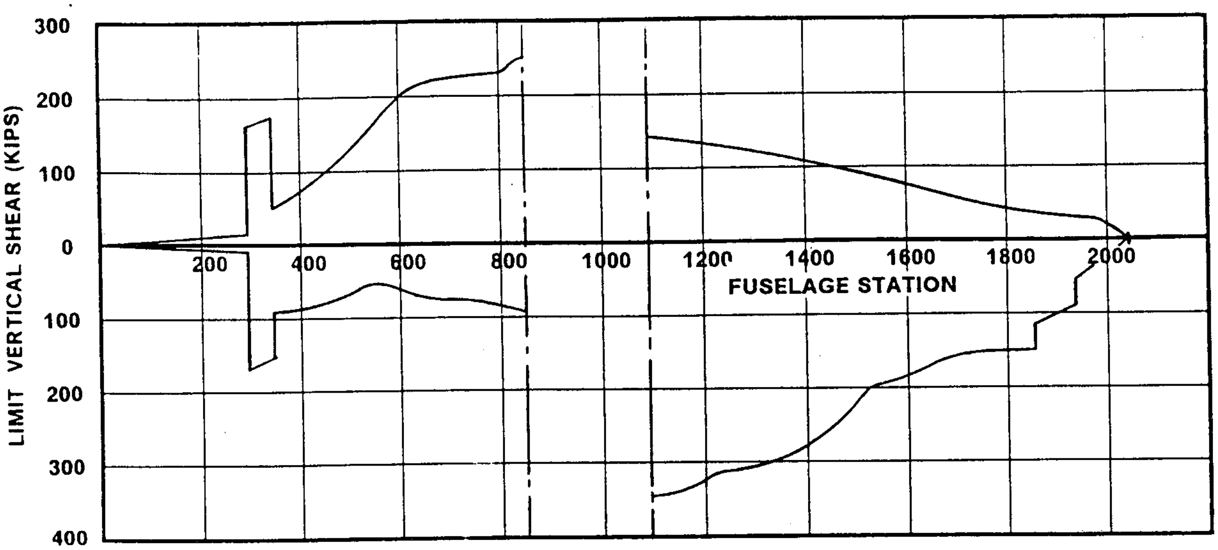


Figure 17. Fuselage Vertical Shear

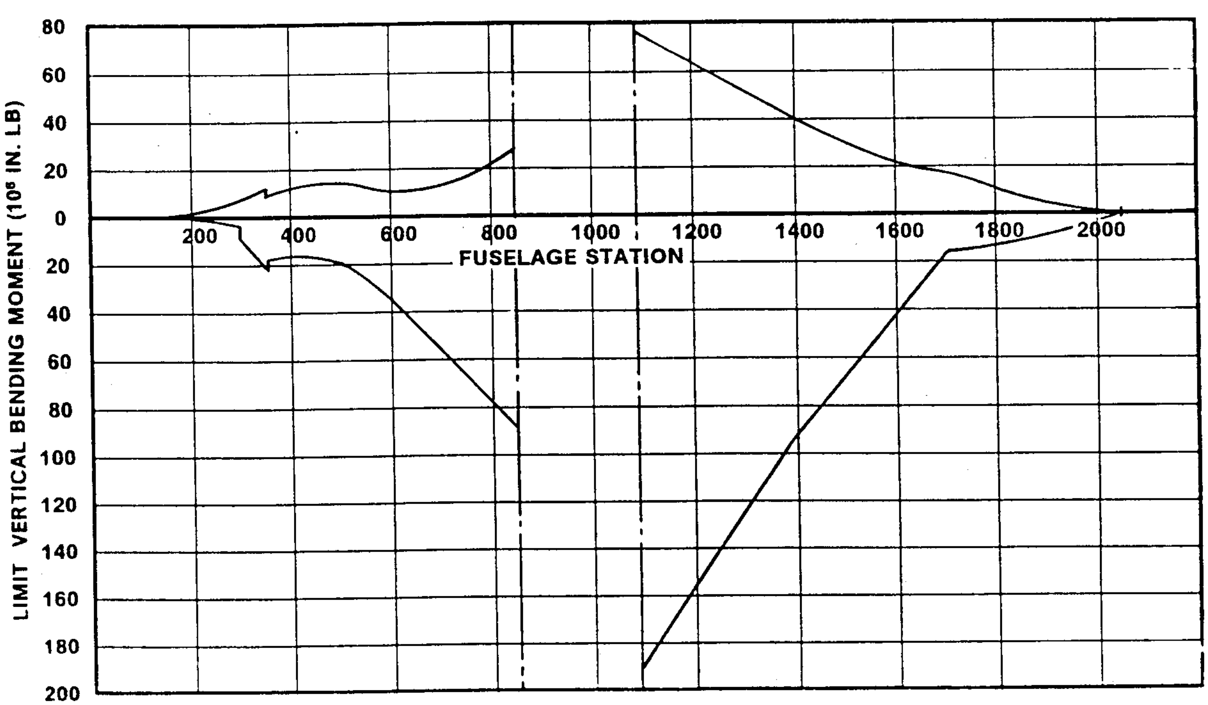


Figure 18. Fuselage Vertical Bending Moments

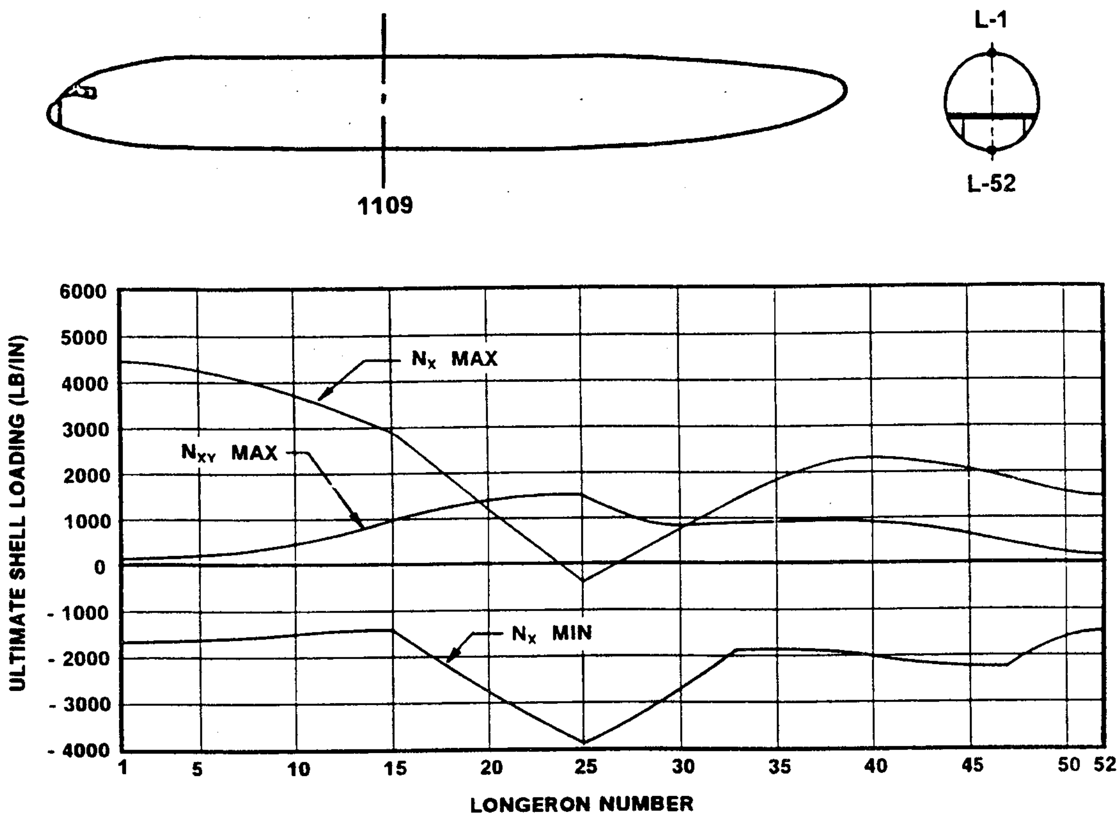


Figure 19. Load Distribution Around Shell

Table IV. Fuselage Design Loading Summary

CONDITION		CROWN	SIDE	KEEL
MAXIMUM LONGITUDINAL TENSION LOADING	N_x LB/IN.	4,600	3,200	2,300
MAXIMUM LONGITUDINAL COMPRESSION LOADING	N_x LB/IN.	-1,700	-3,000	-2,100
MAXIMUM SHEAR FLOW	N_{xy} LB/IN.	800	1,500	1,000
MAXIMUM HOOP TENSION LOADING (PRESSURE = 9.1 PSI)	N_y LB/IN.	2,157 (AT 2p)		

ALL LOADS ARE ULTIMATE

4.2 Design Description

The fuselage concept consists of four stiffened skin panels which are joined by mechanically fastened longitudinal splices to form a complete barrel section. For damage resistance, a minimum gauge skin was selected to be a $(0,90,+45,0,-45,90)_s$ carbon epoxy material layup, 0.072 inch thick. The skin provides hoop pressure and shear load carrying capability. The bending and longitudinal pressure loads are carried by both the skin and the longerons. Straps consisting of four plies of tape oriented in the hoop direction are incorporated in the skin at each frame station. There are two major reasons for this design. First, the straps act as crack arrestment strips, and second, they reduce the adverse effects of the "mouse hole" cutout discontinuity at each longeron/frame intersection. The skins are reinforced to a quasi-isotropic layup at the skin splices and cutouts. Additional reinforcement is used near the rear of the barrel section to prevent premature shear buckling.

The fuselage skins are stabilized by "J" section longerons which are co-cured with the skin. For ease of manufacturing only two longeron layups are used. The flanges of the basic longeron consists of a 33-percent 0° material layup, and this is used in the majority of the fuselage barrel. A 50-percent 0° material is used in areas of high axial load such as the crown and keel regions (Figure 20). The longerons are spliced by back-to-back Z- and L-section straps.

Frames at 20-inch spacing are used to support the stiffened shell structure, and consist of a "Z" cross-section with quasi-isotropic webs (Figure 21). The flanges have a layup containing a higher percentage of 0° material. The webs are mechanically attached to the shear tees and shear clips. The shear tees (Figure 22) use cloth material for enhanced drapability. Longerons and shear tees are secondarily bonded to the skin with FM-300 adhesive. Intersections between the longerons and frames are provided for by "mouse holes" in the shear tees at each longeron location to allow the longerons to pass through. The frames pass over the longerons as shown in Figure 23. Stability of the shell is enhanced by tension clips between the frame and longeron, where required. A photograph of a completed fuselage ATP 6-longeron panel is shown in Figure 24.

The baseline fuselage design contains cutouts for windows, and for passenger and cargo doors. The reinforcement around the windows consists of a window belt doubler which extends 9 inches above and below the windows. The doubler is made of quasi-isotropic material and has a maximum thickness of 0.33 inch , excluding the skin.

The longitudinal splice, located midbay between longerons, is a four-row double-shear design utilizing both internal and external splice straps. The basic 0.072 inch skin panel is increased to a 0.096 inch quasi-isotropic laminate for countersink depth requirements. The transverse skin splice is designed with a single internal splice strap for aerodynamic flushness.

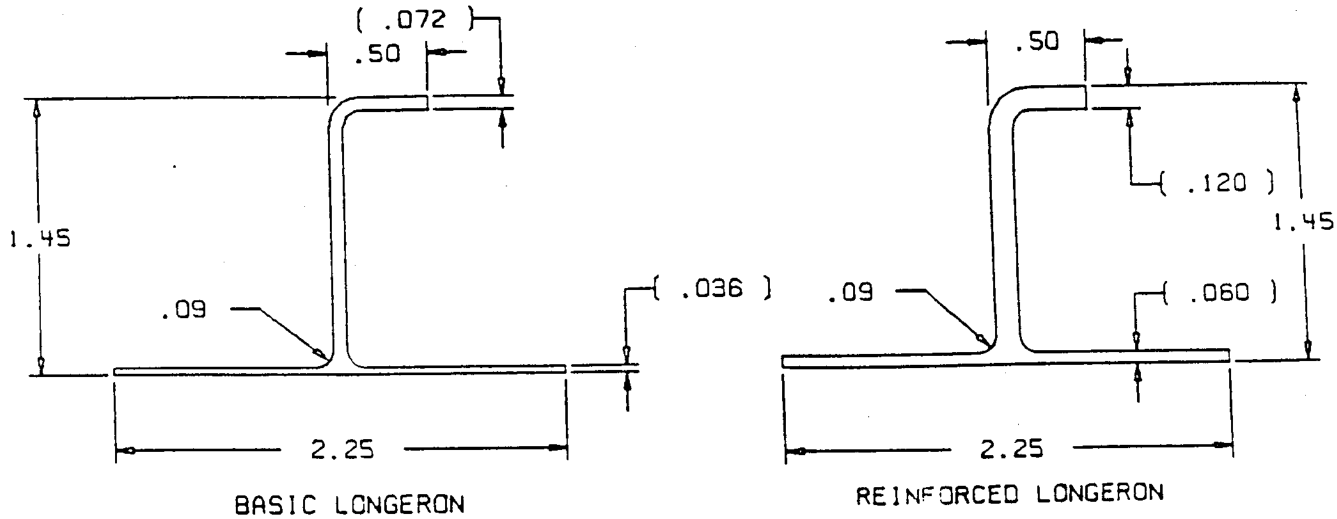


Figure 20. Longeron Sections

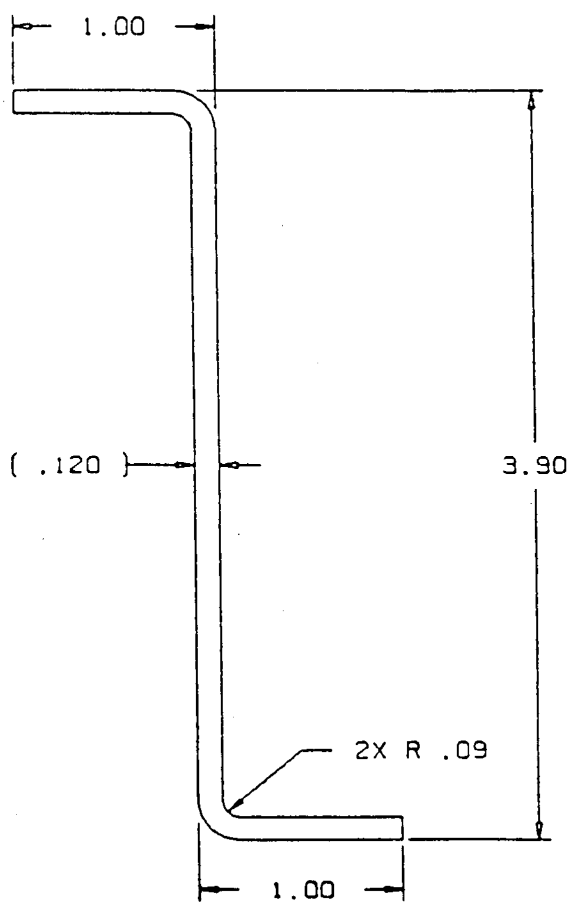


Figure 21. Frame Section

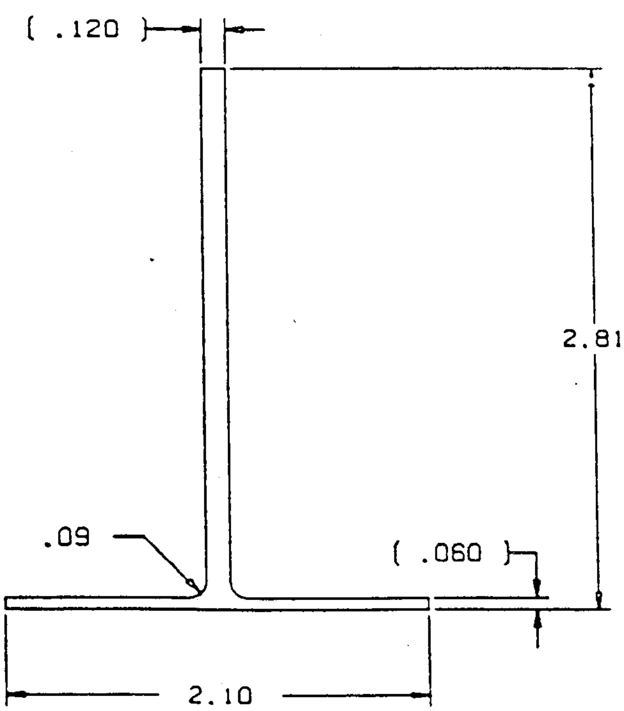


Figure 22. Shear Tee Section

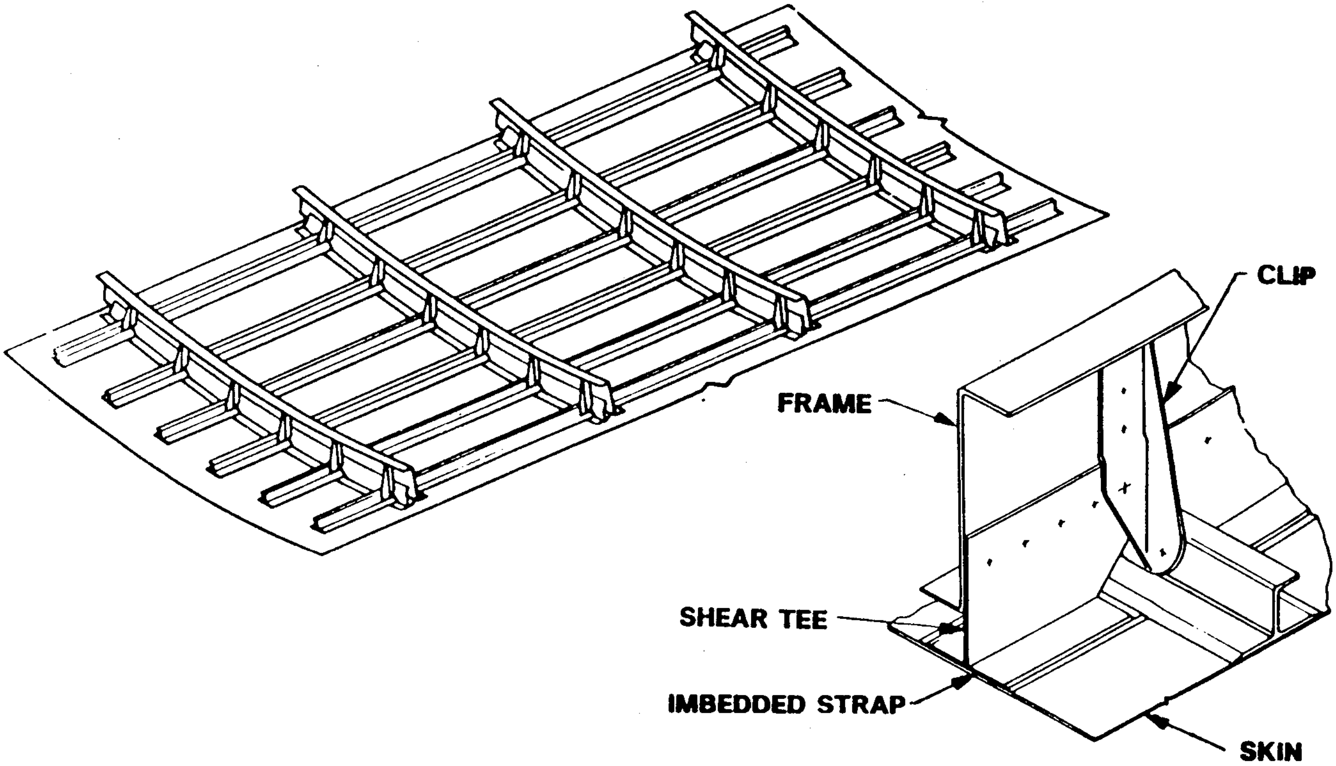


Figure 23. Skin Panel Assembly

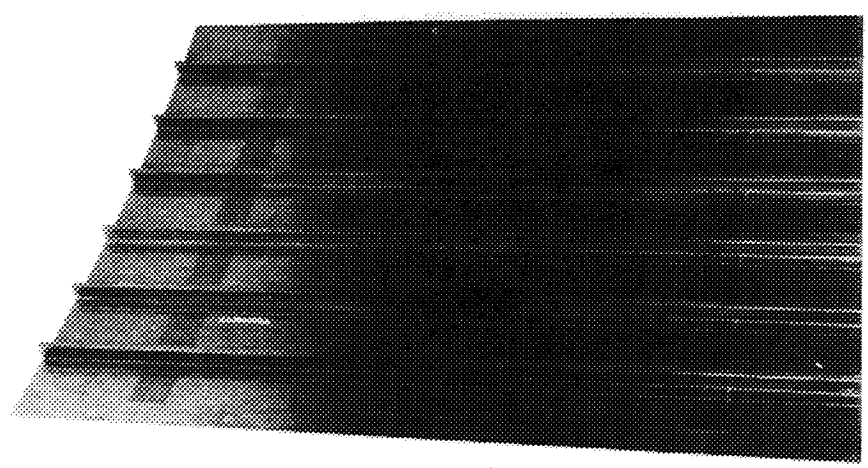


Figure 24. Completed Fuselage ATP Panel

Since the fuselage skin is in a swept-stroke lightning attachment region, it has to be capable of withstanding a lightning restrike without sustaining an unreasonable amount of damage. Ideally, any damage inflicted by a typical lightning strike has to be easily repairable by a nonstructural cosmetic repair. This level of lightning strike resistance probably requires some type of skin protection. The layup of the outer two layers of tape was designed so that a biwoven cloth-based lightning protection system could be substituted for them. A nickel-coated carbon fiber protection system and a system composed of fine aluminum wires woven into carbon cloth were evaluated. Tests indicated that the latter was superior for lightning protection and this was chosen for use in the conceptual design.

4.? ICAPS Panel Specimens

Fuselage panel test specimens, essentially similar to the baseline configurations described above, are being fabricated by two different fabrication approaches. These are the Automated Tow Placement (ATP) and Resin Transfer Molding (RTM) approaches, respectively.

The Hercules Aerospace Company is under subcontract to fabricate the ATP panels. Skins are being tow-placed on winding mandrels by the Hercules ATP machine. Longerons are made in separate forming and cure tools and are cured with the skins for each specimen.

Dry preforms for the RTM specimens are being fabricated by stitching together individual layers of unidirectional cloth material. Stitching is used selectively in skin regions to reduce damage size and to alleviate the propagation of delaminations. Stitching is also used to attach longeron preforms to skin preforms, to enhance resistance to longeron separation when impacted and when in a post-buckled state. Resin transfer tooling is being developed to fabricate panels by the pressure method rather than by the film infusion process being developed for the wing.

In both approaches, frame shear tees are attached to the skins by secondary bonding, and the frames are attached to the tees by bolts.

SECTION 5
STITCHING DEVELOPMENT

Stitching has long been recognized as a means of inhibiting secondary through-the-thickness failures in laminated structures. Early stitching experiments with prepreg materials were considered largely unsatisfactory because of a significant loss of material properties due to fiber breakage during the stitching operation. With the advent of the dry fiber preforms used with RTM technology, stitching is more successful because the fibers are deflected easily by the stitching needle. Even so, there is some loss of properties and the effect on structural weight must be considered.

Degradation of material modulus values could be a serious deterrent for stiffness and buckling critical structures. Reduction in undamaged material strength is not significant in itself unless it is reflected in compression-after-impact (CAI) or loaded bolt hole behavior. Indeed, the major benefit of stitching arises from the fact that the allowable working stress level in a compression-loaded structure is enhanced by stitching even though the ultimate undamaged strength of the material is decreased (Figure 25). This advantage is particularly apparent with the lower cost, and more brittle, 3501-6 type of resin material and it allows these materials to compete effectively with the high-cost tough epoxies. It is this cost advantage, together with a potential reduction in fabrication costs due to automation, which is sought in this program.

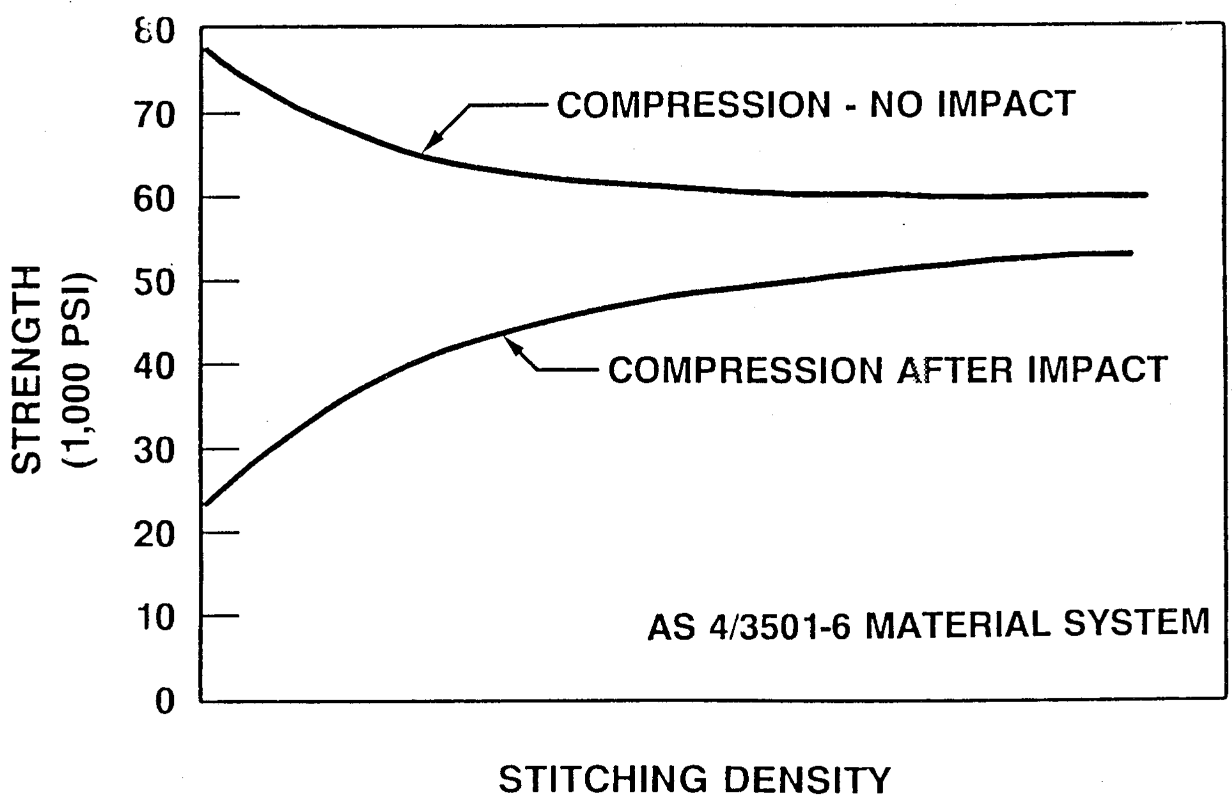


Figure 25. Benefit of Stitching on Compression Behavior

It is not expected that stitching will show an advantage in the weight of tension structures. However, there can be a significant economic benefit if the incidence and size of damage can be reduced. Maintenance and repair are major life-cycle cost items and these costs must be balanced against the acquisition cost of each component. The designer will have to choose selectively whether stitching is beneficial, and which stitching variables should be selected for use in any particular part of the structure.

The sequence of stitching developed for making wing panels is shown in Figure 26. A prior step, not shown, is the pre-plying of the nine-ply stacks by light stitching. Light stitching has the capability to produce a preform sheet of multilayers of material, of any desired ply stacking orientation, that can be cut and handled without major distortion of individual plies. The cost benefit of laying a multi-layer stack instead of individual plies can be appreciated. The stitching process shown in Figure 26 was used to make a series of 3-stringer panels under NASA contract (NAS1-17701). Subsequent compression testing at NASA demonstrated extremely high retained strength of damaged panels.

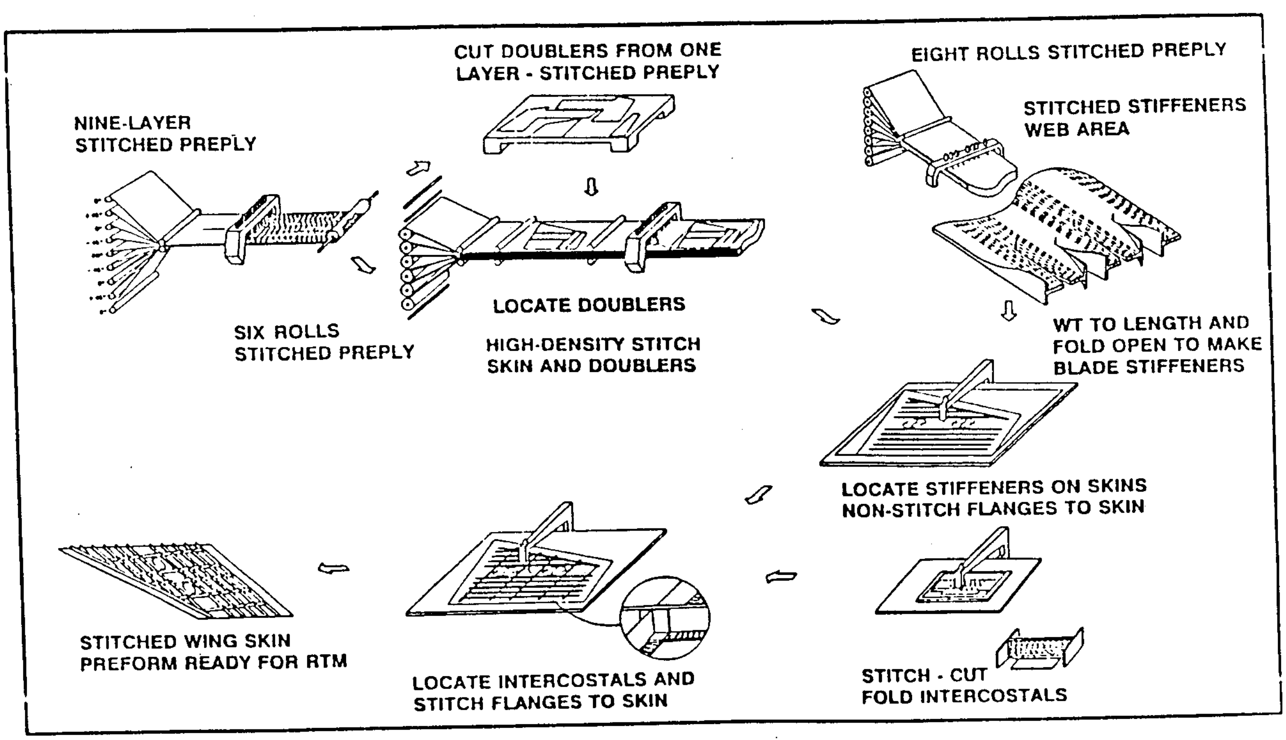


Figure 26. Stiffened Panel Stitching Process

The stitching process is covered by a Douglas patent (Reference 11). The development was at first a cost-sharing cooperative program with Textile Products, Inc., Anaheim, California. This company, now called Ketema, is under subcontract to perform all the lock-stitch operations in this phase of the ICAPS program. Chain stitching was performed by Puritan Industries. Two new stitching machines have been purchased in this program specifically to produce the required cover panel preforms. It is intended to move the machines to Douglas when checkout is complete.

5.1 Stitching Variables Study

In the study of stitching variables described in this section, the baseline dry material form used was unidirectional, non-crimp, non-twist cloth, with 95 percent or 97.5 percent of 3K AS4 fibers in the warp direction. A light non-twist E-glass epoxy-compatible fill thread with 8-10 yarns per inch holds the carbon tows together. The areal weight of the dry cloth is 145 gm/m^2. Each laminate test specimen contained 48 layers of cloth in a $[45°/0°/-45°/90°]_{6S}$ symmetric quasi-isotropic arrangement.

Individual test specimens with this layup were stitched in rows spaced at a specified distance apart. In each row the needle moved forward a distance, known as the step length, between needle penetrations. The number of rows per inch multiplied by the number of steps per inch in each row gives the total number of needle penetrations per square inch. Since in each penetration, the thread goes both into the preform and then out again, the number of threads is double the number of penetrations. Stitching density is defined as the number of threads per square inch multiplied by the tensile strength of the thread, and is measured in Lb/square inch,

The stitched dry preforms were filled at Douglas with 3501-6 epoxy resin by the vacuum impregnation molding process, and test specimens were prepared. A series of tests was conducted at NASA Langley Research Center to investigate the influence of stitch type, stitching thread, the pattern of stitching, and the difference between thick and thin laminates. In the first sequence of tests a comparison was made between lock and chain stitching in which the following parameters were held constant:

Penetration Thread	Fiberglass S2-449-1250, Untwisted (Baseline)
Lock Tie Thread	Kevlar 29, 2 end, 200 d, Twisted *
Stitch Step	1/8" Forward Per Penetration
Stitch Pattern	0° Parallel Rows 1/8" Apart

The characteristic features of lock and chain stitching are shown in Figure 27. In standard lock stitching, the knots formed by the needle and bobbin threads are located within the layers. However in this study, to minimize carbon fiber damage, thread tensions were adjusted to provide a modified lock stitch which positioned the knots on the outer surface of the stacked fabrics.

* Kevlar is a Du Pont trade name

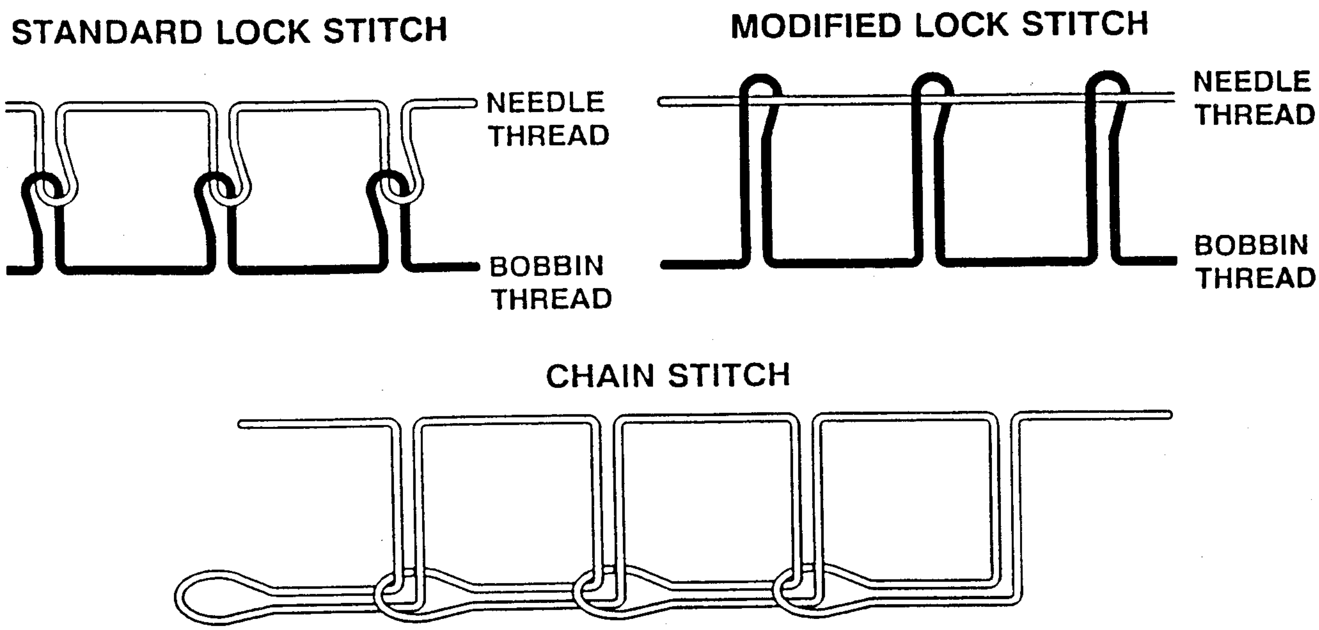


Figure 27. Stitching Types

Lock stitching requires access to both surfaces of the material being sewn and the bobbins require frequent replacement. However, this type of stitching has the advantage in being able to stitch in any direction, even to the extent of having the capability to stitch identification codes into the laminate during the manufacturing process. Lock stitching also uses a less heavy thread than that used in chain stitching and, for a given stitching density, this provides a weight advantage.

Chain stitching has the advantage of using a single thread; however, most chain stitching machines use a needle motion to move the material being stitched, and current machines cannot be used to stitch fabric preforms of the size required for aircraft structures. Test results showed that chain stitched panels had slightly better damage tolerance properties than lock stitched panels (Figure 28). This result was in contradiction to earlier in-house testing that showed the reverse to be true. Both types gave very acceptable results but the modified lock stitch was selected as a standard for this contract effort. Lock stitch machines were judged to have the better capability for stitching large complex preforms having skin taper and localized doubler buildups.

A second series of tests evaluated the different stitching penetration threads listed below:

Thread Number	Thread Type	Thread Weight (Yds/Lb)	Thread Strength (Pounds)	Penetrations/ Square Inch	Stitching Density (Lb/Sq. In.)
Baseline	S-2 Glass 449 1250 Untwisted	1250	59	64.0	7552
1	S-2 Glass CG-150 10/0 Untwisted	1500	49	64.0	6272
2	S-2 Glass CG-150 8/0 Untwisted	1875	39	64.0	4992
3	S-2 Glass 463 750 Untwisted	750	98	28.4	5575
4	Kevlar 29 1000 d Untwisted	4470	36	64.0	4608

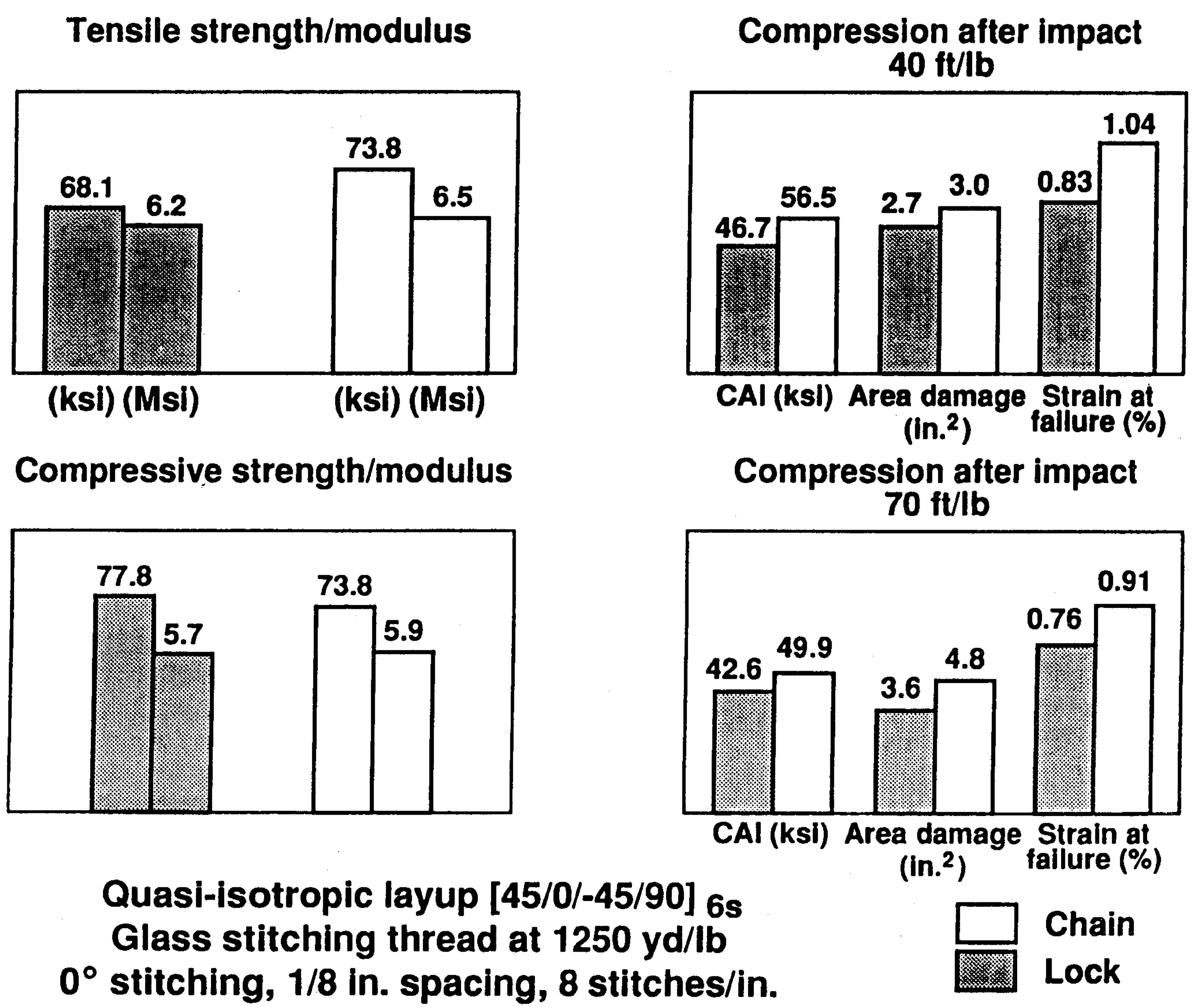


Figure 28. Comparison of Stitch Types

Baseline results were available from the "stitch type" tests which used a cloth with 95 percent AS4 fibers. An improved cloth with 97.5 percent AS4 fibers was used for the four other threads. Strength and stiffness data from the tension and compression tests are shown in Figure 29. The values are the average from three test specimens. The best results were obtained from laminates stitched with Kevlar thread. In laminates stitched with glass threads, the strength values decreased with increased thread weight. Strength and modulus values obtained with the three lightest stitching threads (Kevlar-29, 1875 glass and 1500 glass) show considerable improvement over the values obtained with the baseline 1250 glass thread shown in Figure 28. Part of the property improvements is attributed to lighter stitching threads, which causes less crimping and breaking of the carbon fibers. The remainder is contributed by the carbon fabric itself since this has fewer fiberglass fill yarns which again reduces fiber crimping.

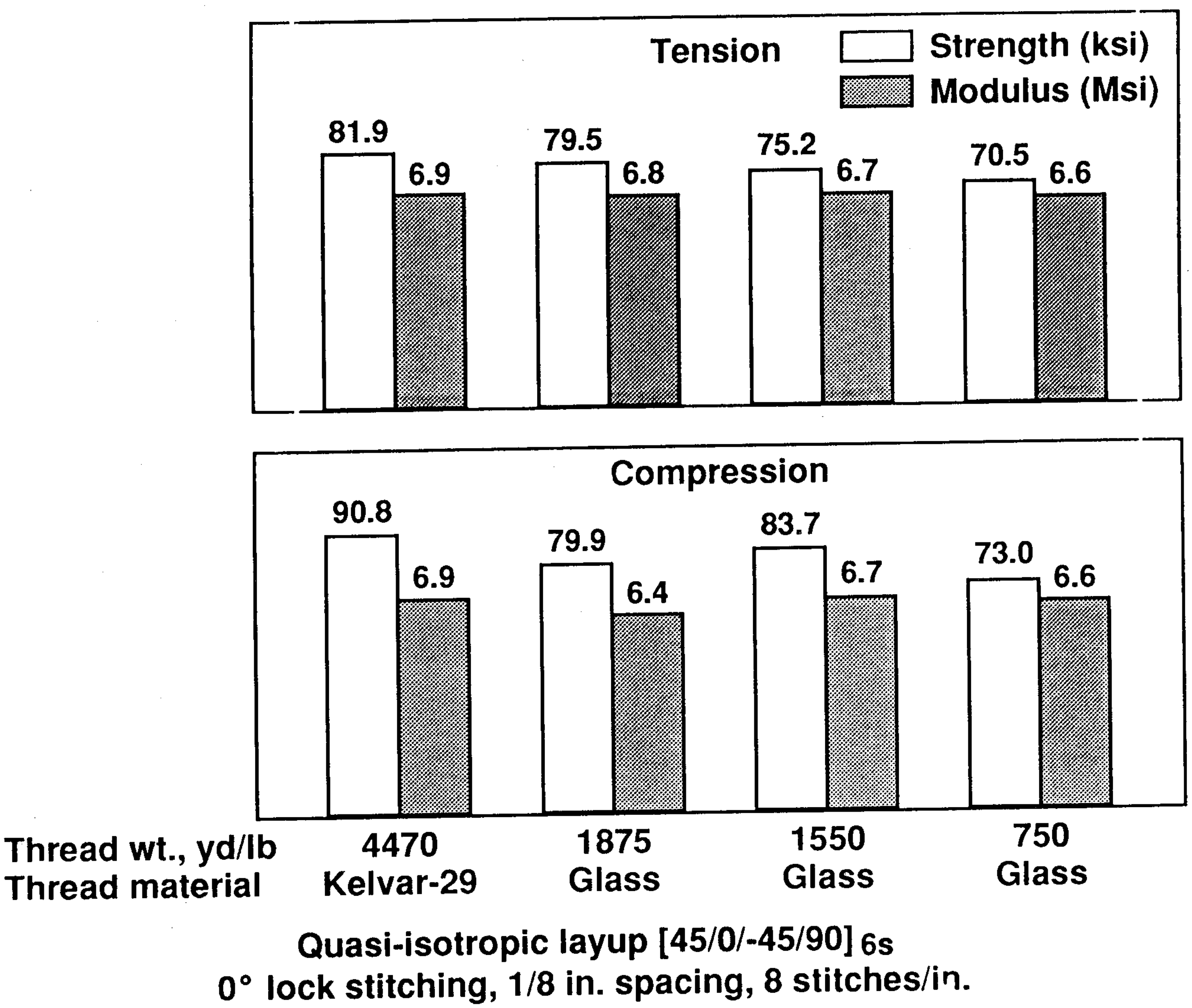


Figure 29. Comparison of Stitch Threads

Otner considerations in thread selection were stitching density, the fact that Kevlar threads are most favored by the machine operator, and that stitching cost favors a lower number of penetrations per inch. After assessing all of the relevant factors there appeared to be little justification to change from the existing 1250 glass baseline thread and this thread was selected for the remainder of the stitching assessment and for the initial phase of the contract effort.

An investigation was conducted to determine whether stitching pattern had an influence on material behavior. Earlier in-house studies with crossed lines of stitching at 0/90° and ±45° to the load direction showed that no advantage could be gained from stitching in two directions. This program concentrated on variations of stitch spacing and stitch step length for lines of stitching at 0° and 90° to the load direction. The following stitch pattern types were evaluated:

Pattern Number	Row Spacing Inch	Stitch Step Inch	Row Direction Degree	Penetrations Per Sq. Inch	Stitching Density Lb/Sq. Inch
1	1/8	1/8	90	64.0	7552
2	1/8	1/6	0	48.0	5664
3	1/8	3/16	0	42.7	5035
4	3/16	1/8	0	42.7	5035
5	1/4	1/8	0	32.0	3776
6	3/16	3/16	0	28.4	3356
7	3/16	3/16	90	28.4	3356
8	1/16	1/8	0	128.0	15104

Tension and compression testing was conducted on specimens having these pattern types, and average results for the 0° specimens are shown in Figure 30. At the extreme high density of 128 penetrations per square inch there was a significant reduction in properties. For the two patterns with 43 penetrations per square inch, the 3/16 row spacing gave better results than those for 1/8 spacing.

In assessing the results of these tests, stitching cost and convenience were factors. Certain stitching machines, for example, can only conveniently stitch in the 90° direction. One very significant observation during testing was that fiber crimp appeared to make a greater contribution than fiber breakage to the reduction of modulus and tensile strength. In Figure 31, it can be seen how fibers perpendicular to the stitching direction are pulled sideways by the stitches. That this is not so much a problem with 0° fibers is confirmed by the fact that 0° stitches performed better than those at 90°. Taking all factors into account, the decision was made to select 3/16" spacing at 0° and a stitch step length of 1/8" for future work in this phase of the contract.

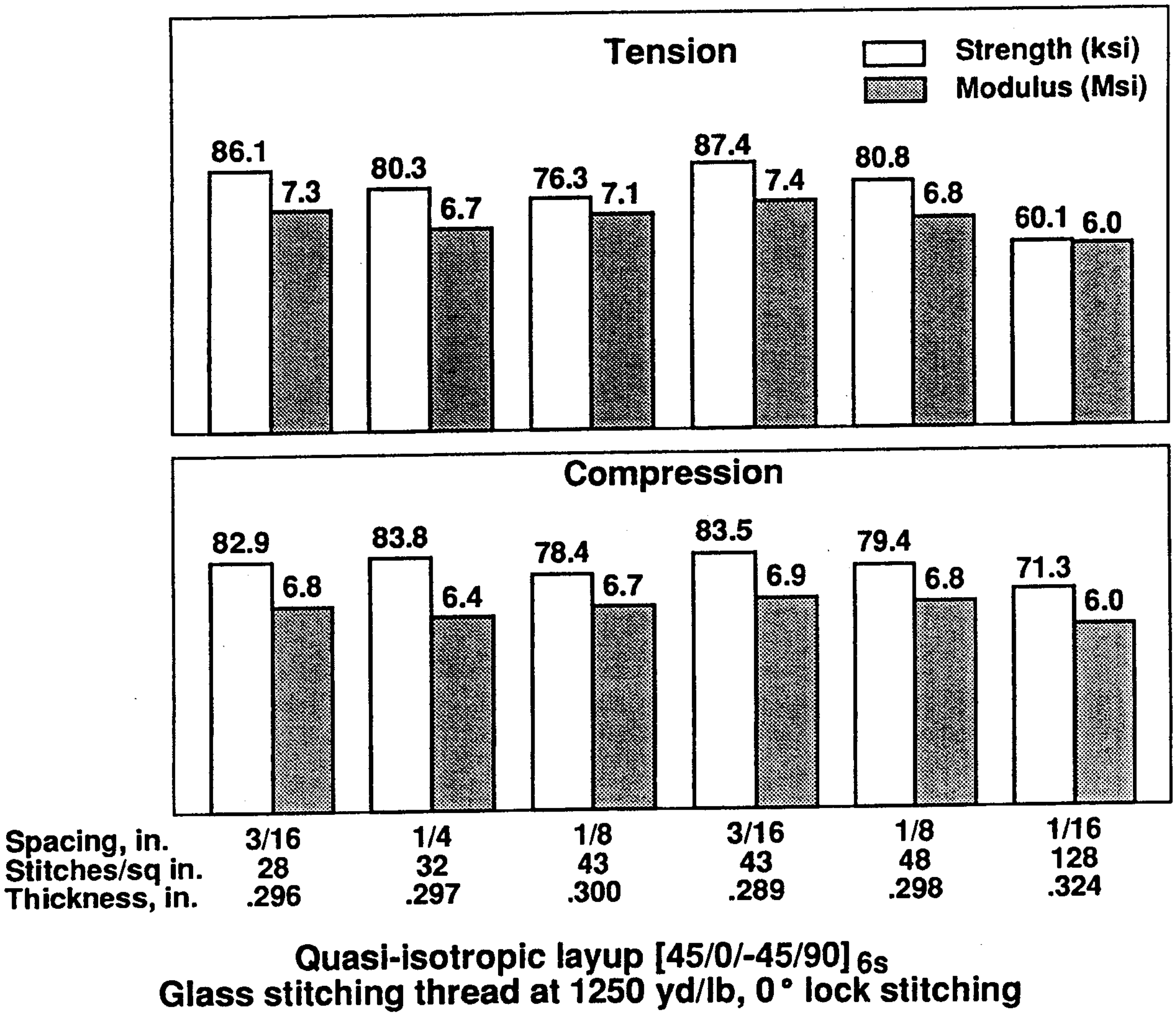


Figure 30. Comparison of Stitching Densities

5.2 Strength and Modulus

It was apparent that the effect of crimping, and therefore the loss of properties, affects the surface plies much more than it affects the plies in the center of the laminate. It followed also that a thick laminate should have better properties than a thin laminate. In order to formulate a modified laminate theory for stitched material, it was necessary to account for the effect of thickness and also the variation of properties between the 0° and 90° stitching directions. Tests were conducted to address some of these problems.

The effect of stitching on monolayer properties was characterized by testing tension (1"x9") and compression (1.5" x 1.75") coupon specimens. High density stitching, with 3/16 row spacing and 1/8 step length, was applied at 0°, 90° and 45° to the fiber direction. Non-stitched panels were also tested to provide a basis for comparison. Derived monolayer properties used for design purposes are listed below:

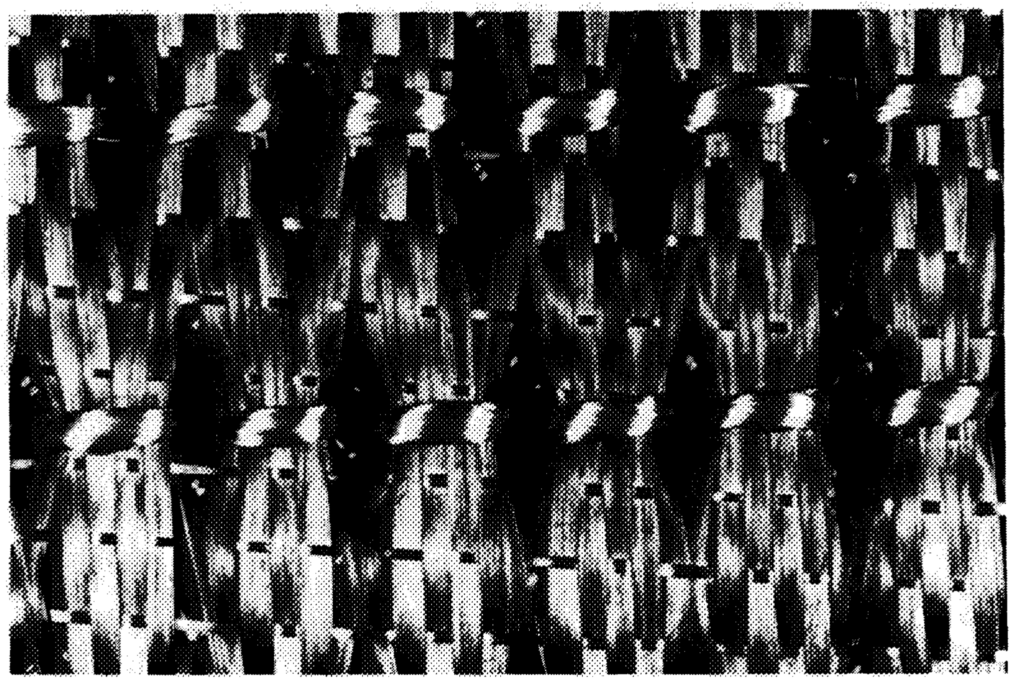


Figure 31. Fiber Crimp Due to Stitching

Monolayer Property	Unstitched Laminate	Stitched Laminate Stitch Angle to Fiber Direction 0°	45°	90°
E_L(Msi)	20.1	17.8/18.0	17.7	17.5
E_T(Msi)	1.7	1.6	1.6	1.6
G(Msi)	0.9	0.8	0.8	0.8
ν	0.37	0.34	0.34	0.34
F_{LT}(ksi)	250	221/225	195/200	175/185
F_{LC}(ksi)	165	138/145	126/140	121/135
F_{TT}(ksi)	4.0	5.0	5.0	5.0
F_{TC}(ksi)	15	31	31	31
F_S(ksi)	15	17.5	17.5	17.5

All property values are normalized to 5.5 and 6.0 mil ply thickness for unstitched and stitched laminates respectively. The extra thickness for the stitched laminates is accounted for by the stitching threads and the lesser compaction and does not imply that there are a greater number of carbon fibers. Where two values are given for a particular property, the first is used for laminate thicknesses of 16 to 30 layers, and the second is for thicknesses greater than 30 layers. One conclusion drawn from this testing was that the preferred stitching direction is along the major load direction.

The above design monolayer values were used to predict tension and compression properties for a 16-ply 0/90 laminate, a 48-ply quasi-isotropic laminate, and a 54-ply (0/45/0/-45/90/-45/0/45/0) wing skin laminate. The results shown in Figure 32 gave close agreement on modulus values but under-estimated strengths in tension and compression by as much as 15 and 30 percent respectively.

5.: Compression-After-Impact Strength

A series of tests were conducted on 48-ply quasi-isotropic laminates to study the effect of stitching variables on compression-after-impact (CAI) strength. Specimens were impacted by a dropped weight at energy levels of 40, 70 and 100 ft.lb. to represent severe damage conditions. In one set of tests, a number of different thread types were used to apply modified lock stitching at 1/8 spacing and with 1/8 step length, in the 0 degree direction. Results shown in Figure 33 confirmed that CAI strength increases with increasing stitching density as anticipated from earlier testing (see Figure 25).

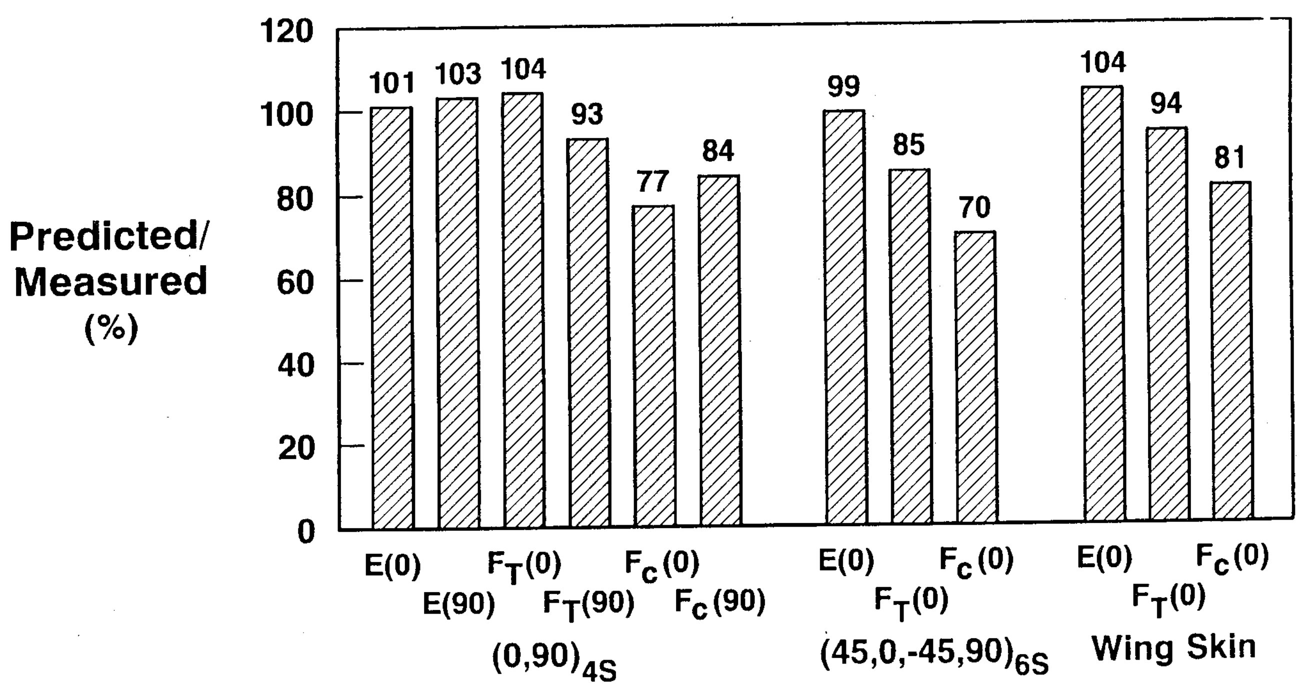


Figure 32. Predicted vs. Measured Laminate Properties

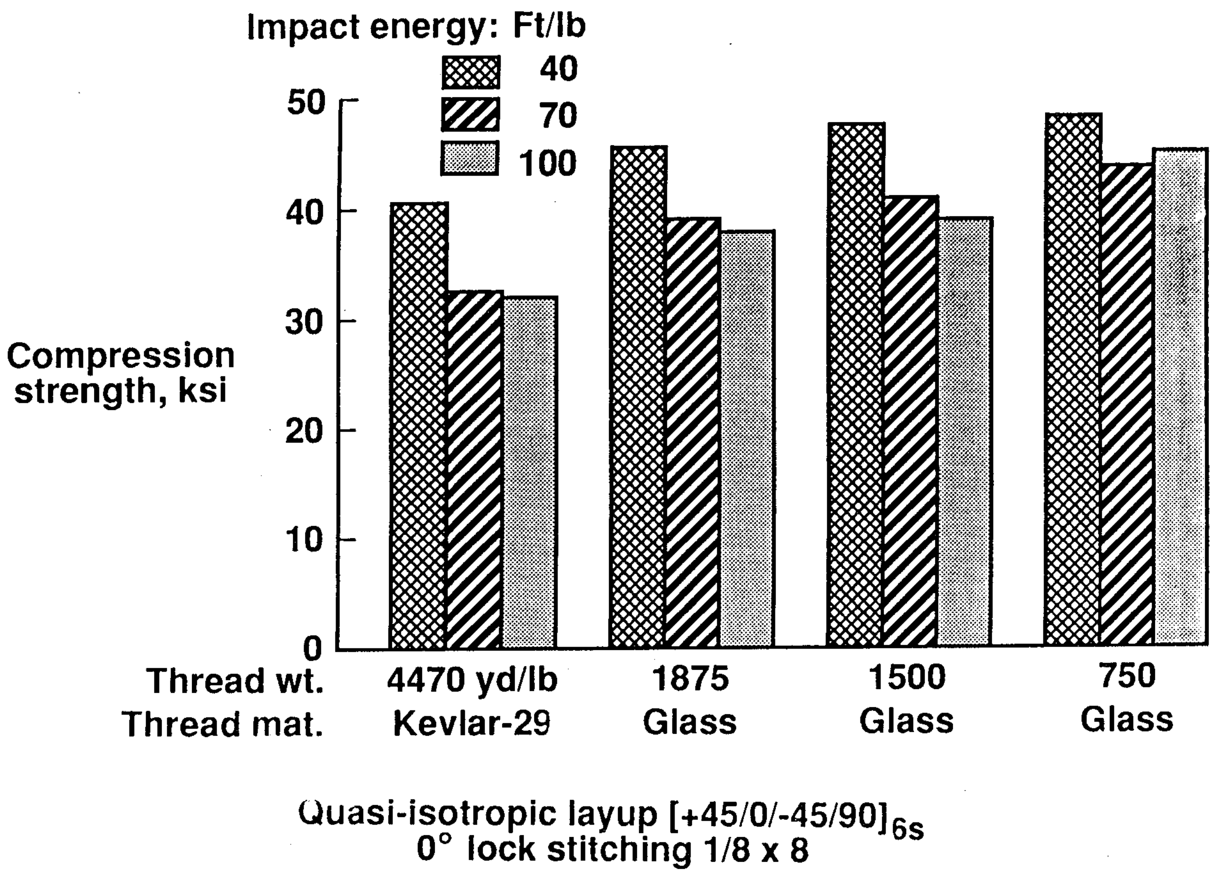


Figure 33. Influence of Stitching Thread on CAI Strength

This trend was confirmed in a second set of tests in which a 1250 yd/lb glass stitching thread was used at a variety of penetrations per square inch (Figure 34). The highest stitching density columns resulted in an impressive CAI strength of 55 ksi at an impact energy of 70 ft.lb. However, the thickness and hence the weight of the laminate is correspondingly increased also. In Figure 35, these CAI strengths and tension strengths from Figure 30 are divided by thickness to give a better comparison of relative efficiencies.

The results of these tests were combined with those from earlier NASA contract and in-house testing (see Table V), to produce a more complete picture of CAI strength. Lower bound values for each impact energy level show a fairly uniform pattern as shown in Figure 36. Taking these curves, normalizing for thickness, and applying a carpeting technique to smooth out the discrepancies, the design chart in Figure 37 was produced. In the extremities of the chart where few test points are available, accuracy can not be guaranteed. In the region where most testing was conducted, the chart can be expected to give conservative results for design purposes.

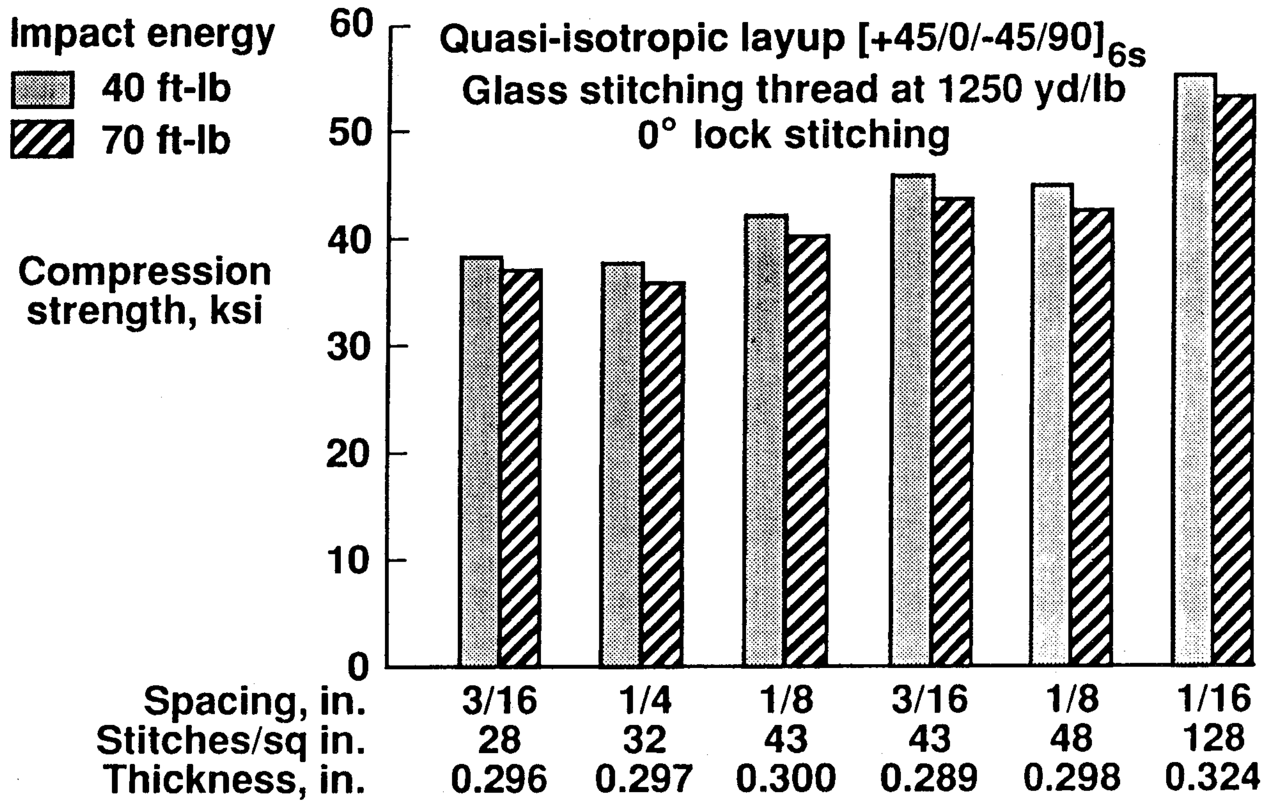


Figure 34. Influence of Stitching Density on CAI Strength

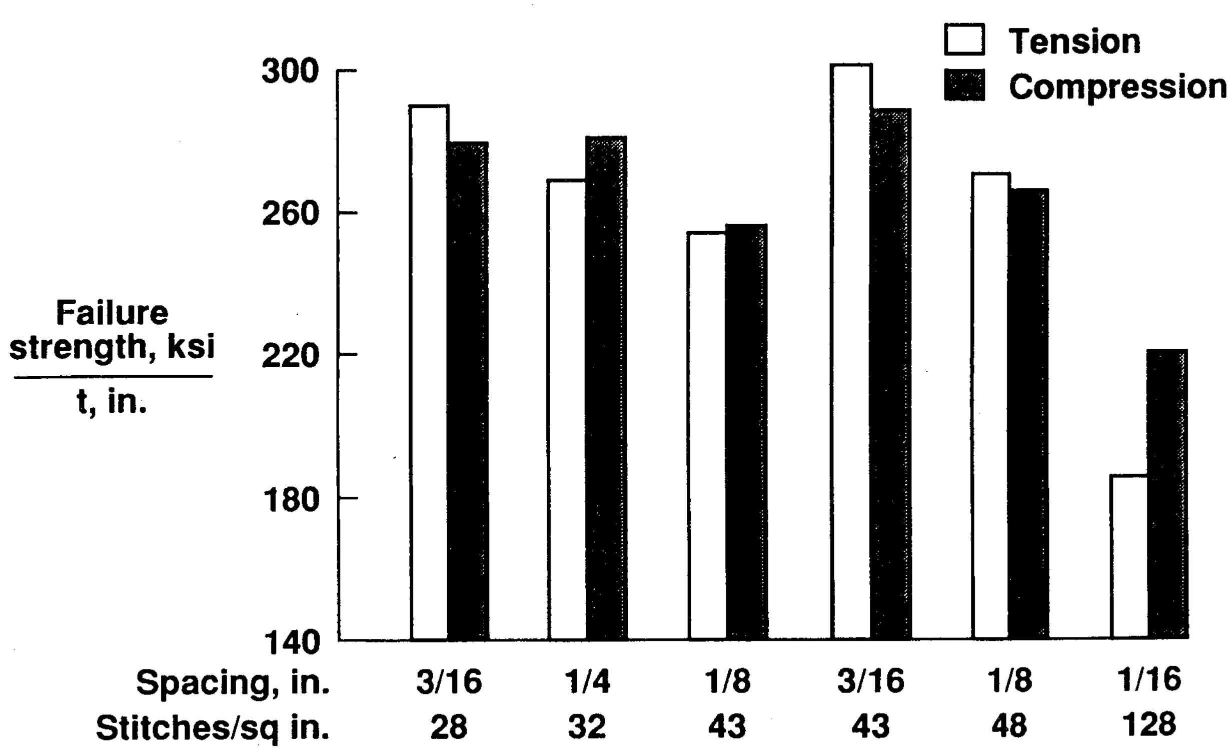


Figure 35. Comparison of Strength/Thickness Values

Table V. Prior Stitched CAI Results

STITCHING DESCRIPTION						CAI STRENGTH (PSI)		
	ROW SPACING (IN)		STEP	PENETRATIONS	DENSITY	IMPACT ENERGY (FT-LB)		
THREAD	0°Rows	90°Rows	LENGTH (IN)	PER IN^2	LB/IN^2	CONTROL 0 (4)	20 (5)	50 (5)
None				0	0	74,900	25,000	18.900
Kevlar 29 (1)	1	1	1/16	32	384	68.000	26,400	20.800
2-End	1/2	1/2	1/16	64	768	60,500	37,400	26.200
Twisted 200d	1/4	1/4	1/16	128	1536	72.600	41,200	37,200
Kevlar 29 (1)	1/4	1/4	1/4	32	1712	68,200	49,800	40.000
1500d	(2) 1/4	1/4	1/8	64	3424	62,500	56.400	48.800
Untwisted	(2) 1/8	1/8	1/8	128	6848	67.000	55,200	52.800
200d	1/8	1/8	(3) 1/8	128	6848	65.000	55.500	52.600
	1/4	1/4	(3) 1/8	64	3424	65.200	48.600	44.600
	1/8		1/8	64	3424	76.800	52.400	47,400
		1/8	1/8	64	3424	73,800	51.500	43.200
T-900 1k	1/4	1/4	1/8	64	2508	71.100	54.400	44,400
4-End Twisted	1/4	1/4	1/8	64	2508	70,700	50,300	38,100
Kevlar 29 (1) 1796d Scoured Twisted	1/4	1/4	1/8	64	2304	67.700	45,100	39,700
Glass 52-467	1/4	1/4	1/8	64	3776	65,000	51,700	51.700
1250	1/8		1/8	64	3776	66,200	56,900	49,800
Untwisted	(2)	1/8	1/8	64	3776	70,700	56,900	49.800

(1) Kevlar 29 aramid fibersmanufactured by E. I. Dupont de Nemours and Co. In.
(2) Data from DAC/NASA Contract NAS1 - 17701
(3) Chain stitch (others are modified lock stitch)
(4) Average of three tests
(5) Single data points
Basic panel is AS4 uniwoven / 3501-6 $(45/0/-45/90)_{6S}$

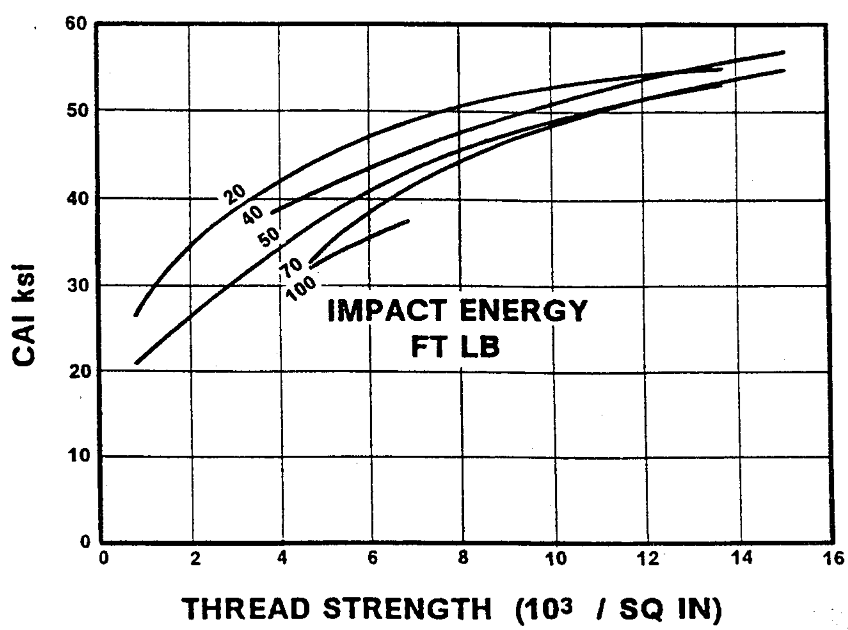


Figure 36. Lower Bound CAI Strength Values

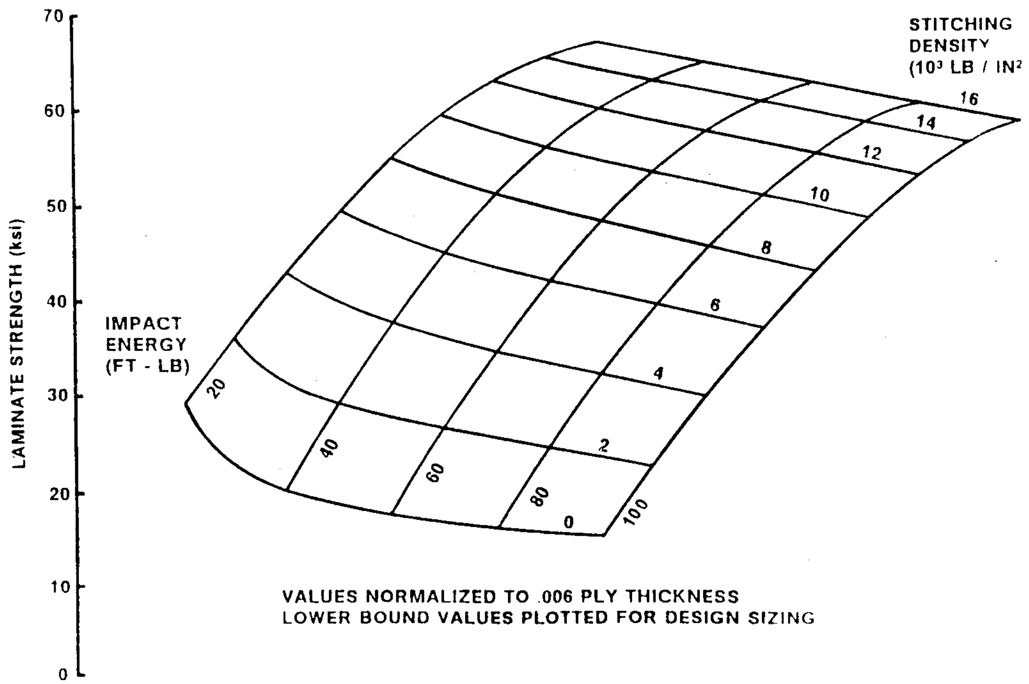


Figure 37. CAI Strength Design Chart

5.4 Stitching Machines

An intensive survey of available stitching machines and support equipment manufacturers in Germany, Switzerland, and France, as well as in the United States, identified the following fourteen suppliers as having the potential capability of meeting our requirements.

1)	ASI Robotic Systems	Jeffersonville, Indiana
2)	Chicago Sewing Machine Exchange	Chicago, Illinois
3)	Gribetz International	Sunrise, Florida
4)	Hauser	Inman, South Carolina
5)	Ideal Equipment	Montreal, Canada
6)	Jentschmany AG	Zurich, Switzerland
7)	Mitsubishi Electric Sales	Irving, Texas
8)	Pathe	Irvington, New Jersey
9)	Puritan Industries	Avon, Connecticut
10)	Sauer Textile Machine Group	Greenville, South Carolina
11)	Sotexi	Paris, France
12)	Tatum Textiles	Louisville, Kentucky
13)	TD Quilting Machinery	Los Angeles, California
14)	The Charles Stark Draper Laboratory	Cambridge, Massachusetts

A preliminary specification for a multi-needle stitching machine and a request for cost and delivery data was sent to each company. Ten companies responded and four of these indicated their willingness to bid. A final specification was sent to the three companies that submitted technically acceptable proposals. A contract was eventually awarded to the Pathe Company.

It was found necessary to purchase two separate machines, one for broad area stitching and one for more detailed automated work. The characteristics of the two machines are given below·

Multi-Needle - Lock Stitch
- 128 Inches Wide
- 256 Needles Available
- Mechanical Stitch Pattern Control
- Speed 200-500/Minute (depends on thickness and stitch thread)
- Thickness Up to 0.5 Inch

Single-Needle - Lock Stitch
- Working Area: 9'x15'
- Full X-Y Computer Stitch Control
- Thickness Up to 1 Inch
- Speed 200-2,000 Penetrations/Minute (depends on thickness and stitch thread)
- Vertical Clearance: 4 Inches

These two machines are depicted are in Figures 38 and 39 respectively.

Note: On the basis of experience gained in stitching with these machines, the penetration thread baseline became 1600d Kevlar 29. The stitching row spacing was increased from 3/16 inch to 0.2 inch.

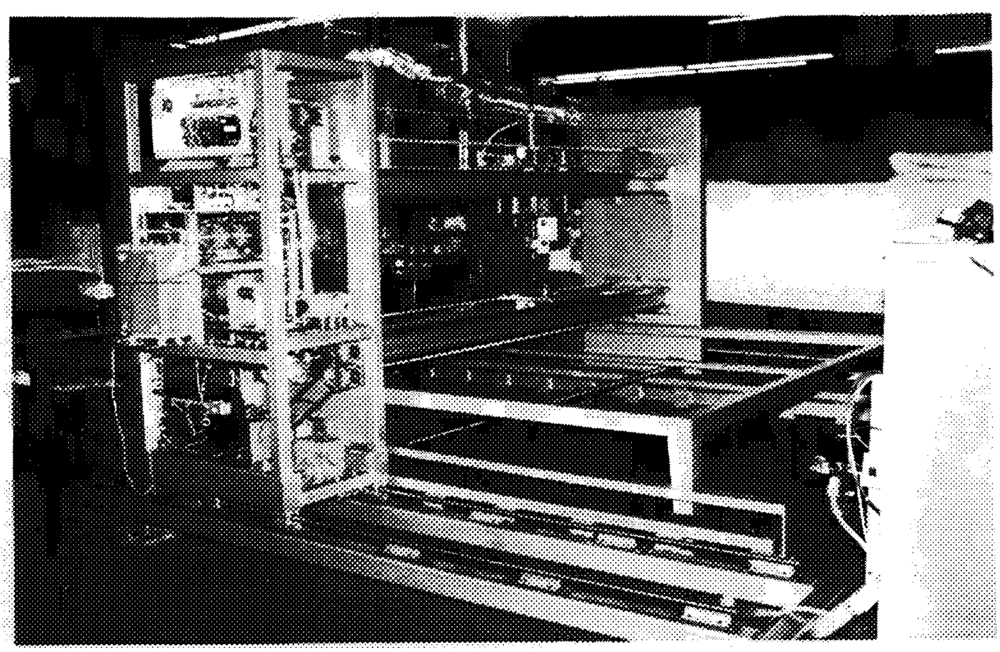
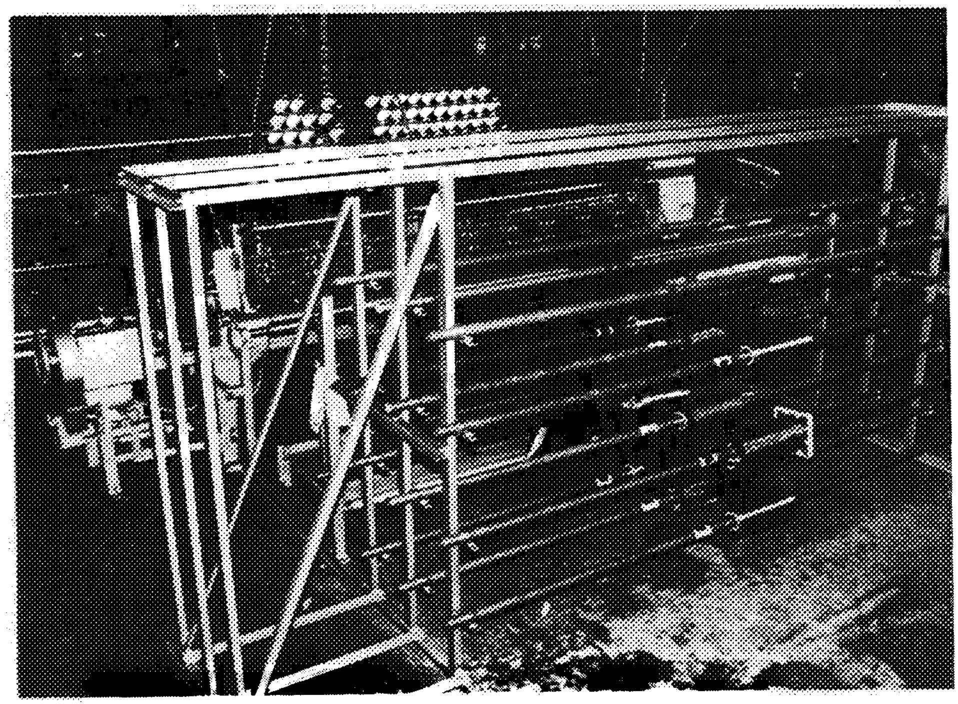
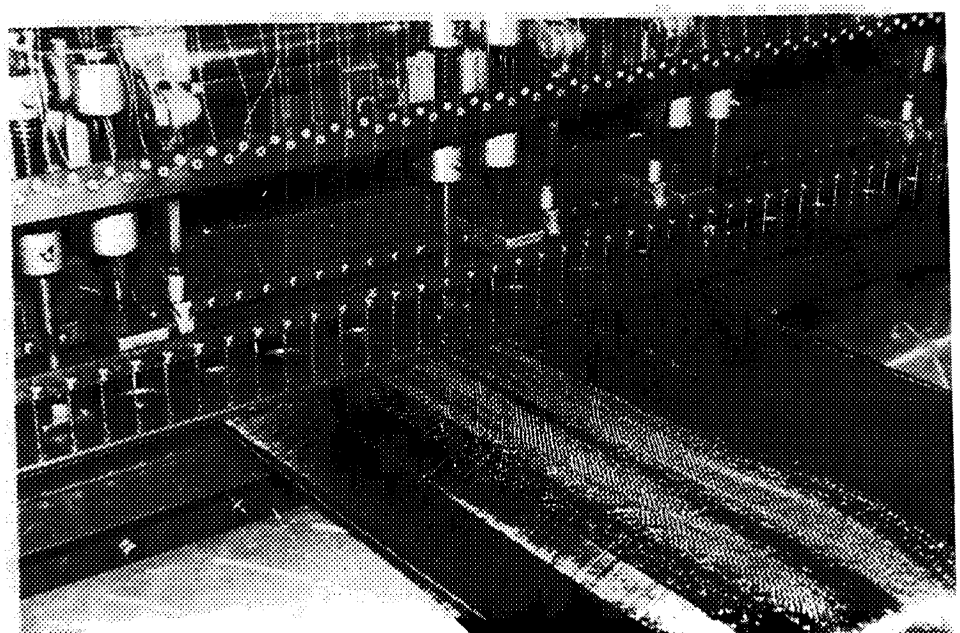


Figure 38. Multi-Needle Stitching Machine



a) STORAGE RACK FOR ROLLS OF MATERIAL



b) MATERIAL PASSING THROUGH STITCHING HEAD

Figure 39. Single Needle Stitching Machine

SECTION 6
RESIN TRANSFER MOLDING DEVELOPMENT

Resin transfer molding (RTM) has the potential to achieve a major breakthrough in reducing the cost of primary structural components for large transport aircraft. In this process, it is necessary to assemble a dry fiber preform of near-net shape, into and around which the resin is introduced by transfer molding. The preform material has to be effectively tacked together to allow it to hold its shape and be handled easily.

The dry fiber preforms may be impregnated by creating a vacuum inside the tool and using differential pressure to push the resin in, or by driving the resin in with external pressure. Both of these methods are being intensively researched in this program. Vacuum impregnation was selected as the principal method for wing components and pressure impregnation was specified for the thinner, more difficult to fabricate, fuselage panels.

6.1 Thermoformable Preforms

One method of tacking the preform layers together is by the use of binder material systems. An industry survey was conducted to evaluate the effectivity of the materials available for this purpose, the processibility of these materials under RTM conditions, and the chemical compatibility with the basic epoxy resin system. Three binder systems were selected for evaluation. Two of these, Dow Tactix 226 and Shell RSS 1630, were non-catalyzed high molecular weight Bis-Phenol-A epoxy resins, supplied in a pulverized (powder) form. The third binder was the Hexcel XC 1144 fabric preform system having Celion 6K carbon-epoxy warp fibers and light fiberglass fill fibers. These fill fibers are coated with approximately five percent by weight of a hot melt thermoplastic binder resin.

The three binder systems were compared by making a variety of 18 in. x 18 in. x 12-ply panels, with the fuselage skin layup. For the Dow Tactix 226 and Shell RSS-1630 binders, twelve layers of AS4 3K fabric were cut to size and individually weighed. These layers were stacked ply by ply to the prescribed fiber pattern and sprinkled with approximately 3 percent by weight of the dry powdered binder between adjacent plies. The 12-ply segment was placed under a vacuum bag and heated to 125°F and then cooled to room temperature. The binder melted and flowed enough to bind and hold the material together when cooled and compacted.

Prior to impregnation with Tactix 138/H41 resin, the resin and the tool were preheated at 150°F for 20 minutes and at 250°F for 40 minutes to allow optimum flow of the resin. During impregnation, resin was allowed to flow from four corner vents independently until the resin stream was bubble-free. After closure of all four vents, additional resin was pumped into the cavity until a hydrostatic pressure of 40 psi was achieved. The entire impregnation cycle took 10 minutes and the parts were then cured at 350°F for two hours. The cured parts contained approximately 59-percent fiber volume and had excellent visual quality. The measured mechanical tension and compression properties of the Shell RSS 1630 binder panels were about 10 percent below those for the Tactix 226 binder panels.

The panels made with the same RTM resin and Hexcel binder system with 6K Celion fibers proved to be more impervious to flow than the 3K fabric used with the other binders. After being subjected to a full heat and compaction cycle, the binder appeared to act as a flow dam, restricting flow in all directions. This resulted in extremely high pressures in the resin and considerable fiber wash in the panels. Panels were also made by vacuum impregnation but the same problem was encountered. The measured tension and compression properties were close to those of the baseline AS4/3501-6 system. The material did have good handling and ply-cutting characteristics, but it was dropped from further consideration because of the difficulty of processing.

A 23in. X 25in. 54-ply AS4 carbon-epoxy panel was then made with the Dow binder and molded with Shell 862 resin, using the same processing technique as before. For this thickness of panel, the plies slipped against one another when the preform was handled and it was concluded that the binder process was better suited to smaller and thinner parts. The mechanical properties were similar to a nonbinder panel, of the same material. In both cases, the mechanical properties were generally lower than the same panel fiber construction made by the autoclave curing process with Hercules 3501-6 as shown below.

	Tensile Strength/ Modulus	Compressive Strength/ Modulus
AS4/3501-6	133.2 ksi/9.7 Msi	84.1 ksi/9.4 msi
AS4/862/226	109.9 ksi/9.9 Msi	93.4 ksi/9.1 msi

Questions about the chemical effect that the Dow binder system may have on the epoxy resin were posed to representatives from Dow Chemical. They stated that the chemistry effect of the binder and the base epoxy resin are so similar, that there could not be any detectable chemistry effect of the binder on the resin system. However, if a significant amount of binder remains within the panels (i.e. not washed out during the bleeding process), it could affect the fiber/matrix interfacial properties. Liquid chromatography (LC/GPC) and photo electron energy loss spectroscopy (PEELS) tests could be used to determine how much binder remains in the panel after processing, and if the binder does in fact build up around the fiber/matrix interface. However, this was not investigated further.

Of the combinations tested, the Dow Tactix binder used in conjunction with Dow Tactix 138/H41 impregnation resin was the preferred system. This combination had no processing difficulties and exhibited no drop in material properties. However no further work was conducted on binder materials because stitching was judged to be a better way to make the pre-plied stack material.

6.2 Vacuum Impregnation Molding

The advantages offered by the vacuum impregnation molding process, compared with the normal fabrication approach using prepreg material, are:

- o lower potential material costs
- o no 0° storage of prepreg
- o no resin toxicity problem (no direct human contact with resin)
- o high quality void free fabrication
- o low facility costs
- o lower quality control costs (no "B" stage material)

The basic process, shown schematically in Figure 40, starts by placing the fiber preform in place on the tool. A vacuum bag is placed over the preform and a vacuum source, attached to one end of the tool, is activated to evacuate the air in the bag. This creates a differential pressure condition that forces liquid resin from an external container, through an impregnation sprue and into the opposite end of the tool. The resin then flows through the cavities in the tool, from one end to the other, to impregnate the preform directly on the tool. The cure cycle is completed under vacuum bag pressure and heat only. Complete impregnation occurs as a result of kinetic advancement of the resin until final gelation is accomplished. This process is covered under a DAC patent (Reference 12).

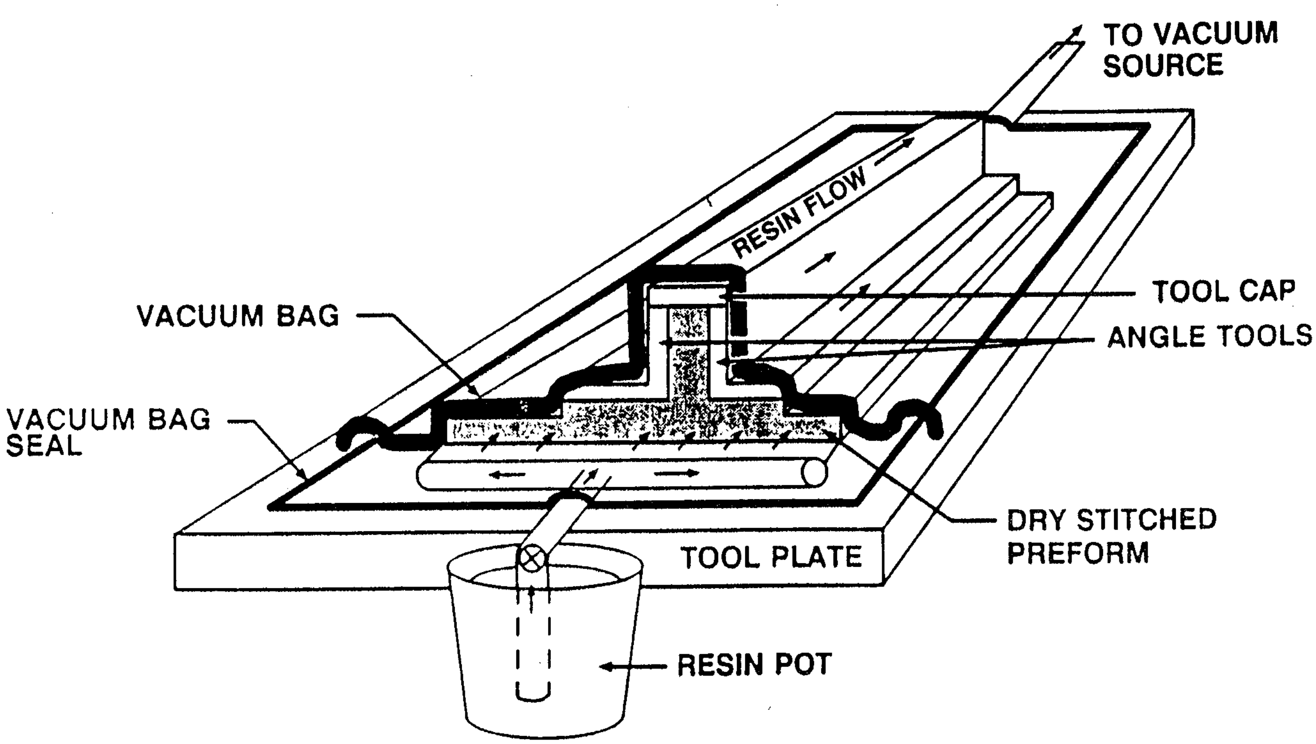


Figure 40. Resin Vacuum Impregnation

The disadvantage of flowing resin along the length of the tool is that it will only flow so far and therefore there is a limit to the size of parts that can be made by this method. This problem can be overcome if the resin can be fed quickly over the entire surface of the part so that it then has to flow through the depth of the part only. To accomplish this, the tool is supplied with a mold surface caul plate having a resin runner system from a central inlet source on the under side. The size of the runner and the diameter of the access holes through the caul plate control the flow of resin into the preform. A vacuum bag is placed over the preform and the vacuum source attached to the upper side. Liquid resin is drawn from an external container into the center of the lower side of the caul plate and spreads through the runner system and upwards through the access holes to impregnate the dry preform within the tool. The cure cycle can be completed with vacuum bag pressure only or with a combination of vacuum and autoclave pressure.

The major advantages of through-the-thickness impregnation are:

- o resin travels a minimum distance within the preform,
- o the resin spreads in a very few seconds to fill the runner system and impregnate the entire preform, and
- o the runner/access hole system allows a control capability for more, or less, resin in thick or thin areas.

Difficulties with producing an adequate runner system for this process led to the replacement of liquid resin with a solid film of cast resin. This film infusion process has been selected as the baseline method for making wing cover panels. The approach is to weigh the dry preform and cast the resin to have about 36-38 percent of the preform weight. The resin is cast to conform to the planform shape of the preform and is placed under the preform, and in alignment with it, onto the tool surface. Physical dams are created around the periphery of the resin to prevent resin flow beyond those boundaries. Inner mold-line tooling is assembled onto the preform/resin combination and the entire assembly is then double vacuum-bagged and cured under a modified 350°F two-step cure cycle (Figure 41). The amount of resin in the cast film is sufficient to fill the part as it melts under heat and flows up through the part depth. In this way, the amount of resin wasted is kept to a minimum in contrast to the excess resin used with a liquid resin approach.

6.3 Resin Pressure Molding

The use of positive pressure to drive the resin into the preform was selected as the baseline process for the thinner curved fuselage panels. Resin is introduced into the tool through a sprue and vent combination under sufficient pressure for it to flow through the tool and permeate the preform (Figure 42). The choice of pressure is critical because it determines the flow rate of the resin. If the flow is too fast or too slow it may result in porosity or dry spots, or the pot life of the resin might be exceeded. Pressure RTM requires more complex tooling but provides better dimensional control of molded parts. Experience with the tooling development for fuselage element panels has led to a number of changes being incorporated into the tooling design recommendations for larger components. These changes are:

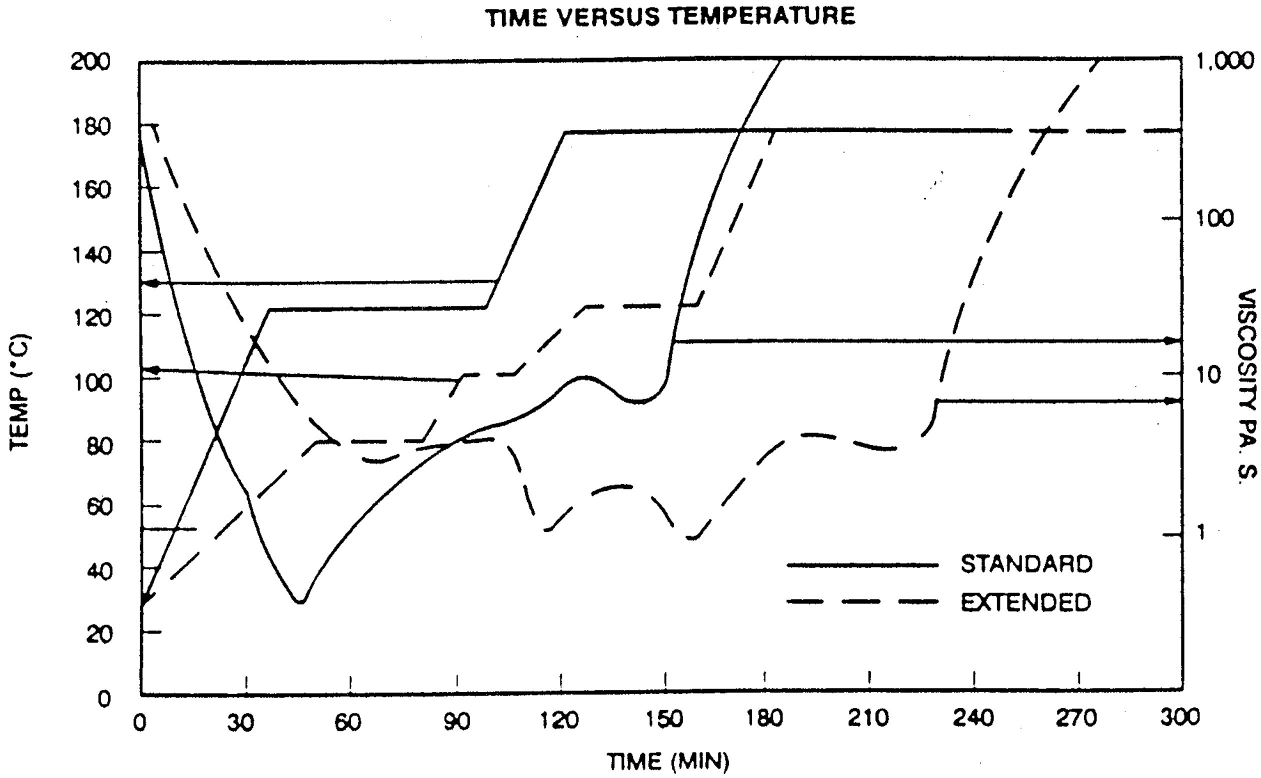


Figure 41. Resin Infusion Process Cycle

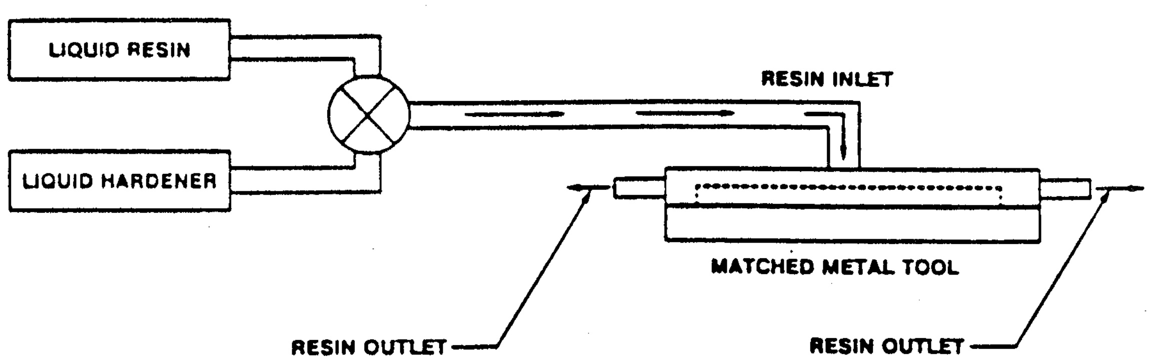


Figure 42. Resin Pressure Molding

1) The tool base should be permanent and act as a foundation for the tool and the restraining fixture. For larger sizes, the idea of two tool halves being joined together and placed in a press becomes too cumbersome to handle.
2) The tool base, which contains and restricts mandrel motion, should be oversized, and a method, preferably set screws, should be incorporated to provide sideways compaction of the preform.
3) Tooling tolerances have a significant effect on resin flow profiles for thin structures. Typically no more than ±0.007 for a 0.072 thick skin can be allowed, without disrupting critical flow patterns.
4) A Moen hot air heating system should replace the expensive and unreliable electric cartridge heaters previously used. The Moen system uses a copper tube network to distribute hot air along the tool base surface. Hot air is generated from a 50 kw heater system that is coupled to the tool distribution manifold. The heater system itself is an independent unit that can be used to heat several tools. A savings of $50,000 is predicted when the integration of the heat system to the tool is included.

It was planned that the upper cavity tool would be a carbon/epoxy reinforced airpad rubber to seal and contain the resin. While metal tool halves, which have to close against stops and uniformly compact the sealant gland, worked well for smaller tools, there was some concern about their feasibility for larger parts. The new proposed cavity could be fabricated directly into the tool to ensure no tolerance mismatch. It would be lighter than metal and would help insulate against the thermal losses that exist in metal tools.

In evaluating this approach, it was determined that the caul sheet was not strong enough to withstand the significant pressures resulting from resin expansion during heat-up to cure temperature. A modified upper tool cavity concept was then developed to overcome this problem (Figure 43). The fuselage subcomponent tool shown has a 60" x 80" carbon/epoxy base which is filled with concrete. Semicircular grooves are molded into the 126" radius top of the base to support Moen heater manifolds used to heat the tool.

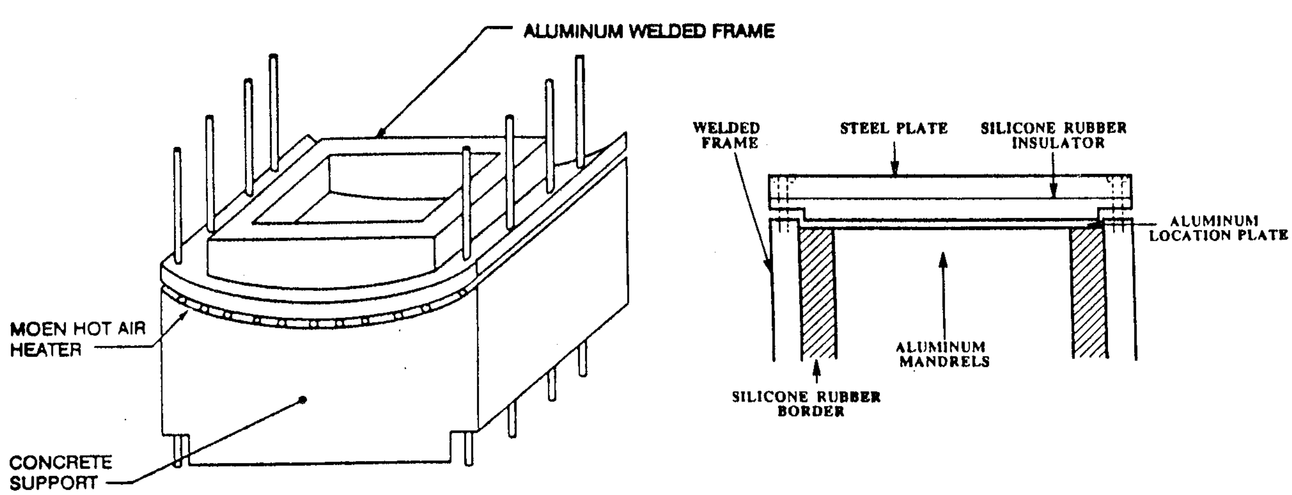


Figure 43. Fuselage Subcomponent Tool Concept

The outer mold line (OML) skin tool rests on top of the tool base and a metal frame is seam-welded to the tool to ensure sealing. This frame is oversized to allow for any unusual expansion effects that might occur. Rubber is cast between the frame and the mandrels to fill in any gaps where resin could collect and cause difficulty in removing the part from the tool. The rubber also generates the pressure necessary to compact the mandrels at impregnation temperature. Sealing is accomplished with a bolted upper plate and backup structure to ensure that deflection is not a problem.

6.4 Resin Evaluation

The baseline resin selected for this program is the Hercules 3501-6 system. This resin is readily available and its characteristics are well understood, but being of an earlier generation, it does not have the toughness of many more recent resins. This deficiency is offset by stitching. The resin has made acceptable parts by the vacuum impregnation process but requires 0° storage and has exotherm problems when used in large quantities in the pressure RTM process.

Many resin suppliers are formulating resins specifically for RTM and these promise many advantages over 3501-6. The chemistry is more simple since there is no prepreg requirement, costs could be much lower, and storage at room temperature is usually available. The objective is to select a resin with these advantages and which has material properties and processing characteristics as good as those for the 3501-6 system. Generally, film and solid material forms are satisfactory for vacuum impregnation, but a liquid form is necessary for pressure RTM. A large number of these new resins have been tested with in-house funding. In this contract program, the following new resins were selected for further evaluation.

Manufacturer	Designation	Form
Hercules (Baseline)	3501-6	Film or Solid
Dow	Tactix 138	Liquid
	Tactix 695	Solid
	Tactix 695	Liquid
	CET-2	Solid
Shell	862	Liquid
	1895	Liquid
BP (U.S. Polymeric)	E-905	Solid
	E-905L	Liquid
Ciba-Geigy	Matramid 5292	Liquid/Solid

The first Dow resin, Tactix 138/HT41, was unsatisfactory because the Tactix panels had high void content and surface porosity, and it was changed to the Tactix 695 resin system. Originally, the 695 system was a two component liquid with a solid hardener, but was changed to a single component material that is solid at room temperature. This resin was claimed to have a long room temperature storage life, and viscosity characteristics that allowed it to be processed by vacuum impregnation. However, it was difficult to consistently produce void-free panels by this method.

A new resin, CET-2, developed by Dow under a separate NASA contract, was then evaluated. This resin is a one-component epoxy system that is solid at room temperature and can be stored at that temperature for at least one year. After considerable development, acceptable 8-ply panels were made with vacuum pressure only, but thicker unstitched 32-ply panels could not be impregnated through the thickness, even under 100 psi pressure. Another disadvantage with this resin was that its specific gravity of 1.77 was much higher than the 1.3 value of other epoxies.

Good quality panels were made with the Shell RSL 1645/RSC 763 (DRL-862) resin, and with a developmental Shell RSL 1895/W resin. Acceptable panels were also made with the U.S. Polymeric E-905L resin by vacuum impregnation. It was found necessary to modify the tool to help the E-905L, a more viscous resin, to flow into a stitched preform. A disadvantage of this resin is its relatively high cost. Panels made with the Ciba-Geigy Matramid 5292 A/B resin suffered from occasional unacceptably high void contents. At the completion of the processing and mechanical properties evaluation, the Shell 1895 resin was selected for pressure RTM.

SECTION 7
MANUFACTURE OF STITCHED/RTM WING PANELS

The successful manufacture of good quality 3-stringer and 6-stringer wing subcomponent test panels represents a major milestone in demonstrating the feasibility of the stitching and RTM approaches being developed in this contract program. Stringers were located along the length of each panel at 7 inch spacing. Both skin and stringers were stitched independently and the stringer flanges were also stitched through the skin to make a complete dry preform of near-net shape. The panels were fabricated by the "Resin Film Infusion" RTM process, using the baseline Hercules 3501-6 resin system and an autoclave cure. Details of the fabrication steps are given in this section.

7.1 Dry Preform

The dry preform was assembled from Hexcel AS4 unidirectional cloth layers. These were first laid up by hand to form the balanced 9-ply stack material selected as the basic building block for wing skins and stringers. The edges of the stacks were taped to maintain the fiber directions throughout subsequent handling operations.

Six of the 9-ply stacks were passed through a Katema manual single-needle stitching machine to form 54-ply skins. The stitching pattern was eight stitches per inch in rows 0.2 inches apart and parallel to the 0° fiber direction. Needle thread was 2-end twisted Kevlar 29 and the penetration (bobbin) thread was 1250 weight 3678d, S-2 glass with an epoxy-compatible finish.

Fabrication of the stringers was accomplished in a similar way by passing eight of the 9-ply stacks through the single-needle stitching machine. The stitching was arranged in bands to allow individual stringers to be cut and folded as shown in Figure 44. The stringer flanges were then stitched to the skin using the single-needle machine. Typical instructions for completing the stitching operation are listed in Figure 45. The dimensional accuracy required to make a high quality preform was controlled within prescribed standards, as shown in Figure 46 for a 3-stringer panel. Accuracy was ensured by specialized tooling created for stitching the skins, stringers and stringer/skin attachments (Figure 47). After inspection of the dry preforms was completed, they were delivered to Douglas for impregnation. A completed 3-stringer panel preform is shown in Figure 48.

7.2 Tooling and Part Preparation

Metal tooling components were made from 6061-T6 aluminum. The tool for each panel consisted of 0.500 top and bottom plates and 2 inch x 2 inch x 72 inch internal mandrels. The top plate was drilled to create a bleed path for excess resin. Silicone rubber bars, 2 inch x 2 inch x 72 inch, were used to apply pressure sideways to the stringer blades (Figure 49). All tooling was thoroughly sanded and given six coats of Freekote 33 mold release agent. The release coats were baked on at 350°F for one hour.

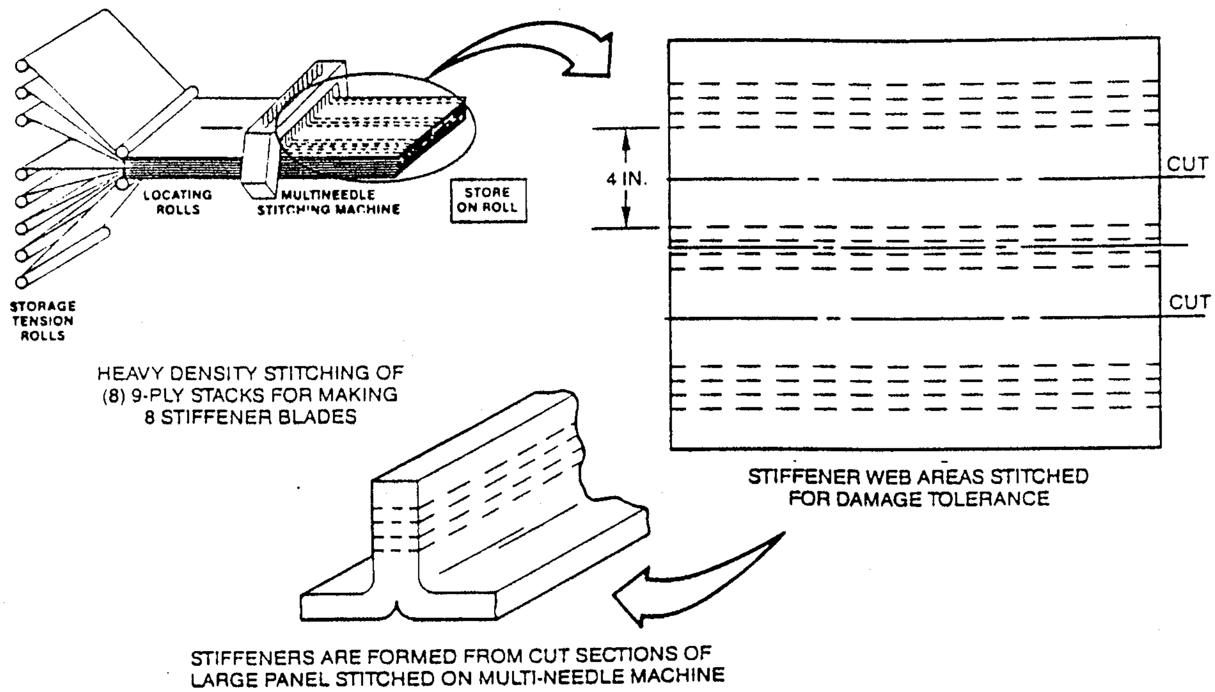


Figure 44. Stringer Fabrication

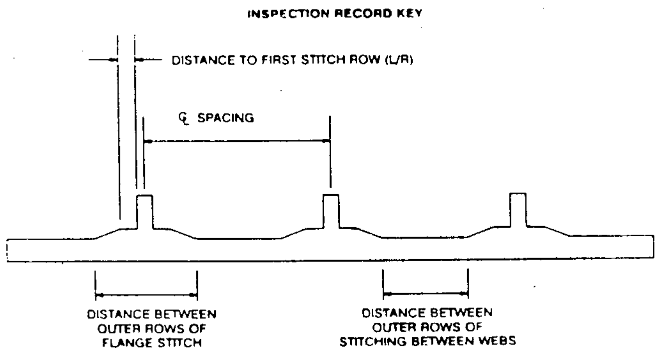
In setting tooling requirements for blade stiffened wing panels to be made by using the stitched preform/RTM fabrication method, a good deal of adaptability, flexibility and forgiveness had to be built into the tooling due to limited knowledge about tool requirements. A schematic of the type of tooling used to make the 3-stringer and 6-stringer wing panels is shown in Figure 50(a). A comparison can be made with the tool for liquid resin infusion shown in Figure 50 (b).

Tooling assembly began by placing the cast resin on the bottom tooling plate and then placing the preform on top of this. The aluminum mandrels were set in place sequentially starting at the center of the panel and working toward the edges. Insertion of the rubber bars between the mandrels required the use of a rubber mallet because of the slightly oversize thickness of the dry stringer blades. After checking that all the tooling elements were positioned correctly, the top tooling plate was set in place.

Two separate steps were taken to seal the perimeter of the tool, the first being mylar tape running between the top and bottom plates. The second barrier was an inner bag, using F.E.P. release film and bag sealant tape. The release film covered the entire perimeter of the part and was sealed off at the top and bottom plates with bag sealant tape. Three plies of 1534 glass bleeder were placed over the part and covered with F.E.P. release film which was sealed to the bottom plate. The release film was pricked with a pin at various locations to allow air to escape, and a final four plies of 1534 glass were applied as a breather. The entire part was then vacuum bagged, checked for leaks, and placed in the autoclave.

OPERATION	DESCRIPTION
Stitch Skin (54 plies 6x9 ply elements)	Stitch 45" x 78" Skin Panel.
Light Stitch Stringers	Light Stitch 9 Ply Stringer Elements On The Pathe Machine Stitch Yarn: Nylon Stitch Spacing: 1/8" step x 1" row spacing
Assemble & Heavy Stitch Stringers (72 plies 8x9 ply elements)	(A) Stack 8 9 Ply Elements From Step #2 Above (B) Heavy stitch Notes: (1) Two stringers are being made at the same time (2) 9 ply elements are symmetrical (0^{0} plies on both faces) (C) Cut into two stringers along centerline (D) Produce a total of 6 stringers
Trim Skin & Stringers	(A) Stitch around periphery of panel (B) Trim masking tape from skin and stringers
Stitch Stringers To Skin	Insert 26 ends of AS4 12K in the void at the base of the stringer Mount stringer in holding fixture and align on frame Peel back the first layer of fabric on the outside ply set and trim the remaining plies to the 1/2" dimension shown Stitch all layers of the first ply set with one row as shown This row is 3/8" or less from the side of the web Bobbin Yarn: Kevlar 29 1500 de Stitch Spacing: 1/8" step Trim the second ply set at a distance of 3/4" from the side of the web. Do not trim the outer ply of the first ply set. Stitch the second ply set with two rows spaced at 1/8" as shown (stitch down the outer ply "flap" of the first ply set during this operation. Trim the third ply set at a distance of 1.0" from the side of the web. Do not trim the outer ply of the first ply set. Stitch the third ply set with two rows spaced at 1/8" as shown (stitch down the outer ply "flap" of the first ply set during this operation. Trim the fourth ply set at a distance of 1.25" from the side of the web. Do not trim the outer ply of the first ply set. Stitch the fourth ply set with two rows spaced at 1/8" as shown (stitch down the outer ply "flap" of the first ply set during this operation. Reverse the tool and stitch the other side of the "T" Stitch the remaining 5 stringers in place a in a similar manner. NOTE: Verify that adjacent stringers are on 7.00" centers before stitching.
Inspection	Perform a dimensional inspection

Figure 45. Typical Stitching Instructions



PROGRAM QUALITY REQUIREMENTS

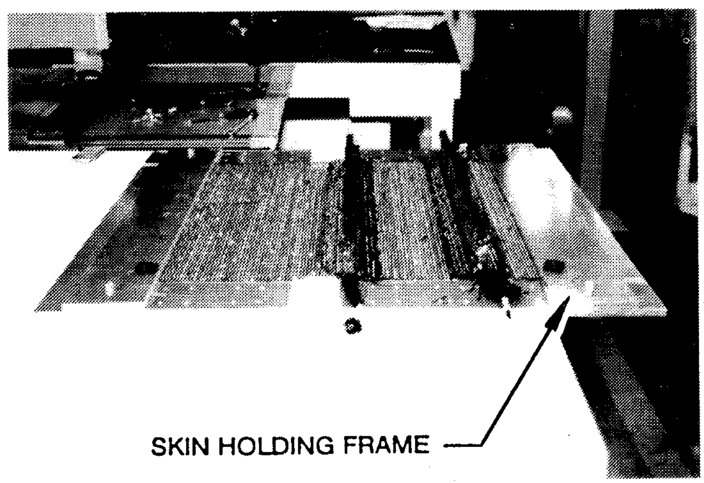
ITEM 1

NO 3SE
RIGHT STIFFENER WAS 8 ROWS OF STITCHING

S/N ________ DATE 4/4/91__ STITCHING STEP AVERAGE -

CHARACTERISTIC	VALUE	LEFT WEB		CENTER WEB		RIGHT WEB		LEFT	RIGHT
		L	R	L	R	L	R		
℄ - ℄ SPACING BETWEEN STIFFENERS	7.0	-		-		-			
DISTANCE TO 1st STITCH ROW	0.37/ 0.44							-	-
DISTANCE BETWEEN OUTER ROWS OF FLANGE STITCH	2.68/ 2.81							-	-
DISTANCE BETWEEN OUTER ROWS OF STITCHING BETWEEN WEBS	4.19/ 4.32	-		-		-			

Figure 46. Panel Dimensional Control



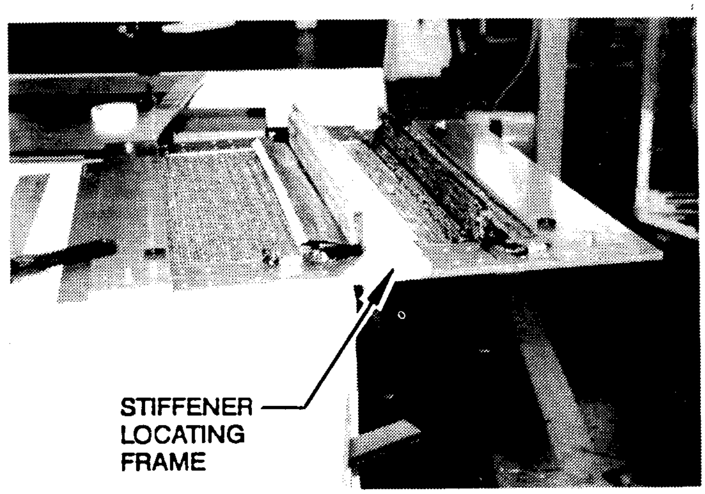


Figure 47. Specialized Tooling Frames

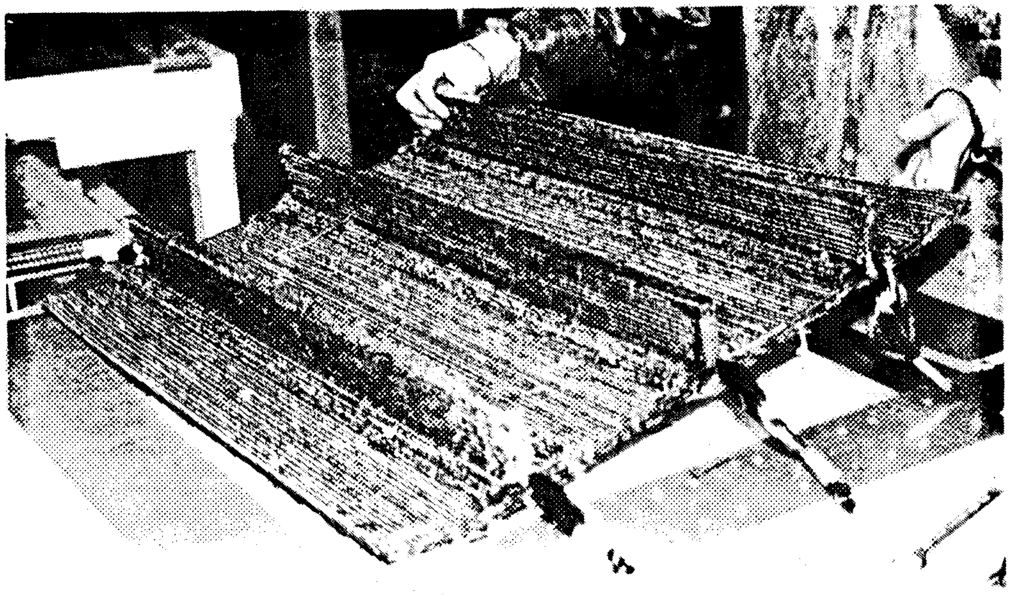


Figure 48. Completed 3-Stringer Panel Preform

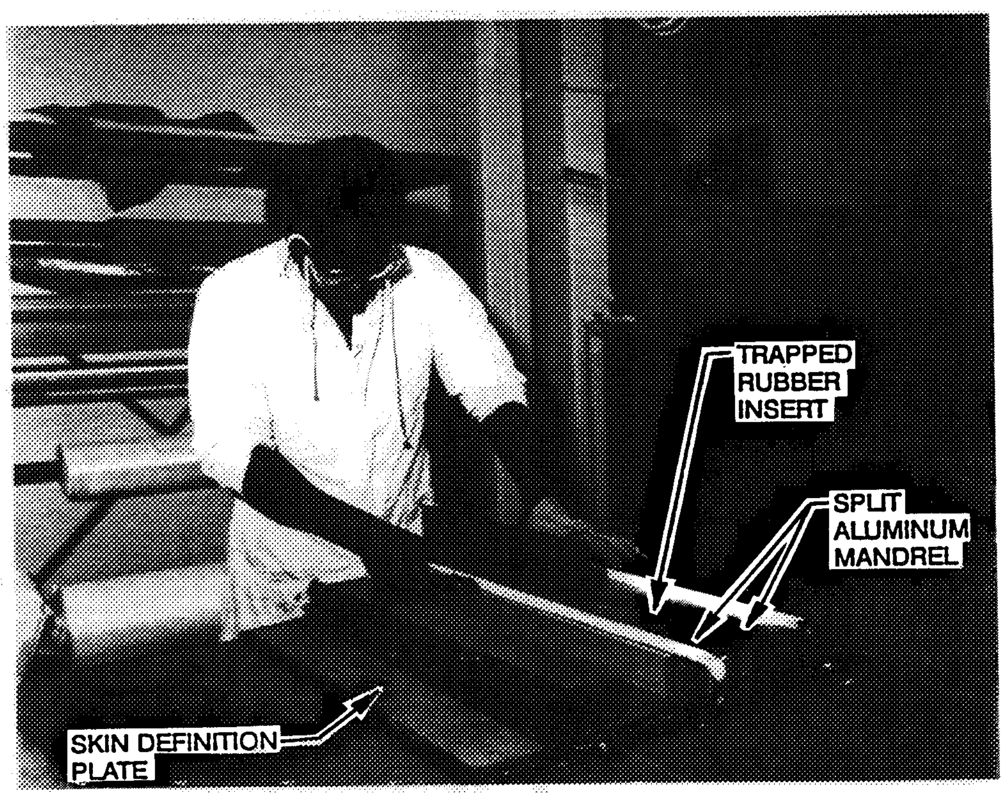


Figure 49. Details of Panel Cure Tool

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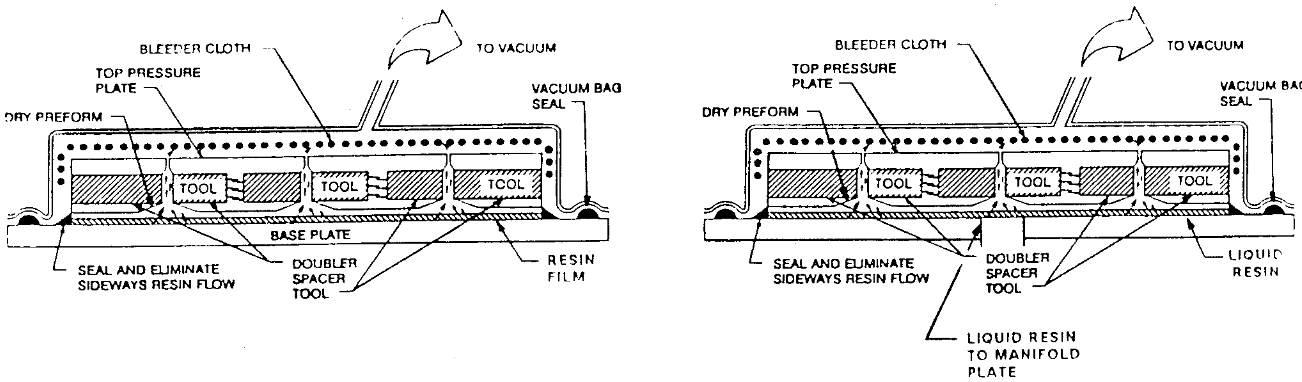


a) FILM INFUSION METHOD

b) LIQUID INFUSION METHOD

Figure 50. Tooling for Vacuum Impregnation Infusion Methods

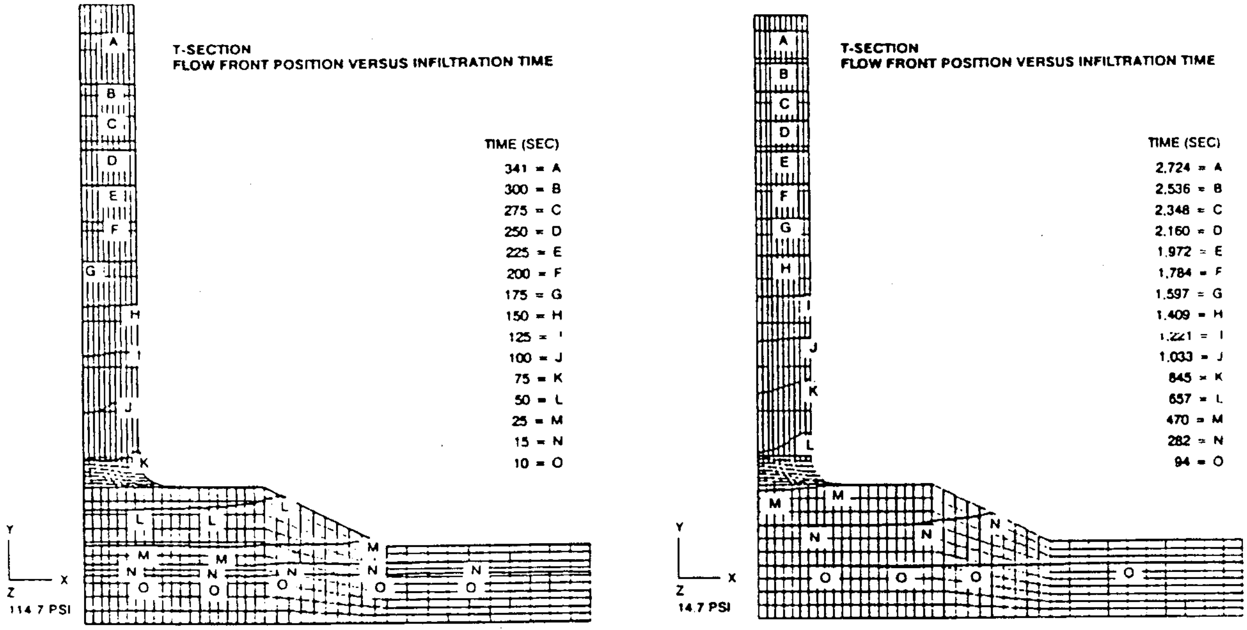
7.3 Impregnation and Cure

In developing a single step resin infiltration and curing cycle, the subcontractor team of Virginia Polytechnic Institute and William and Mary College played a critical role. Findings from their work established that preform thermal equilibrium and application of initial pressure are essential to a single step cure cycle. Flow models (Figure 51) showed that application of 100 psi coupled with a multi-dwell cure cycle promised the best and most expedient results.

AS4/3501-6 panels made with this developed cure cycle were of consistent high quality. The only adjustment necessary was to increase the pressure from 100 psi to 140 psi to account for preform tool mismatches as well as any control differences between the laboratory developed cure cycle and the actual manufacturing application. Figure 52 shows a completed 3-stringer wing panel.

Compression testing was conducted to verify the structural integrity of the panels. A 100 ft-lb impact energy was used to create damage at three critical locations: mid-bay, edge of stringer flange and directly over the center stringer. Results indicated that mid-bay conditions are more critical to residual compression strength after impact. The results below show a comparison of compression strength for the damaged 3-stringer panel with results for similar panels made with toughened resin systems:

AS4/3501-6 (Stitched/RTM) - 550 kips
IM6/1808I (Tape Prepreg) - 460 kips
IM7/8551-7 (Tape Prepreg) - 420 Kips



APPLICATION OF 100 PSI WITH FULL VACUUM YIELDS A RESIN INFILTRATION RATE OF APPROXIMATELY 8 TIMES FASTER THAN THE VACUUM PRESSURE ONLY APPROACH

Figure 51. Typical Flow Model Results

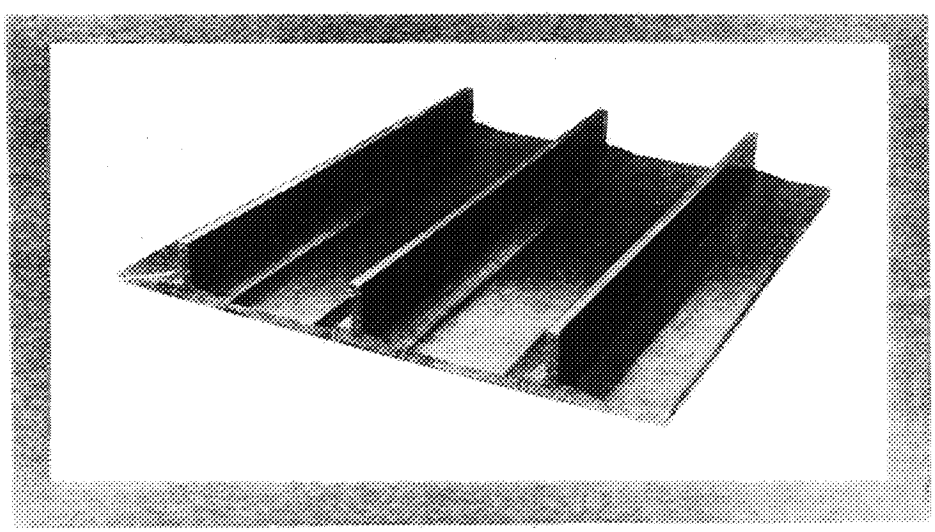


Figure 52. Completed 3-Stringer Panel

Preliminary cost studies and time tracking for fabrication of the small 3-stringer element panels show that a large percentage of the savings from this method of stitched/RTM processing will come from the use of automated stitching machines to make complex 3-D large wing skin preforms. In this way, most cutting, collating and lay-up can be eliminated. Only the processing time will remain equivalent. A comparison of the hours required to make a 2' x 3' panel by RTM/stitching versus hand layup is given below:

RTM / Stitched

(Estimate based on automated stitching machine)

	Manhours
Stitch preform	8.00
Clean tool	6.33
Prepare tool	2.50
Trim preform	2.16
Cast resin	2.00
Assemble tool/preform	7.66
Cure Part	8.66
Trim/machine part	5.00
	42.31

Hand Lay-up

	Manhours
Clean tool	6.33
Prepare tool	2.50
Cut material	20.80
Collate plies	24.18
Bag/unbag	1.50
Cure	5.00
Trim	5.00
	65.31

Fabrication steps for the 6-stringer panel were essentially the same as for the 3-stringer panel described above. A photograph of a completed 48 inch x 72 inch 6-stringer panel is shown in Figure 53.

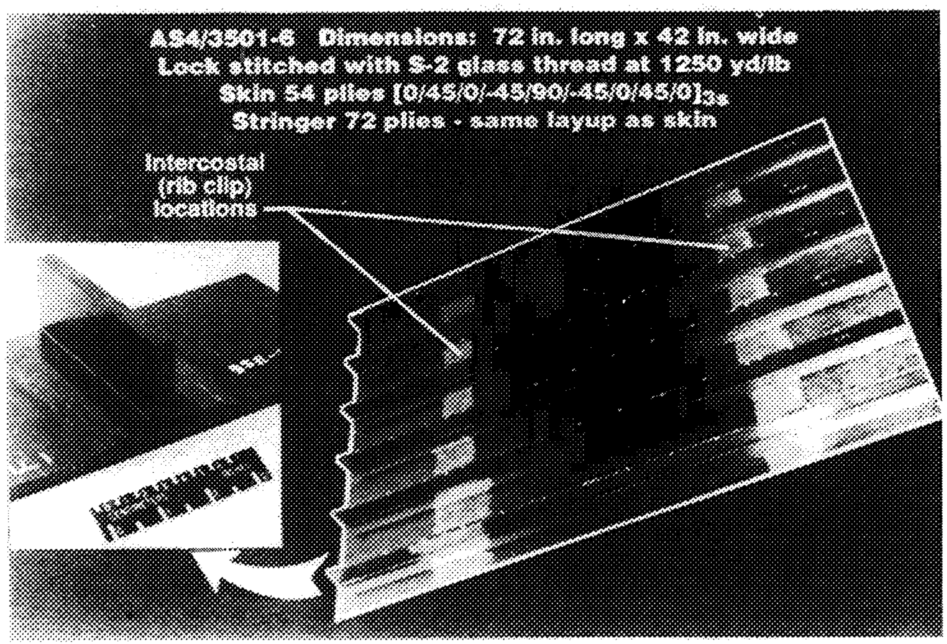


Figure 53. Completed 6-Stringer Panel

SECTION 8
CONCLUDING REMARKS

Results are presented from an advanced composites development program by McDonnell Douglas Aerospace-West called "Innovative Composite Aircraft Primary Structures (ICAPS)." Details are presented concerning the design criteria and design philosophy for composite primary structures on transport aircraft. Baseline design loads and panel configurations are given for wing and fuselage applications. Test data are presented from a comprehensive investigation of stitched laminates made from dry carbon fabric preforms with resin transfer molding. The results show that through-the-thickness stitching with closely spaced threads combined with RTM produces outstanding damage tolerance in laminates made with relatively inexpensive brittle epoxy resins.

Stitching, rather than binder resins, was determined to be the preferred method for holding the nine-ply basic stack material together. Following the evaluation of stitching variables the following selections were made for stitching the panel elements together in the succeeding phase of the program:

Stitch Type:	Modified lock stitch
Stitch Thread:	S-2 glass 449-1250 untwisted
Stitch Pattern:	0° rows / 3/16 inch spacing /1/8 inch steps.

Two approaches to RTM are described: a vacuum impregnation molding for thick wing structure and a resin pressure molding for thin fuselage panels. The tooling for RTM fabrication is described in considerable detail and the fabrication of a blade stiffened wing panel is described in step-by-step detail from preform stitching through resin film infusion (RFI). The RFI process, using a cast film of 3501-6 resin and autoclave pressure, was selected for future wing panel development. The Shell 1895 resin and resin pressure injection was selected for fuselage panel studies.

Cost studies on the three-stringer wing panel specimen indicated a 35 percent reduction compared with the hand layup procedure.

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This report covers work accomplished in the Innovative Composite Aircraft Primary Structure (ICAPS) program, NASA Contract NAS1-18862. An account is given of the design criteria and philospphy that guides the development of composite primary components. Wing and fuselage components used as a baseline for development are described. The major thrust of the program is to achieve a major cost breakthrough through development of stitched dry preforms and resin transfer molding (RTM), and progress on these processes is reported. A full description is provided on the fabrication of the stitched RTM wing panels. Test data are presented.

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