NASA Contractor Report 159150

Preliminary Design of Graphite Composite Wing Panels for Commercial Transport Aircraft

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BOEING COMMERCIAL AIRPLANE COMPANY
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I. Abstract

This commercial aircraft wing surface panel configuration study subjectively assessed practical and producible graphite/epoxy designs. Key experienced engineering, manufacturing, and quality control personnel provided the assessment information, using definitive data as well as their experience and judgement in screening and selection of the final panel configuration.

A multilevel screening procedure was used to review the panel designs, considering the following areas:

- Structural functions
- Efficiency
- Manufacturing and producibility
- Costs
- Maintainability
- Inspectability

As each progressive screening level was reviewed, more definitive information on the structural efficiency (weight), manufacturing, and inspection procedures was established to support the design selection. The final design selection represents a reasonable compromise between all requirements.

The configuration features that enhance producibility of the final selected design can be used as a generic base for application to other wing panel designs. The selected panel design showed a weight saving of 25% over a conventional aluminum design meeting the same design requirements. The estimated cost reduction in manufacturing was 20%, based on 200 aircraft and projected 1985 automated composites manufacturing capability. The panel design background information developed will be used in the follow-on tasks on this contract to ensure that future panel development represents practical and producible design approaches to graphite/epoxy wing surface panels.

II. TABLE OF CONTENTS

III. INTRODUCTION

The structural efficiency of stiffened graphite/epoxy compression panel configurations has been studied by several authors, and typical results are presented in references 1 through 4. These studies have established a good analytical design base for graphite/epoxy panels, several configurations of which may be suitable for commercial aircraft wing panel applications. The reference studies have addressed only the panel structural efficiency; however, other factors that affect the selection of a wing panel configuration must also be accounted for in practical aircraft designs. These factors may include the effect of cutouts and holes, fail safety, rib and stringer attachment, and fuel containment, as well as many others. Nonstructural aspects, which must also be considered, typically include manufacturing requirements and costs, as well as service and environmental conditions.

The present study addresses the design of stringer-stiffened graphite/epoxy composite wing panels, not only as a continuation of the referenced structural efficiency studies, but also as a state-of-the-art assessment of their producibility and cost. The study was conducted by first establishing structural requirements and design goals. The initial structural requirements were established by NASA as minimum requirements for the final panel design. Additional requirements were established by Boeing, to make the final configuration compatible with a total wing structure and to meet practical requirements previously noted. A multilevel screening procedure was used, so that several configurations could be reviewed at the initial level, and this could provide an in-depth look at the designs as the number of configurations was reduced. The early screening procedure involving many configurations used subjective inputs from several disciplines including materials and processes, design, manufacturing, structural analysis, engineerings, and production planning and tooling. This multiple discipline approach ensured that realistic panel designs emerging from the study would not only be structurally efficient, but would also be producible and competitive on a cost basis with present-day aluminum panels. One feature of the selected design is the potential to utilize an automated production process.

The structural analysis for this program was performed using the NASA programs VIPASA and PASCO, (refs. 5, 6). These programs were used to assess and optimize the structural efficiency of the compression panel designs.

The recently completed Advanced Composites Wing Study program (ref. 7) was used as background information for the present investigation. Its information base aided in assessing the ability of the panel designs to meet all of the wing's functional, as well as structural, requirements. A number of individuals who participated in the Advanced Composites Wing Study program also assisted in the screening review of the present study.

IV. ACKNOWLEDGEMENTS

The following individuals made significant contributions to the material contained in this report:

John McCarty	Advanced Structural Concepts
Trent Logan	Advanced Composites Design group
Wayne Nakamura	Advanced Composites Design group
John Gleadle	Advanced Composites Design group
Jim Barry	Industrial Engineering
James LeMert	Manufacturing Research and Development
Wang Yu	Quality Control

V. SYMBOLS

A	Panel surface area
B	Panel width
E_t	Smeared extensional stiffness
E_1	Lamina elastic modulus in fiber direction
E_2	Lamina elastic modulus in transverse direction
G_t	Smeared shear stiffness
G_{12}	Lamina inplane shear stiffness
L	Panel length
N_x	Inplane compression loading
N_{xy}	Inplane shear loading
W	Panel weight
γ_{12}	Allowable inplane shear strain
ϵ_1	Allowable strain in fiber direction
ϵ_2	Allowable transverse strain
ρ	Density
μ_{12}	Poisson's ratio
μ_{21}	Poisson's ratio

VI. WING PANEL DESIGN REQUIREMENTS

The following design requirements and the multilevel screening procedure were established to discipline the design, analysis, review, and selection process of this preliminary design wing panel study. The panel design requirements served two purposes: 1) to guide the design development, and 2) to act as a baseline against which to measure the various design configurations. The requirements listed encompass structural design requirements, and other requirements ranging from wing design criteria to study goals.

These requirements were developed from the contract study requirements specified by NASA and specific and/or implied design goals. In addition, Boeing added requirements to bound the study scope and expose some practical considerations that should be reviewed during the panel design development and screening process. Since this was a preliminary design study, many of the requirements listed could only be reviewed in cursory and subjective manner by the design and review team. Therefore, not all of the requirements were met in a quantitive manner during the design and review process. Many of the items considered relied on the information developed and reviewed in the Advanced Composites Wing Study program (ref. 7). In summary, the requirements place some bounds on tnis study and the designs developed while providing uniform criteria for design evaluation and selection.

A. Structural Requirements

The following list of requirements was established by NASA as definitive final panel design requirements for this study.

NASA Structural Requirements

a) Panel shall be capable of simultaneously carrying 2.63 mn/m (15,000 lb/in) ultimate axial load and .45 mn/m (2,600 lb/in) shear ultimate load

b) Panel shall have an applied axial strain equal to or greater than 0.004 at design ultimate load

c) Rib spacing shall be 76.2 cm (30 in)

d) Panel shall have a shear stiffness of approximately .149 gn/m (0.85 x 10^6 lbf/in)

e) Panel designs shall be constrained by realistic wing box consideration

f) Current design properties for Narmco's 5208-T300 graphite/epoxy material shall be used as the material data base for design of the panels

Additional structural requirements were applied to the panels studied by Boeing to further bound the design and study. These and all requirements used in the study were with concurrence of the NASA technical monitor.

Boeing Structural Requirements

a) All designs developed during this study will be reviewed for compliance with the current FAA certification requirements and recommendations (ref. FAA-FAR-25 and Advisory Circular No. AC20-107)

b) All panels must resist skin buckling below limit load if buckling might affect fuel containment or fatigue

c) All laminates will be balanced and symmetrical or quasi-symmetrical by use of repeated sequences

d) All laminates will contain a minimum of 6 % of 90^{o} plies

e) Panel-to-rib joints will be designed for a wing internal pressure condition of 103 kpa (15 lb/in^2) ultimate, acting alone. (This condition results from a refueling valve malfunction)

B. Other Requirements

The requirements listed here were established by both NASA and Boeing. The NASA general work statement for this study identified or implied most of these requirements. Again, Boeing listed these requirements as attempting to aid the design and review team in bounding the study and expanding the elements review criteria. The following list, therefore, spans the previously noted area of goals, guide lines, and design review parameters.

a) The wing total planform to be used in this study is shown in Figure 1, along with the structural arrangement incorporated in the Advanced Composites Wing Study program

b) All panels considered in this study will be reviewed relative to meeting the requirements of the above configuration

c) Design cost comparisons will be established from simple, brief manufacturing plans for both the aluminum baseline and the composite panel designs

d) Comparative cost reviews will be made of all designs. The reviews will assume a production lot of 200 aircraft

e) Designs will be evaluated to assess the effect on the panel design of such functional design features as access doors, drainage requirements, and fuel vents and concentrated load introduction at engine, flap, and landing gear attachments.

f) Cost reduction will be targeted in the design development toward minimum panel costs and reduction of assembly time

g) The weight goal, measured relative to comparable aluminum design, will be to achieve a 25% reduction in weight. The weight comparison will be made against the structural surface panel weight only

h) Cost objective, measured relative to comparable aluminum design, will be to achieve no increase in manufacturing cost

VII. ANALYSIS AND DESIGN METHODS

This study employed the conventional design layout and stress analysis of the panel, along with both a multilevel screening procedure and computer design synthesis of the panels. The design sketches, layouts, and final drawings were developed appropriate to the screening level and in conjunction with the information generated with design synthesis tools. Results of the multilevel screening procedure and the synthesis analysis are presented in Section VIII. The screening procedure and the synthesis analysis used in the study are described in this section. This multilevel screening process has been applied to other Boeing studies, such as the Advanced Composites Wing Study program (ref. 7) and the Advanced Metallic Structures: Fuselage Design for Improved Cost, Weight and Integrity Study (ref. 8). The key to applying this procedure in a prelimiary design study is the use of a review team to guide the design and selection process. For this study, Boeing team members represented the following desciplines and organizations:

- Materials and Processes
- Design Manufacturing
- Structure Analysis
- Quality Assurance
- Industrial Engineering
- Production Planning Tooling

Screening of the candidate panel configurations was divided into three distinct levels or groups. The purpose of the division was to allow assessment of each panel configuration to a detail commensurable with the number of candidate configurations being evaluated. The three levels of screening employed are:

Level 1-Preliminary concept evaluation
Level 2-Secondary concepts screening
Level 3-Final design

For Level 1, nine panel configurations were developed and reviewed. The evaluation team assessed the configurations from the point of view of their respective discipline. They were required to do this subjectively, based principally on their experience and with appropriate depth and expenditure of time for this first-level review. Screening comments were developed covering areas of design suitability, structural efficiency, producibility, and maintainability. After reviewing these comments, Engineering selected the configurations for Level 2 review.

During the Level 2 screening period, four panel configurations from Level 1 were further developed. Small drawings of the panel cross section and its typical attachments were completed. Analysis of these panels was performed to assess panel weight, extensional stiffness, and shear stiffness. Manufacturing reviewed each concept and established relative cost factors, allowing a cost comparison. This cost information, along with the weight information, was reviewed for final selection of the Level 3 configuration.

In Level 3, final screening of the two configurations in Level 2 receiving the highest rating was completed with refining of the design. Another review was conducted to resolve the smallest of design differences of the surviving configurations. From this final selection, the configuration representing the best combination of features for further design study was selected. Key parameters that affected the final selection were the panel weight and the relative manufacturing cost. Since this is a preliminary design study, definitive evaluation of all the design parameters cannot be completely quantified. Therefore, panel weight and the relative manufacturing cost perform the function of describing the relative efficiency of each configuration.

Key design considerations and their interactions with each other that were reviewed by the designers are shown in Table 1. This type of listing was used to continuously remind the team of evaluators that the design requirements of their particular discipline would be constrained and compromised when the designs were reviewed to produce a final efficient and producible design.

The analysis objective was to obtain the stiffener configuration with minimum weight that satisfied the panel design requirements and material property limitations. The initial design constraints consisted of the loads, shear stiffness, and extensional stiffness, as shown in Table 2. These are representative of present-day wing stiffness. For a substitution wing (aluminum to graphite/epoxy), the wing stiffness distribution should be identical.

The material properties used in the analysis are shown in Table 3. The strain cutoff of 0.004 was selected based on design criteria and NASA test results of compression panels with damage. The design criteria dictate that the wing panels with nonvisible damage should be capable of carrying ultimate load (ultimate strain). NASA results indicate that lightly impacted graphite/epoxy panels resulting in nonvisible damage failed at a compressive strain near 0.004 (ref. 9). A strain evaluation of typical 727/737 upper surface wing panels was performed to compare with the limiting material allowable strain. The results, as shown in Figure 2, indicate that strains of 0.004 are exceeded over a significant portion of present-day wings. It would appear that composite wings will require greater bending stiffness (lower strain), or that better materials are required. For this study, the material strain limitation of 0.004 has been used.

The panel configuration structural efficiency analysis was conducted using the NASA-developed PASCO panel sizing code (ref. 6). This computer program combines a rigorous stability analysis (VIPASA, ref. 5) with an optimization code. The analysis capability was used to evaluate and size the various stiffener configurations at Level 2. The same capability was used to define the designs in Level 3. The NASA mode shape plotting program was converted to be compatible with available Boeing software. Plots, as shown in Figure 3, were used to check buckling modes. As an example of this analysis, the hat stringer model is discussed. In applying these design codes to the analysis, design constraints were imposed by linking some of the design variables. A summary of some of these constraints and geometry linkage is shown in Figure 4. As indicated in the figure, some of the design parameters were linked in order to make a practical configuration. As an example, linking was used to:

1) Maintain a 11.4 cm (4.5-in) stringer spacing

2) Maintain the same total thickness of 45-deg fabric in both the skin and stringer portions. This approximates the practical feature of distributing the 45-deg fabric between the skin and stringers to maintain a consistent number of fabric layers

3) Maintain constraints that will yield practical manufacturing stringer configurations

Analysis results, including stiffener dimensions and thicknesses, are shown in Figure 4. The configuration resulting from this design synthesis is a minimum weight design that satisfies the load, strain, stability, and geometry constraints. A comparison of this hat design with those from Reference 4 is shown in Figure 5

The interaction between panel weight, strength design, and stiffness design is one of interest. While major portions of the study involved an evaluation of configurations with all imposed constraints, a few cases were evaluated where the panel shear stiffness and extensional stiffness requirements were relaxed. This provides a measure of the weight penalty for the imposed stiffness contraints in relation to panels designed to carry the loads only. Information of this type will be useful for trade studies of new generation wing geometry.

Results of the study for blade and hat stiffener configurations are shown in Figures 6 and 7. In Figure 6, shear and extensional stiffnesses for composite upper surface wing panels are plotted as a function of load index (end load/rib spacing). Also shown in the figure are data for 727/737 upper surface aluminum wing panels in order to relate study results to present-day aluminum wing structures. The results demonstrate that the shear stiffness of resulting panel design is the same as current aluminum wing panels, while the extensional stiffness is somewhat higher due to the 0.004 strain limitations. This is to be expected, since Figure 2 indicates that significant portions, of the upper surface wing have design ultimate strains in excess of 0.004. When stiffness constraints are relaxed, the resulting panel designs have stiffnesses well below existing wing panels. The resulting shear stiffness is particularly low. With such a drastic reduction in shear stiffness, the resulting panel weights are expected to be considerably less. This is borne out, as shown in Figure 7. The panel weight reduction for a relaxed extensional stiffness is considerably less than that due to shear stiffness. This is probably due to the fact that considerable extensional stiffness is required for compression stability, while the skins of predominately $+45^{\circ}$ easily carry the wing shear load. It is evident that the shear stiffness requirement is the major contributor to composite wing panel weight.

The combination of configuration development, multilevel screening, and computer synthesis tools provided the design and analysis approach used throughout this study. A flow diagram of the study procedure is shown in Figure 8, and can be used to guide the reader through the results of each evaluation.

VIII. RESULTS AND DISCUSSION

A. Preliminary Concept Evaluation

Panel configurations selected for the first-level screening process were based on qualitative judgement, previous studies, and results of the Advanced Composites Wing Study program (ref. 7). That study evaluated a wide variety of composite wing design concepts and wing panel configurations. To establish a background for the detailed assessment of panel concepts to be conducted as part of this study, a summary of the major considerations involved in wing panel design follows.

General planform design and manufacturing considerations show that, despite the desirability of one-piece skin panels from a structural efficiency point of view, practical considerations require a splice at the side-of-body (sweepbreak). As the splice is typically heavy and costly, stringer configurations should be compatible with desirable joint designs. As with the panel assembly, structural efficiency of single-piece wing-box cross sections is offset by practical production considerations, so that a built-up box is used as a baseline design. This aspect does not impact the detail skin panel configuration strongly, but can have significant impact, depending on design strain level and use of mechanical attachments or bonding for the spar-to-skin panel joint. The baseline configuration for the Reference 7 study assumes use of mechanical attachments, and that baseline was also used for this panel study.

A multirib configuration was selected because of competitive structural efficiency and the capability of the multirib design to carry concentrated loads generated by major fitting for such items as landing gear and flap tracks. In addition, the same basic ribs can serve as fuel bulkheads, whereas the other configurations require separate fuel bulkhead designs.

The motivation for classifying stiffener configurations was to provide a set of designs to which any particular stringer shape could be compared. For example, stringers were classified into closed and open sections, as shown in Tables 4 and 5. In addition, discrete and integral stiffeners were considered. These classifications are representative configurations without considering small differences in all of the possible shapes. In order to further simplify the screening process, skin panel-to-rib attachments were considered on a separate basis.

Subjective evaluation of Level 1 concepts were separated into four major categories, which included:

- Design suitability
- Structural efficiency
- Producibility
- Maintainability

Primary emphasis was given to the producibility aspects of the designs, since the manufacturing cost dictates whether a design should be further evaluated. While the selection process was categorized into four areas, other design considerations were reviewed throughout the study.

Specific producibility requirements for wing panel concepts include:

- Capability to taper stringer area
- Producible in long wing sections
- Must be readily inspectable
- Low-cost fabrication of panel
- Low-cost assembly of panel to adjacent structure

The practical constraints eliminate a number of potential wing panel concepts.

Design suitability addressed not only the basic panel configuration but also a number of wing design details, illustrated by the alumimum wing design shown in Figure 9. Specific details reviewed in the concept screening include:

- Side-of-body splices
- Rib and spar attachment
- Stringer runout
- Concentrated load introduction

Only subjective evaluations of these detail design areas were performed in Level 1. In the Level 2 and Level 3 screening, the detail concepts were further developed, and some design layouts were made.

An underlying assumption of the screening methodology was that design features required for damage-tolerant capability will not change overall relative ranking of panel design concepts, as determined by design suitability, structural efficiency, manufacturing producibility, and repairability considerations. Thus, based on preliminary surveys of the state of the art, it is anticipated that essentially the same damage tolerance features would be incorporated into any of the concepts being studied. In addition to these considerations, although the panels were sized to a specific set of load and stiffness conditions, both higher and lower loads were considered during the screening to ensure that the designs selected are appropriate to the total wing surface.

Tables 4 and 5 summarize qualitative judgments made by engineering and manufacturing personnel. Concepts 1, 3, 7, and 9 were ranked highest and were further studied in the Level 2 screening, while the remaining concepts were not considered further. An underlying assumption of the study is that the aircraft will be produced at a rate that will demand a high level of automation. Integral stiffeners (concept 2) would not be cost-competitive with separate stiffeners (concept 3) due to difficulty of automating the fabrication process; any weight savings would be expected to offset hand layup fabrication. Concepts such as 4 and 5 are anticipated to be too costly to fabricate to offset expected weight advantages, primarily due to tooling of the open centers. Concepts 6 and 8 were judged to have sufficiently poor design application to not warrant further study.

B. Secondary Concept Screening

The four panel configurations selected from Level 1 screening were furtner evaluated in Level 2. Analysis and design evaluations were conducted.

Structural Efficiency Evaluation

Stiffener concepts 1, 3, 7, and 9 were evaluated using the design code (PASCO) during the Level 2 screening process. All concepts were evaluated, with identical stringer spacings of 11.4 cm (4.5 in) for comparision purposes. Concept 7 was also evaluated with three additional spacings to determine the sensitivity of stringer spacing on panel efficiency. The results from this analysis are shown in Table 6. Figures 10 and 11 display the relative size of the concepts evaluated. The reported weights do not include any filler for the closed section stringer. The point design weights are considered lower bounds on an actual panel weight. An actual wing panel would be heavier due to inclusion of a few 90° layers in the skin, local padup, core filler, and adhesive weight. From the results of this analysis and earlier Boeing IR&D work (fig. 12) on blade-stiffened panels, the following analysis conclusions have been evolved:

- All designs evaluated have similar structural efficiency (same weight)

- Skins are dominated by 45^{o} ply percentages ranging from 68% to 86%

- Stiffening ratios (stringer load/total load) are high, ranging from 54% in the solid blade (concept 1) to 83% in the hat design (concept 9)

- The average extension modulus for the panels is 75.8 GPa (11.0 x 10^6 lb/in^2) for all sections evaluated

- For the modified blade (concept 7) and J (concept 3), the inner cap carried more extensional load than the outer cap. In the hat (concept 9), the outer cap carries greater extensional load

- Stringer spacing may be substanially increased without a weight penalty

Secondary Concept Screening Results

In addition to the structural efficiency evaluation, the four skin panel configurations (1, 3, 7, and 9) from the Level 1 screening activity were subjected to a more intense review, with emphasis on producibility, adaptability to major load-transfer joining, rib attachment, adaptability to changing load levels, and fabrication costs. Each of the four concepts reviewed in Level 2 represents a viable wing panel configuration. The more intense review sought to expose any long-term objections or short-comings of the concepts, particularly with regard to fabrication of large panel components. In addition, a preliminary assessment of relative costs of fabricating each concept was made. Results of the Level 2 manufacturing evaluation are given in Table 7. In addition, the relative cost and panel weights of the four configurations are summarized. Costs are based on estimated 1985 manufacturing capability, and are consistent with the Advanced Composites Wing Study program (ref. 7) technology assessment.

Design project personnel refined the Level 2 panel configurations. The individual drawings (figs. 13 through 16) summarize the results for each of the four configurations. In addition to the basic panel cross section, potential solutions for rib attachment stringer taper and joints are also shown.

Following a point-by-point evaluation of all four concepts, the blade stiffener was selected as offering the greatest potential for adoption as the basic cross-section concept, with the solid blade used on the lower (tension) surface and the modified blade used on the upper (compression) surface. These concepts were pursued in greater detail in Level 3, and engineering drawings detailing specific upper and lower surface panels were prepared to complete the stiffener section screening process.

C. Final Design

Upper Surface Panel Design

The modified blade configuration was selected from the Level 2 screening as offering the greatest potential for adoption into production. A schematic showing the stiffener skin details is given in Figure 17. The 0° dominated cap areas carry the majority of the end load. Shear stiffness requirements are satisfied with a 45° dominated skin layup.

The cap contains both tape and fabric plies, while the skin and closure plies are all fabric. The closure plies form the webs of the section and are overlapped on the inner cap. Tape ropes fill the corners to provide fillet radii for closure plies. Honeycomb fiberglass core is oriented at 30° to skin plane (fig. 17) to provide support for the webs and cap areas during autoclave cure. While the honeycomb core incorporated into the modified blade is acknowledged to be a weight penalty, some form of interior support is required to support the cap material during the manufacturing/curing cycle. In addition to the basic panel, design sketches are shown in Figure 18 that illustrate potential solutions to such details as spar/skin intersections, shear ties, tank door cutouts, stringer runouts and side-of-body splices. The upper side-of-body joint is of the double plus chord design, similar to existing production airplanes. The double plus chord is a titanium (6Al-4V annealed) formed and machined extrusion. Access doors would be graphite/epoxy construction. Stiffener runouts are made by tapering stiffener ends and adding an end closure piece. Skin panels are mechanically attached to the spars and ribs.

The use of honeycomb core in the upper surface stringer as a fabrication aid raises two questions about: 1) the additional weight of the core in the final structure, and 2) the susceptibility of the core to fuel ingestion. In addressing the cost effectiveness of leaving the core material in the stringer, an evaluation was made by Manufacturing, which concluded that removal of a mandrel over the full length of a commercial aircraft wing does not appear to be an economic or feasible procedure at the current time. The susceptibility of the honeycomb core to ingestion of fluids (either moisture or fuel) must be addressed in terms of the potential of the stringer to damage, and to constraining the detailed design such that no penetrations of the core are made for mechanical fastening or other reasons. Therefore, a design rule must be that no mechanical fastening or penetration will be allowed if a honeycomb core stringer is to be used. At the ends of the stringer, the core space must be enclosed, thereby sealing the stringer over the entire length. If there is no damage to the stringer during fabrication or final assembly due to the stringer being on the upper surface, the likehood of damage due to tool droppage or other similar impact damage is highly unlikely; therefore, moisture entrance through these damage access locations is not considered critical. When viewing the stringer design and these considerations in the overall assessment, the design evaluator believed it offered the best design compromise of all alternative designs and fabrication processess.

The side-of-body joint in most current commercial aircraft wings is the only chordwise splice in the wing. Its location in the wing dictates the transfer of high end loads across the joint. Its design is controlled by the manufacturing assembly requirements. The structural configuration of the joint is, therefore, controlled by three elements: manufacturing assembly requirements, dominant load (tension or compression), and the stringer configuration being spliced.

For a composite wing design, therefore, one of the design considerations unique to composites will be the splicing of the large area of 0^{o} fibers in the stringer. The ability to mechanically attach to these large bundles of 0^{o}fibers will be the key to the side-of-body joint design. The padup of the stringer to incorporate efficient fiber orientation for mechanical splices will have to take place in both the skin and the stringer in the rib bays on each side of the splice. It is anticpated that at least part of the splice plate may be made of titanium for ease of assembly and compactness of details. On the tension side (i.e., lower surface), one possible configuration will be to diffuse the stringer area into the skin at the joint and have a simple double-lap splice joint. The inner splice member would be the chord of the side-of-body rib, and the outer a single splice plate. For the compression side (i.e., upper surface), the skin and the skin area of the stringer could be spliced through as a single piece. The inner chord of the stringer must remain off the skin plane to maintain the out-of-plane compression stability stiffness. Therefore, the splicing of the inner chord must be done separately from the skin and skin chord material. To do this, a titanium splice "T" similar to that shown in Figure 18 can be used as the splice member and the chord of the side body rib.

The splicing method illustrated in Figure 18 could be designed to diffuse the area of the inner chord into the stringer web, which would be padded up with an effective layup orientation for mechanical splicing. The stringer would be spliced to two titanium angles, with the inner leg of the angles tapered to gradually replace the area of the inner cord of 0^{o} fibers in the stringer. The other leg of the angle would be the splicing leg for attachment to the reinforced web of the stinger. These angles would then be spliced to the side-of-body rib "T" cord to complete the load transfer across the splice. Other configurations of the side-of-body splice are also possible, using a tension-type splice rather than the shear splices shown here. Considerable development in this area of splicing for high end loads is required. Since the majority of current two-spar large commercial transport aircraft wings are spliced at the side-of-body and are of a configuration requiring multiple stringer splices, this technology development of major splice configurations and load transfer is an important part of the technology required to support a long-range wing development program.

Lower Surface Panel Design

The solid blade, as indicated previously, was selected for the lower surface panel. A schematic of a typical blade stringer section is shown in

Figure 19. Low compression loads on the lower surface permit the use of a short, stubby solid blade.

The manufacturing process is similar to the upper surface panel. In both the upper and lower panels, considerable lumping of 0^o plies has been shown. Manufacturing costs dictate this approach. Alternate layups that could be evaluated experimentally are shown in Figure 20. The alternate layups may be less susceptible to thermal cracking and have better damage containment.

Manufacturing Concepts

The manufacturing concepts envisioned for fabrication of upper surface wing cover panels are shown in Figures 21 and 22. The panel fabrication process involves automated layup of the basic skin, followed by autoclave cure and nondestructive inspection. The stiffener will be pultruded as a plank and then slit into stiffeners. Stiffener width and height are constant. The stringer cap area is reduced by dropping off plies as a function of pultruded length. Stiffeners will be positioned on the cured skin and the closure layers automatically laid to tie the stiffeners to the basic skin. After autoclave cure, the panel will be reinspected and trimmed on an automated router. The automated layup, pultrusion, and ultrasonic through-transmission inspection are the significant processes that will be employed to produce and inspect the skin panels.

Inspection of the final stringer configuration shown will require special automated ultrasonic equipment and facilities developement. For major wing panels of the size incorporated in a transport wing, the inspection or quality assurance procedure must include preprocess, process, and postprocessing elements. The most important of these is the quality assurance applied during the processing steps. In the preprocess or the layup stage, continuous inspection of each detail layup, whether automated or by hand, must include an inspection such that no further cost of material or labor results from an early defective layup. Some of these procedures are currently in use throughout the industry today. They involve automated layup that is continously monitored through TV or fiber optics, with stacking, and/or orientation automation with orientation marking and optical checking through each layup stage. During the curing (or processing) steps, recording temperature and pressure over the tool surface is an important control. In some cases, it may be necessary for the curing variables to interact in a feedback mode to control the total processing of the part. The final inspection will range from visual inspection of the surfaces, edges, bond flashes, etc., to an automated water-coupled through-transmission multilevel/multihead ultrasonic inspection procedure. Each inspection step must add to the assurance that the end product has structural integrity. For configurations of the stringer shown in this study, ultrasonic transmission through the skin will be used. For the ultrasonic inspections of the stringer area, individual through-transmission procedures using water-coupled heads transversely inspecting the stringers will be required. In critical thick padup areas or areas of potential processing voids such as corner radii, X-ray techniques may be necessary to ensure high quality. Again, however, it will take the total inspection sequence to ensure the quality of the part, and no single procedure can stand alone.

IX. SUMMARY AND CONCLUSIONS

Task 1 of the NASA Durability and Damage Tolerance contract (NAS1-15107) was conducted as an element in the development of damage tolerant wing structure. This wing panel design study addressed the practical features of composite compression wing panel concepts. While damage tolerance of composite stiffened panels cannot be rigorously evaluated, other important structural, design, and manufacturing assessments have been made. This study evaluated a number of compression panel stiffening concepts. Primary objective of the study was to obtain selected designs that have good potential for significant weight savings over aluminum panels and can be manufactured at minimum cost. The study addressed a number of wing details to ensure the practicality of the selected configuration.

The final designs for the upper and lower surface wing panels are considered as a baseline for future evaluation of damage tolerance capability through analysis and testing. The final configuration selected was a "modified blade" (hat-type section with honeycomb core and vertical webs). Analysis results indicated that many stiffener configurations were structurally efficient; however, with the potential of improved material strain allowable, the closed stiffener sections are considered to have the advantage. The honeycomb core, while an acknowledged weight penalty, stablizes the webs and provides support to the section during the autoclave curing cycle. Engineering drawings of the final designs and some specific wing detail sketches are included.

Weight savings over structurally equivalent alumimum panels are approximately 25%. Manufacturing and Industrial Engineering have estimated a 20% reduction in manufacturing cost over alumimum panels. The estimate is based on 200 aircraft and a projected 1985 automated composites manufacturing capability.

Conclusion

The conclusions arrived at as a result of this study show that weight and cost benefits may be achieved in designing composite wing panel structures. Even though design constraints of a strain limitation and stiffness matching are imposed, the benefits that can be achieved are real and attainable. The study did not address such factors as major chordwise splices, rib and spar attachments, or major cutouts in the panels. These important considerations will require further effort and study.

The study did show that due to current strain limitations imposed on the structure, structural efficiency is not a driving force. Structural efficiency will become more dominant as strain levels are increased, which will result in even higher potential weight savings in panel design.

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9. Rhodes, M.D., Williams, J.G., and Starnes, J.H., Jr., " Effect of Low-Velocity Impact Damage on the Compressive Strength of Graphite-Epoxy Hat-Stiffened Panels," NASA TND-8411, 1977.

Table 1. Design Considerations

	Structural impact							
Design considerations	Strength	Damage tolerance	Fatigue	Safety	Maintainability	Weight	Cost	Comments
Design ultimate loads	●			●		●		Sizes most structural components
Allowable material properties	●		●			●		Determines stress levels
Stiffness						●		Required for flutter and dynamic load considerations
Fail safety		●		●		●		Constrains structural concept
Durability			●			●		Determines maintenance costs
Flutter				●		●		Determines wing stiffness requirements
Buckling	●			●		●		Affects fuel sealing and fatigue
Cutouts and holes	●		●		●	●	●	Reduces structural efficiency
Location of hard points			●			●		Design complexity problem
Sonic fatigue			●		●	●		Determines some minimum gauges
Systems location and interaction					●		●	Constrains configuration
Rib and stringer location and attachment					●	●	●	Affects efficiency and cost
Joints	●	●	●	●	●	●	●	Reduced structural efficiency
Location of control surface			●		●	●	●	Constrains configuration
Lightning strike		●		●	●	●	●	Determines some minimum gages
Fuel tankage					●		●	Affects buckling criteria
Material and fabrication costs						●	●	Constrains design configuration
Tooling and equipment requirements							●	Constrains design configuration
Fabrication procedures	●	●	●			●	●	Affects material allowables
Producibility						●	●	Constrains design
Safety and reliability	●	●	●	●	●			Constrains design
Maintainability		●	●	●	●		●	Affects operating costs and safety
Inspectability		●		●	●	●	●	Affects operating costs and safety
Repairability					●		●	Affects operating costs and safety
Environmental degradation	●	●	●	●	●	●		Affects allowables and weight
Flammability and toxicity				●	●		●	Affects crashworthiness

Table 2. Panel Loading and Stiffness Requirements

Symbol	Value in SI units	Value in U.S. customary units
Upper surface compression panel		
End load		
N_x	2.630 MN/m	15,000 lb/in
N_{xy}	0.455 MN/m	2,600 lb/in
Stiffness		
E_t	0.593 GN/m	3.39×10^6 lb/in
G_t	0.149 GN/m	8.5×10^5 lb/in
Lower surface tension panel		
End load		
N_x	2.630 MN/m	15,000 lb/in
N_{xy}	0.455 MN/m	2,600 lb/in
Stiffness		
E_t	0.593 GN/m	3.39×10^6 lb/in
G_t	0.149 GN/m	8.5×10^6 lb/in

Conversion factors:

kg/M^3 = 36.1×10^{-6} lb/in^3 kg/m = 5.59×10^{-2} lb/in

MPa = 145 lb_f/in^2

GN/m = 5.71×10^6 lb_f/in

MN/m = 5.71×10^{-3} lb_f/in .

Table 3. Properties of Graphite/Epxoy Material Used in Panel Analysis

Symbol	Value in SI units	Value in U.S. customary units
Density and elastic properties		
ρ	1580 kg/m^3	0.057 lb/in^3
E_1	131 GPa	19.0×10^6 lb/in^2
E_2	13 GPa	1.89×10^6 lb/in^2
G_{12}	6.4 GPa	0.93×10^6 lb/in^2
μ_{12}	0.380	0.380
μ_{21}	0.0378	0.0378
Allowable strains		
ϵ_1 (ten)	0.004	0.004
ϵ_1 (comp)	0.004	0.004
ϵ_2 (ten)	0.004	0.004
ϵ_2 (comp)	0.004	0.004
γ_{12}	0.010	0.010

Table 4. Closed Wing Skin Panel Concepts—First Level Screening

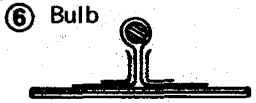
Concept	Design suitability	Structural efficiency	Producibility	Maintainability	Comments	Rank
① Blade  • Panel cocured • Unidirectional layers in center of blade	• Cap material can be built into skin layers • Lacks good ability to tailor shape for variable end loads • Simple splices, and details	• Lacks good compression efficiency at high loads • Least effective skin, acting in compression	• Automated layup difficult • Requires trimming after curing	• Simple ultrasonic through-transmission NDI • Vertical leg requires special fixture for NDI	• Selected for further study due to potential application to lower panels	4
② Integral open section  • Entire panel cocured • Unidirectional material in caps, skin • Wide variety of shapes	• Good ability to tailor for end load changes • More difficult to splice, than ① • Laminate damage propagates directly through lower cap	• Good efficiency over wide range of loads • Similar to concept ③, with slight improvement due to imbedded lower flange	• Automated fabrication very difficult • Requires large amount of hand detail layup	• More difficult to inspect due to upper flange • Internal radii difficult to inspect, compared to concept ①	• Will not be studied further due to anticipated high manufacturing costs compared to "discrete" concepts ③	5
③ Discrete open section  • Stiffener fabricated separately • Cocured or secondary bonded • Unidirectional material in caps • Wide variety of shapes	• Good ability to tailor for variable end loads • Joints, end details more difficult than ①	• Good efficiency over wide range of loads • Better stringer/skin interface, efficiency obtained by cocuring	• Secondary bonding preferred • Cocuring difficult • indentations in skin • stiffener wrinkling during curing • surfaces common to fittings hard to form • higher risk	• Same as concept ②	• Used to generally compare to integral sections by comparing to concept ② • Selected for further study due to high structural efficiency, acceptable producibility	3
⑥ Bulb  • Stiffener fabricated separately • Unidirectional material in cap could add more in skin or add lower cap	• Poor ability to tailor for changing end loads • Questionable design for high end loads • Difficult joints	• Poor efficiency at high end loads	• Fabrication difficult to automate • High cost	• Bulb very difficult to NDI	• Will not be studied further due to cost, lack of design suitability	9
⑧ Modified open section  • Stiffeners fabricated separately • Concentrates unidirectional material for caps	• Good ability to tailor for changing end loads • Difficult joints, end details	• Good stability characteristics • High efficiency	• Difficult to automate fabrication • Difficult to fabricate without indentations due to unidirectional fibers	• Similar to concept ②	• Will not be studied further due to lack of design advantages	7

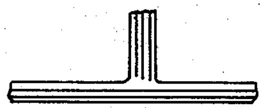
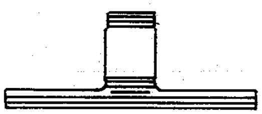
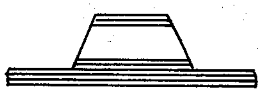
Table 5. Closed Wing Skin Panel Concepts—First Level Screening (Concluded)

Concept	Design suitability	Structural efficiency	Producibility	Maintainability	Comments	Risk
(4) Wye  • Unidirectional material in caps • Stiffener fabricated separately • Cocured or secondary bonded	• Fair ability to tailor for changing end loads • More difficult splices, end details than (3) • Minor fuel volume loss compared to open sections	• High compression load efficiency • Better torsional stiffness than concepts (1) (3)	• Secondary bonding/ cocuring comments same as above • Difficult to form open center section • Tooling/bagging more complex than concepts above	• Same as concept (2) • Open section requires additional NDI; if filled would be more difficult	• Will not be studied further due to anticipated manufacturing difficulties, lack of decisive advantages compared to other concepts	8
(5) Hat  • Stiffener fabricated separately • Cocured or secondary bonded • Wide variety of configurations possible	• Good ability to tailor for changing end loads • Simpler joints, end details than (2), slightly more difficult than (3) • Some fuel volume loss more than (2)	• High compression load efficiency Improves skin buckling due to separated legs • More effective skin in compression • Superior local and torsional stability	• Difficult, costly to produce with open stiffener interior (cocured) • Could be secondary bonded but tooling would be complex	• NDI restricted by closed section • Difficult to automate inspection • Requires development	• Will not be studied further due to excessive projected costs, complex fabrication/ inspection	6
(7) Modified blade  • Stiffener fabricated separately • Unidirectional material in caps • Variety of fillers possible	• Excellent ability to tailor for changing end loads and to improve producibility • Minor fuel volume loss—less than (5) • Joint end detail difficult similar to (5) • Cocuring preferable to secondary bonding	• Efficiency loss due to filler • Good stability in lateral buckling	• Good producibility, ability to automate • Difficult to follow wing contour if precured without mold • See comments for concept (3)	• Filler complicates NDI • May require hand NDI techniques	• Baseline NASA wing study configuration • Selected for further study due to high design suitability and producibility	1
(9) Modified hat  • Stiffener fabricated separately • Unidirectional material in caps or skin • Variety of fillers possible	• Excellent ability to tailor for changing end loads • Joints, end details slightly more difficult than (7) due to slope • Some fuel volume loss—more than (7), but minor	• Less than (5) due to filler • Good panel stability due to separated legs • More effective skin in compression than (7)	• Similar to concept (7) • Greater waste than vertical sides due to trim • Easier bonding tooling than concept (7)	• Similar to (7) Increased NDI cost due to sloping sides	• Will be studied further in connection with concept (1)	2

Table 6. Summary of Level 2 Analysis Results

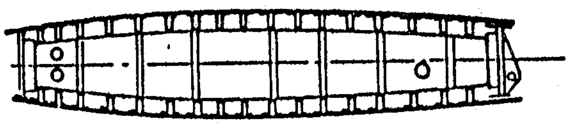
		Concept 1 blade	Concept 9 hat	Concept 7 modified blade 7-1	Concept 7 modified blade 7-2	Concept 7 modified blade 7-3	Concept 7 modified blade 7-4	Concept 3 J
Stringer spacing, cm (in)		11.4 (4.5)	11.4 (4.5)	11.4 (4.5)	13.9 (5.5)	13.9 (5.5)	16.5 (6.5)	11.4 (4.5)
Stringer height, cm (in)		5.8 (2.3)	4.6 (1.8)					
Skin	% 0^o	28	18	19	32	14	19	31
Skin	% 45^o	72	82	81	68	86	81	69
Skin	Width, cm (in)	11.4 (4.5)	6.3 (2.47)	7.1 (2.8)	12.1 (4.76)	8.5 (3.34)	11.4 (4.48)	4.6 (1.8)
Skin	Thickness, cm (in)	0.56 (0.22)	0.51 (0.20)	0.51 (0.20)	0.58 (0.23)	0.51 (0.20)	0.51 (0.20)	0.46 (0.18)
Skin	End load percentage	46	17	21	44	18	23	39
Inner cap (see fig. 1 and 2)	% 0^o	86	89	87	74	89	90	65
Inner cap (see fig. 1 and 2)	% 45^o	14	11	13	26	11	10	35
Inner cap (see fig. 1 and 2)	Width, cm (in)	5.66 (2.23)	3.05 (1.2)	4.2 (1.65)	1.9 (0.73)	5.6 (2.2)	5.1 (2.0)	——
Inner cap (see fig. 1 and 2)	Thickness, cm (in)	0.61 (0.24)	0.69 (0.27)	0.56 (0.22)	1.14 (0.45)	0.66 (0.26)	0.74 (0.29)	0.46 (0.18)
Inner cap (see fig. 1 and 2)	End load percentage	54	34	39	31	51	44	40
Outer cap (see fig. 1 and 2)	% 0^o		59	59	92	46	58	49
Outer cap (see fig. 1 and 2)	% 45^o		41	41	8	54	42	51
Outer cap (see fig. 1 and 2)	Width, cm (in)		5.1 (2.0)	4.2 (1.65)	1.9 (0.73)	5.6 (2.2)	5.1 (2.0)	6.9 (2.7)
Outer cap (see fig. 1 and 2)	Thickness, cm (in)		0.78 (0.31)	0.81 (0.32)	1.2 (0.46)	0.66 (0.26)	0.81 (0.32)	0.30 (0.12)
Outer cap (see fig. 1 and 2)	End load percentage		49	40	25	31	33	21
Panel weight, kg/m^2, (lb/ft^2)		13.7 (2.8)	13.7 (2.8)	14.2 (2.9)	14.2 (2.9)	14.2 (2.9)	13.7 (2.8)	14.6 (3.0)
Stiffening ratio, %		54	83	80	56	82	77	61

Table 7. Level 2 Concept Selection

Concept	Stiffener fabrication	Panel assembly	Quality assurance	Relative cost factor	Panel weight, kg/m^2 (lb/ft^2)
 Blade	• Pultruded	• Closure plies application difficult due to tipping	• Ultrasonic inspection of precured plies • Second inspection of closure plies	1.42	(3.1)
 Discrete open section	• High risk when pultruded*	• Tapered tool required for stringer-to-skin bond • No closure plies required	• Requires both through-transmission and pulse echo inspection	0.99*	(3.1)
 Modified blade	• Pultruded stiffener section • Considerable automation potential	• Amenable to automated assembly	• Inspection marginal with angled core • Ultrasonic inspections of stringer and closure plies	1.00	(3.4)
 Modified hat	• Pultruded stiffener section • Taper complicates stiffener fabrication	• Amenable to automated assembly	• Increased NDI cost due to sloping sides	1.02	(3.4)
Baseline aluminum				1.02	4.7

Final assembly

All concepts require shear ties
All concepts utilize through-the-skin fastening
All concepts require local skin padup for rib attachment



TYPICAL SECTION

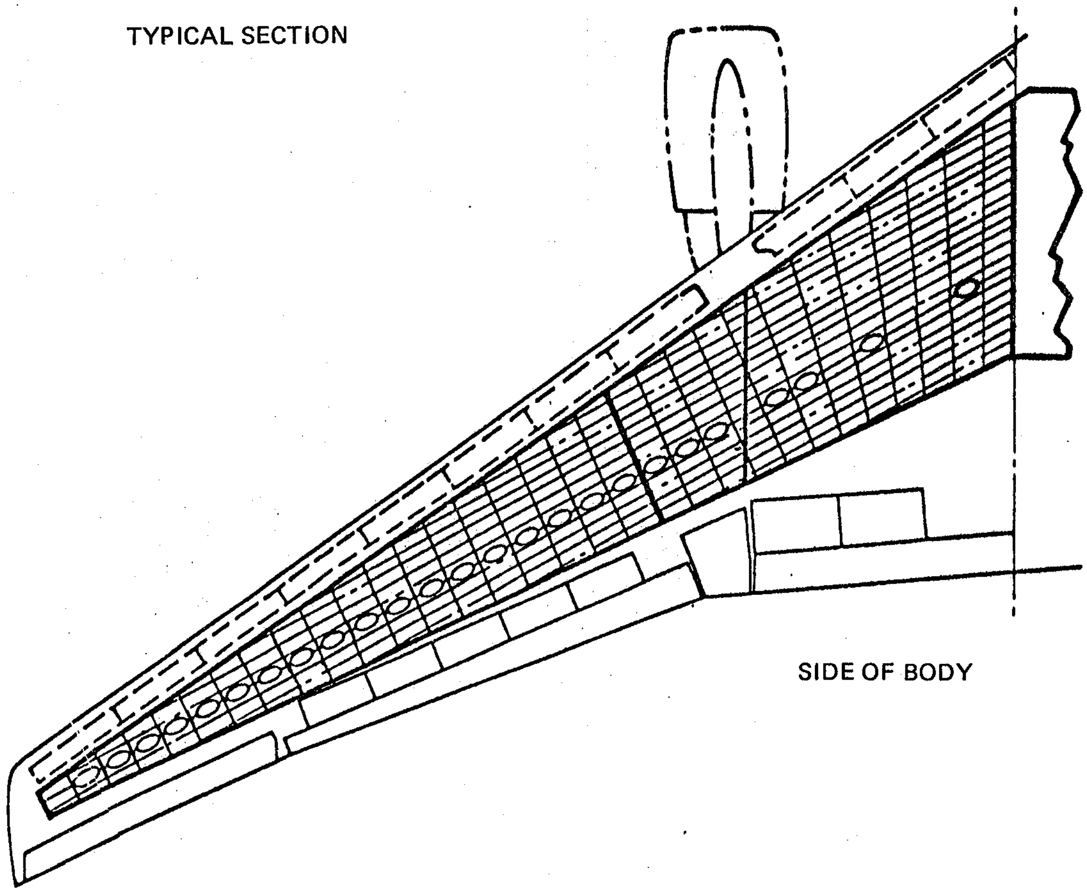


Figure 1. Wing Planform and Structural Arrangement

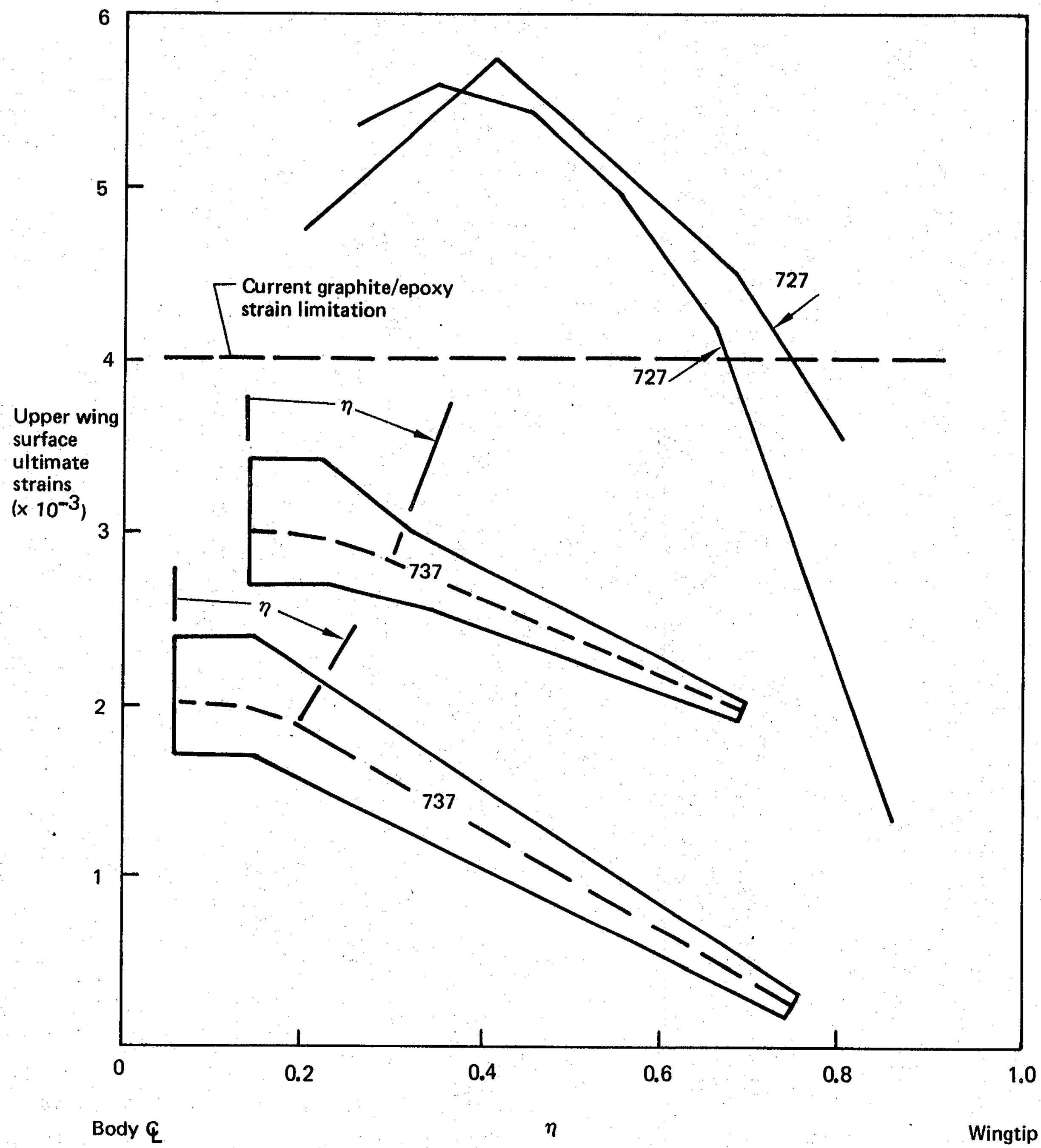
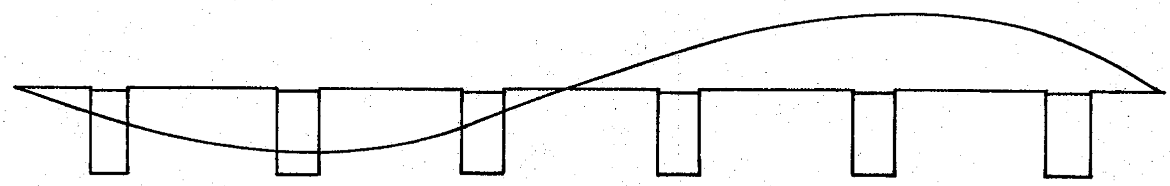
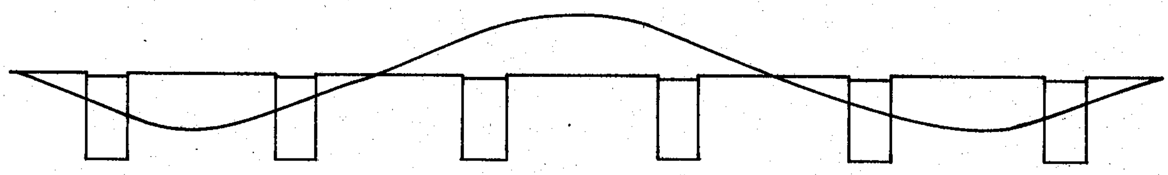


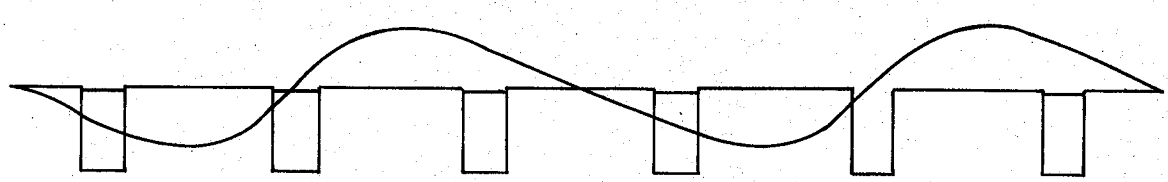
Figure 2. Wing Upper Surface Ultimate Strains



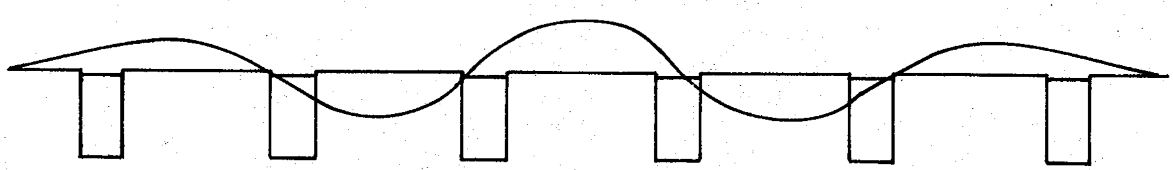
5.5 modified blade compression panel with offsets
Eigenvalue number = 4   Lambda = 30.0000   Factor = 2.6373



5.5 modified blade compression panel with offsets
Eigenvalue number = 3   Lambda = 30.0000   Factor = 2.5574



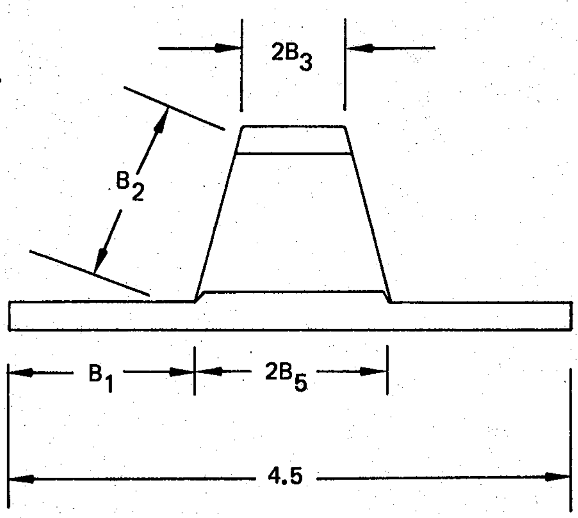
5.5 modified blade compression panel with offsets
Eigenvalue number = 2   Lambda = 30.0000   Factor = 2.5082



5.5 modified blade compression panel with offsets
Eigenvalue number = 1   Lambda = 30.0000   Factor = 2.4920

Figure 3. Buckling Modes

Design parameters	Linked design parameters	Design variables	Final design		
			Width, cm (in)	Total thickness, cm (in)	
$B_1\ (\pm T_1, \mp T_1, T_2)_S$		√	3.43 (1.35)	0.485 (0.191)	
$B_2\ (\pm T_3)_S$	√		4.37 (1.72)	0.076 (0.030)	
$B_3\ (\pm T_3, \mp T_3, T_4)_S$		√	1.19 (0.47)	0.784 (0.309)	
$B_5\ (\pm T_5, \mp T_5, T_6)_S$	√		2.28 (0.90)	0.802 (0.316)	
$T_1 \sim 45$		√			0.0513 (0.0202)
$T_2 \sim 0$		√			0.0383 (0.0151)
$T_3 \sim 45$		√			0.0091 (0.0036)
$T_4 \sim 0$		√			0.356 (0.1400)
$T_5 \sim 45$	√				0.0419 (0.0165)
$T_6 \sim 0$		√			0.233 (0.0917)



Linking relations

1. $2B_1 + 2B_5 = 4.5$
2. $T_1 = T_3 + T_5$
3. $B_2 + 4B_3 - 4B_5 = 0$

Panel weight = 13.13 kg/m^2 (2.69 lb/ft^2)

W/AL = 6.24×10^{-3}

Figure 4. Summary of Hat Design Analysis Results

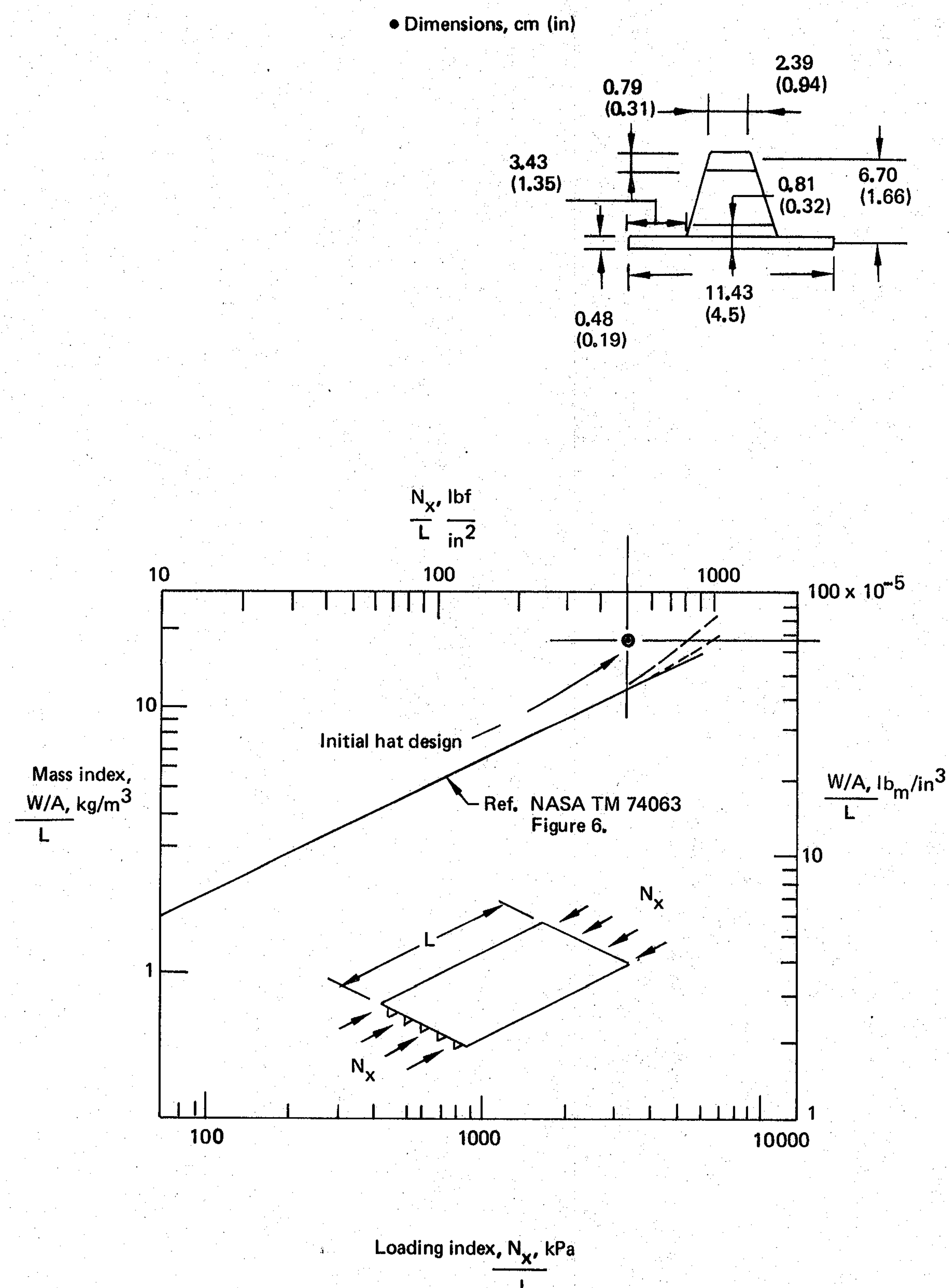


Figure 5. Panel Efficiency

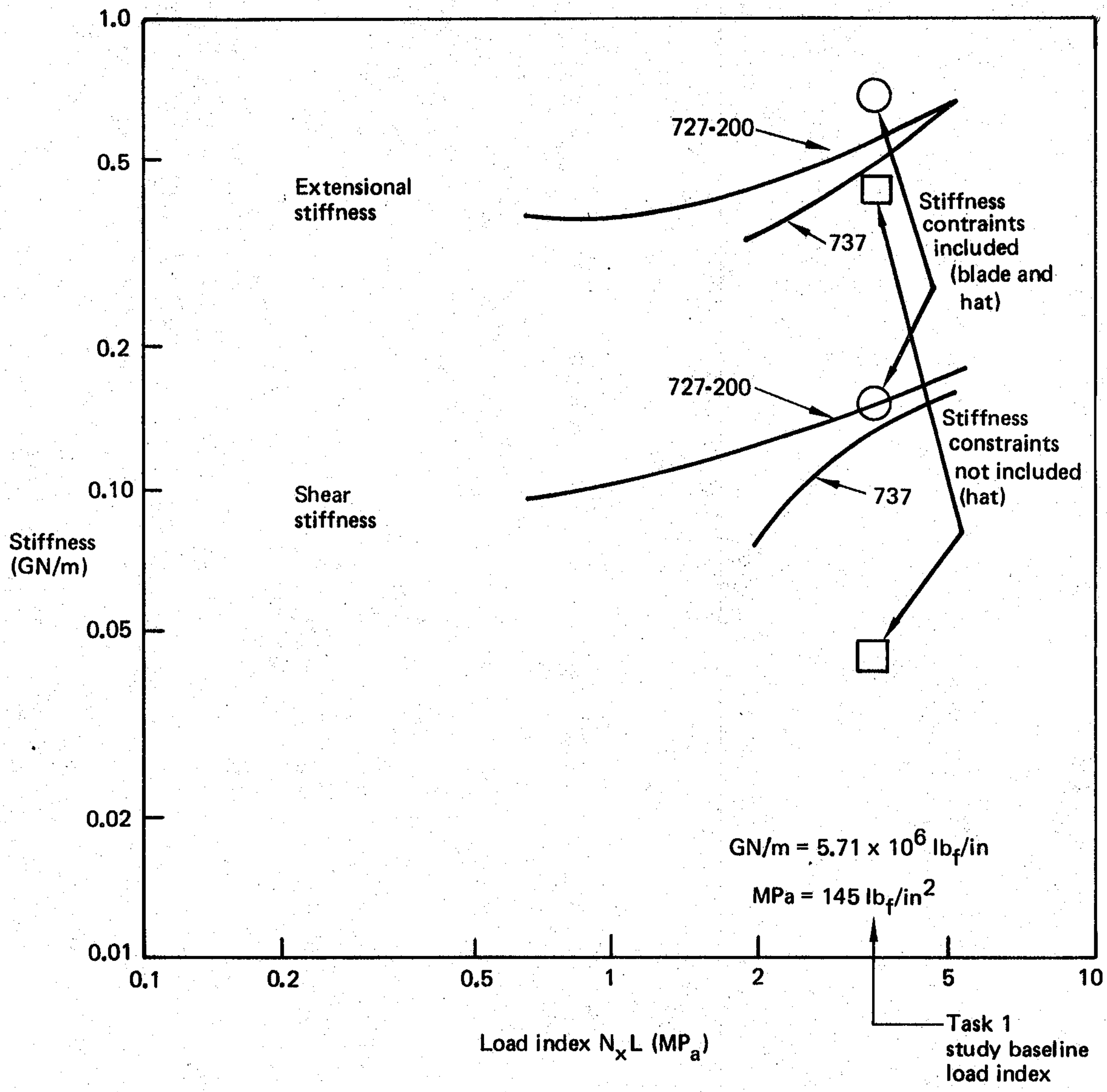


Figure 6. Panel Stiffness Versus Load Index

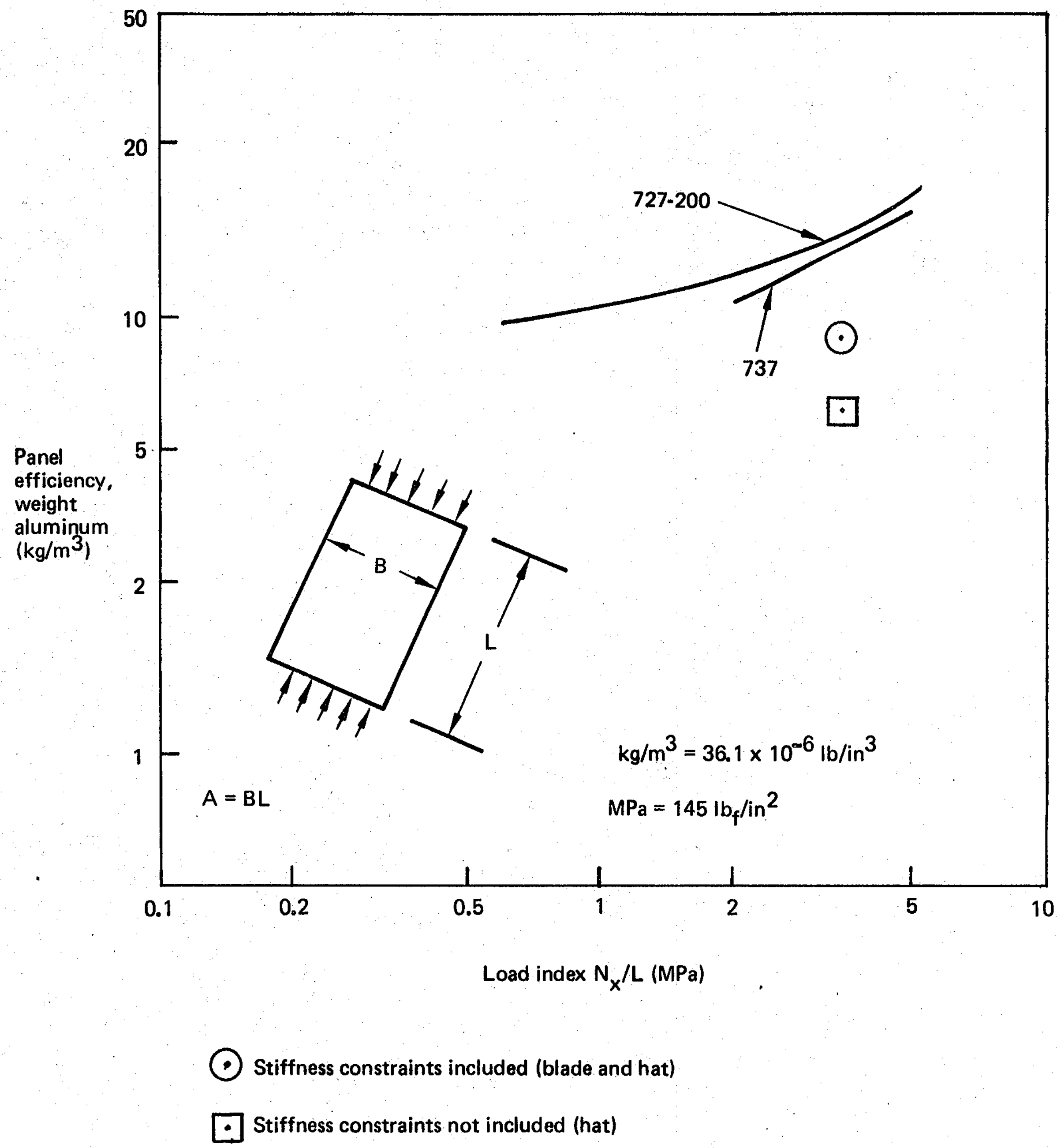


Figure 7. Panel Efficiency Versus Load Index

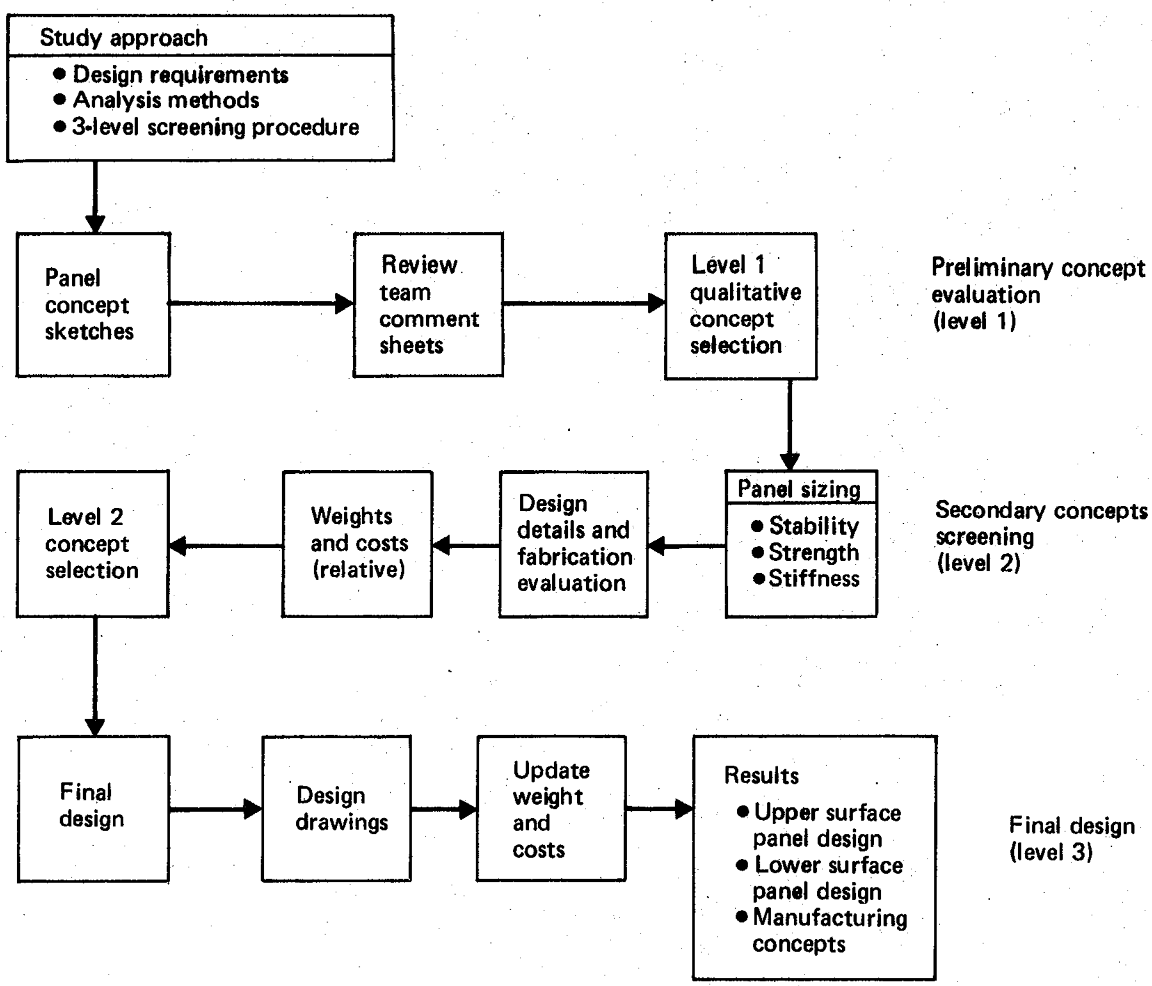


Figure 8. Flow Diagram of Wing Panel Design Study

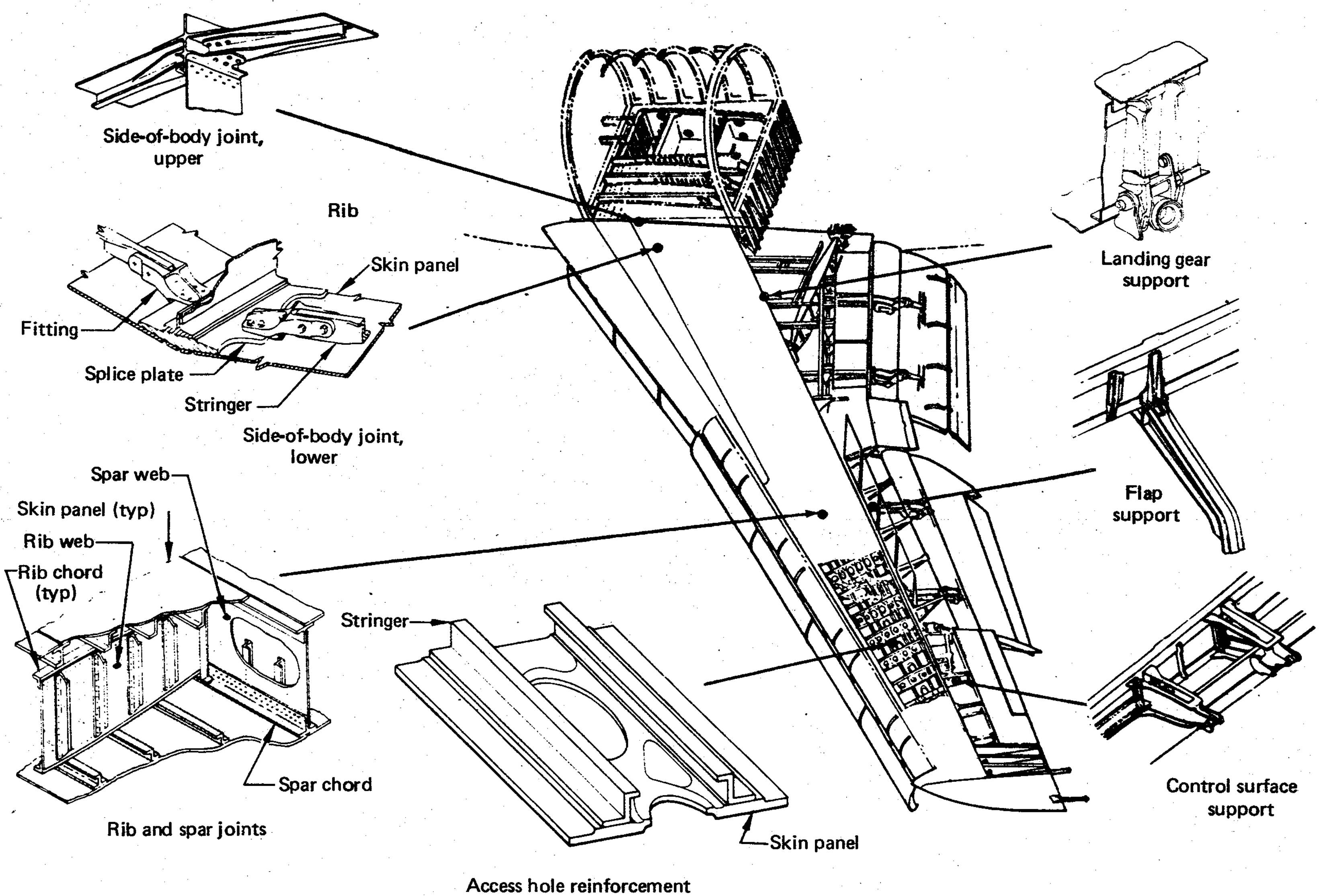


Figure 9. Wing Design Details

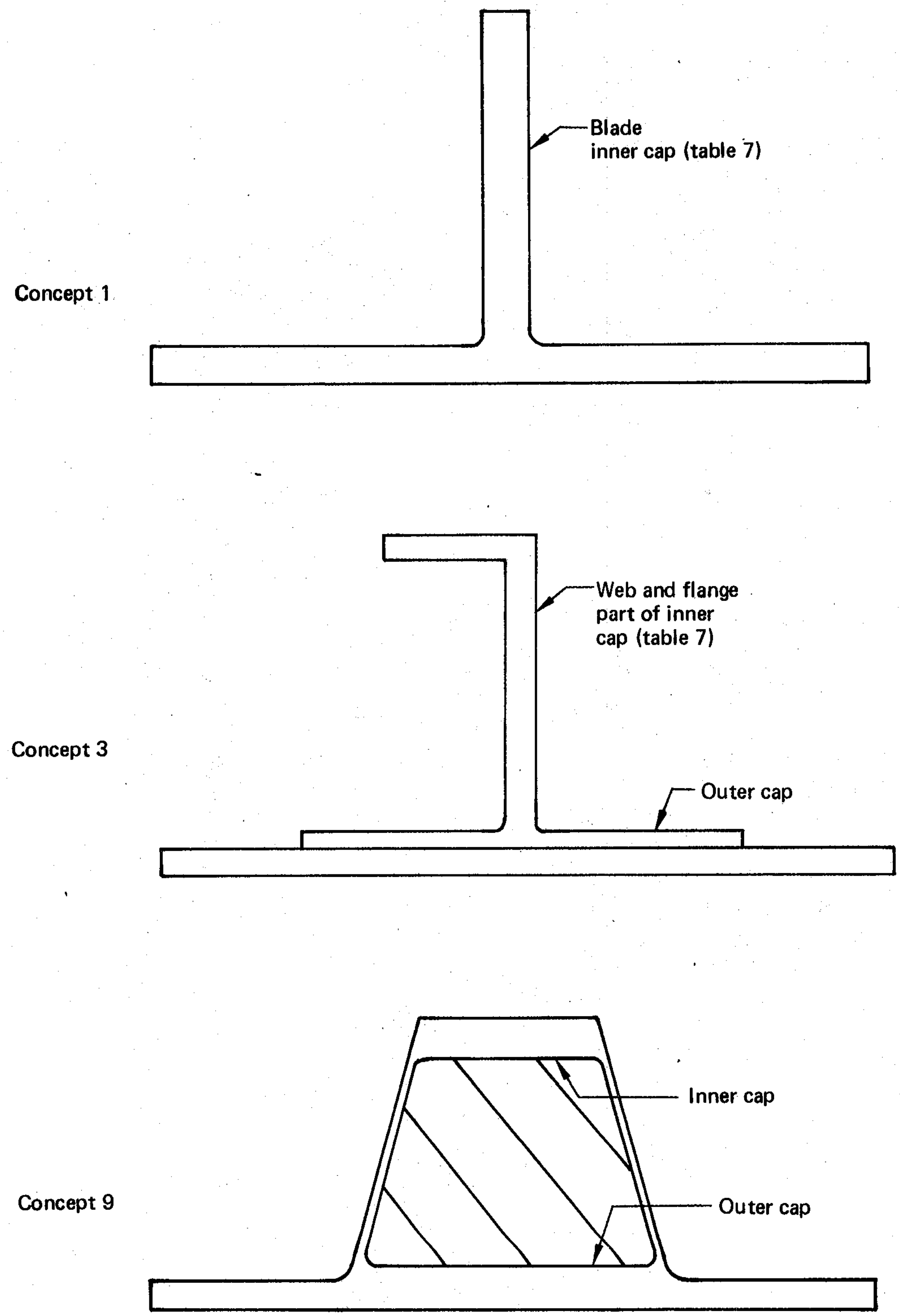


Figure 10. Analysis Model Results

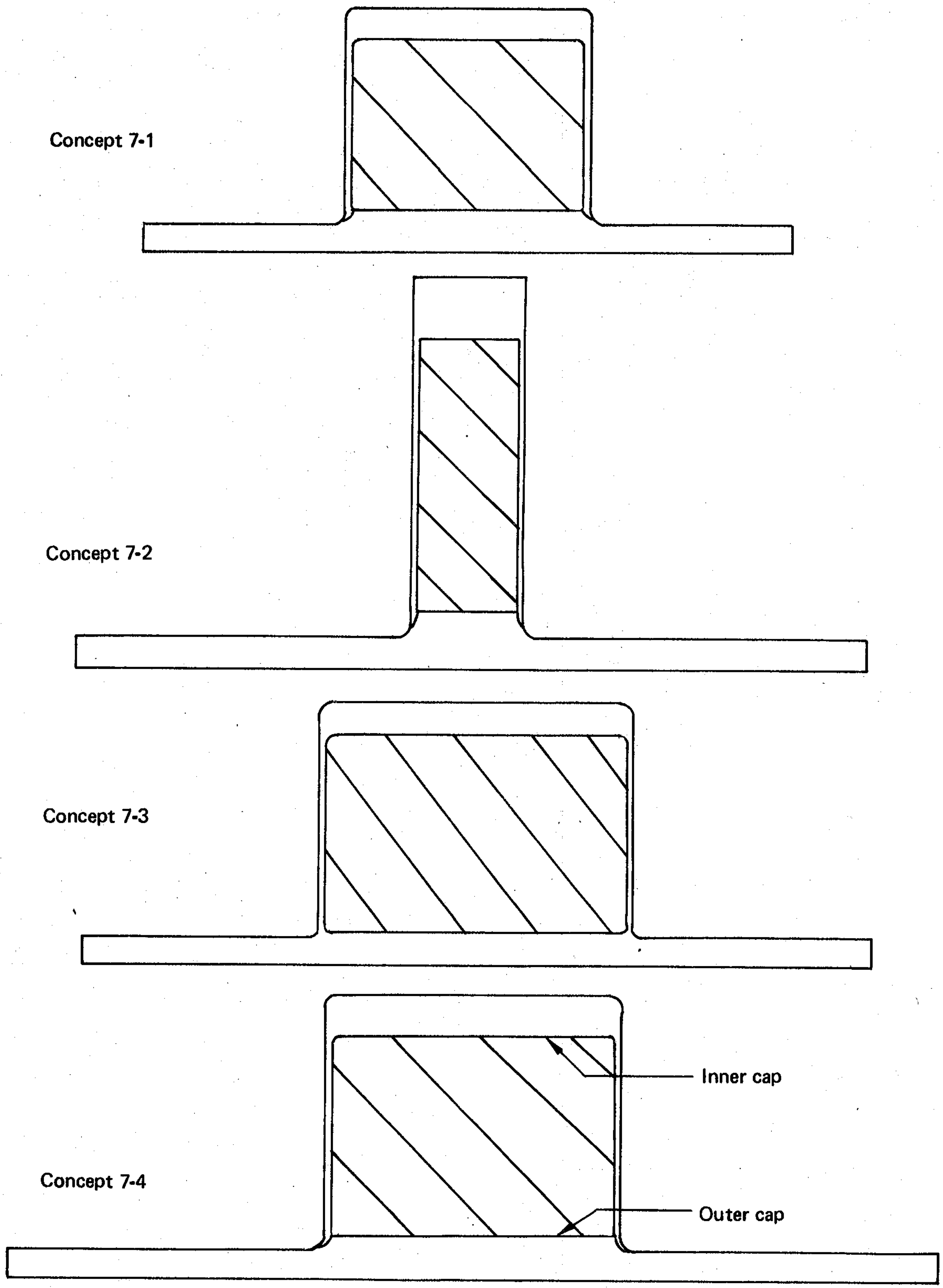


Figure 11. Modified Blade Analysis Models

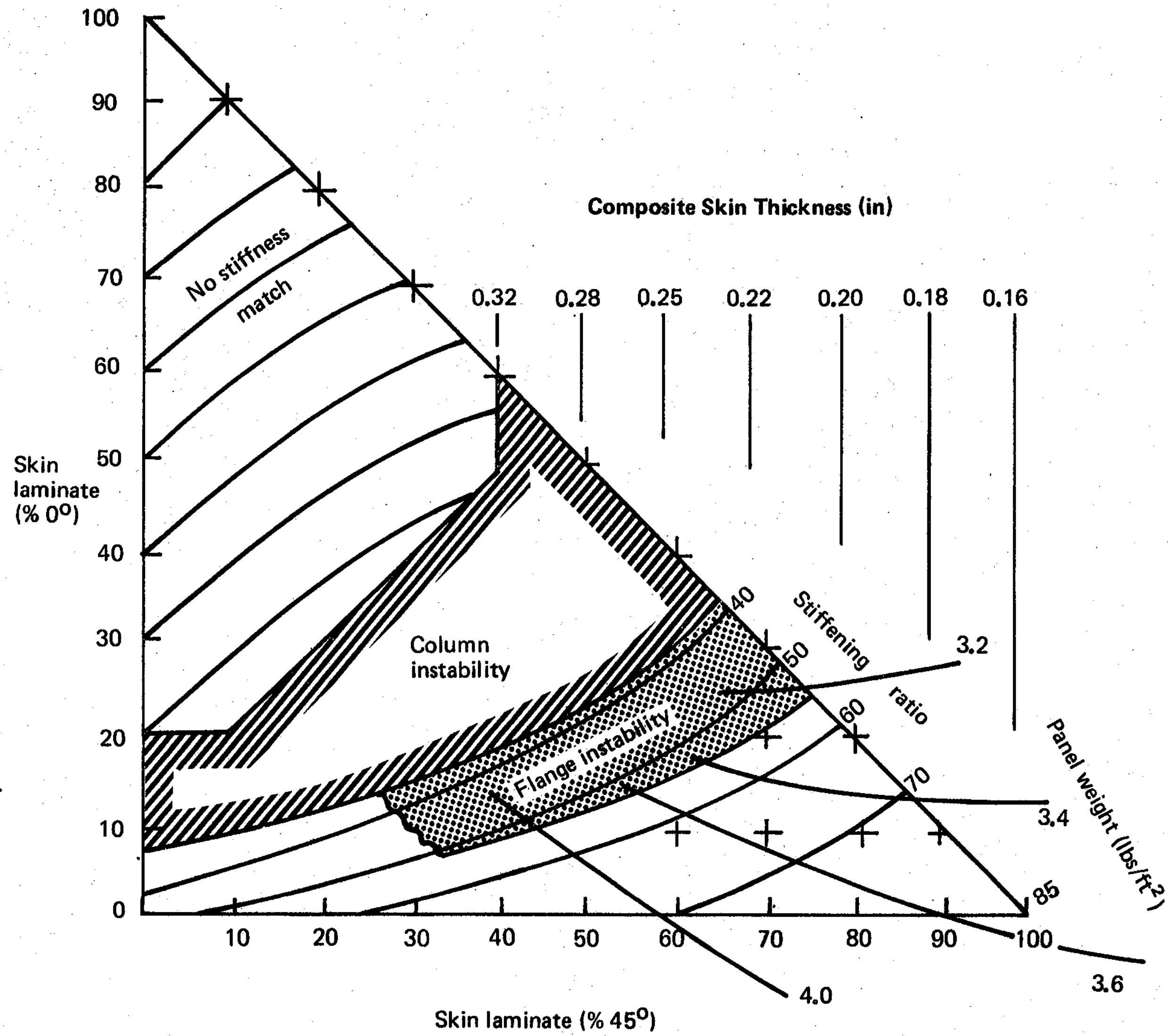
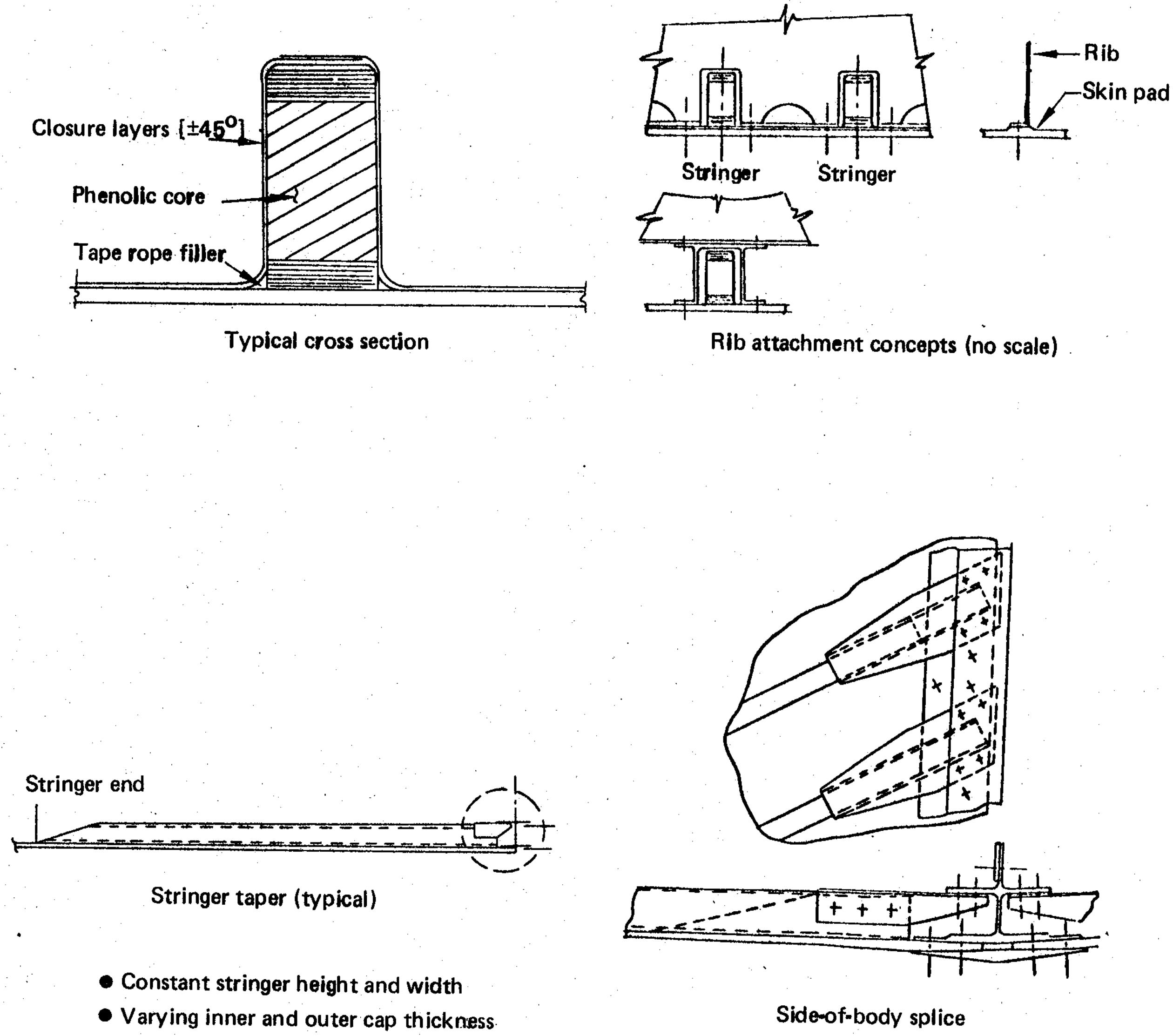


Figure 12. Solid Blade Panel Optimization



Typical cross section

Rib attachment concepts (no scale)

Stringer taper (typical)

- Constant stringer height and width
- Varying inner and outer cap thickness

Side-of-body splice

Figure 13. Durability and Damage Tolerance Level 2 Screening Modified Blade Concept 7

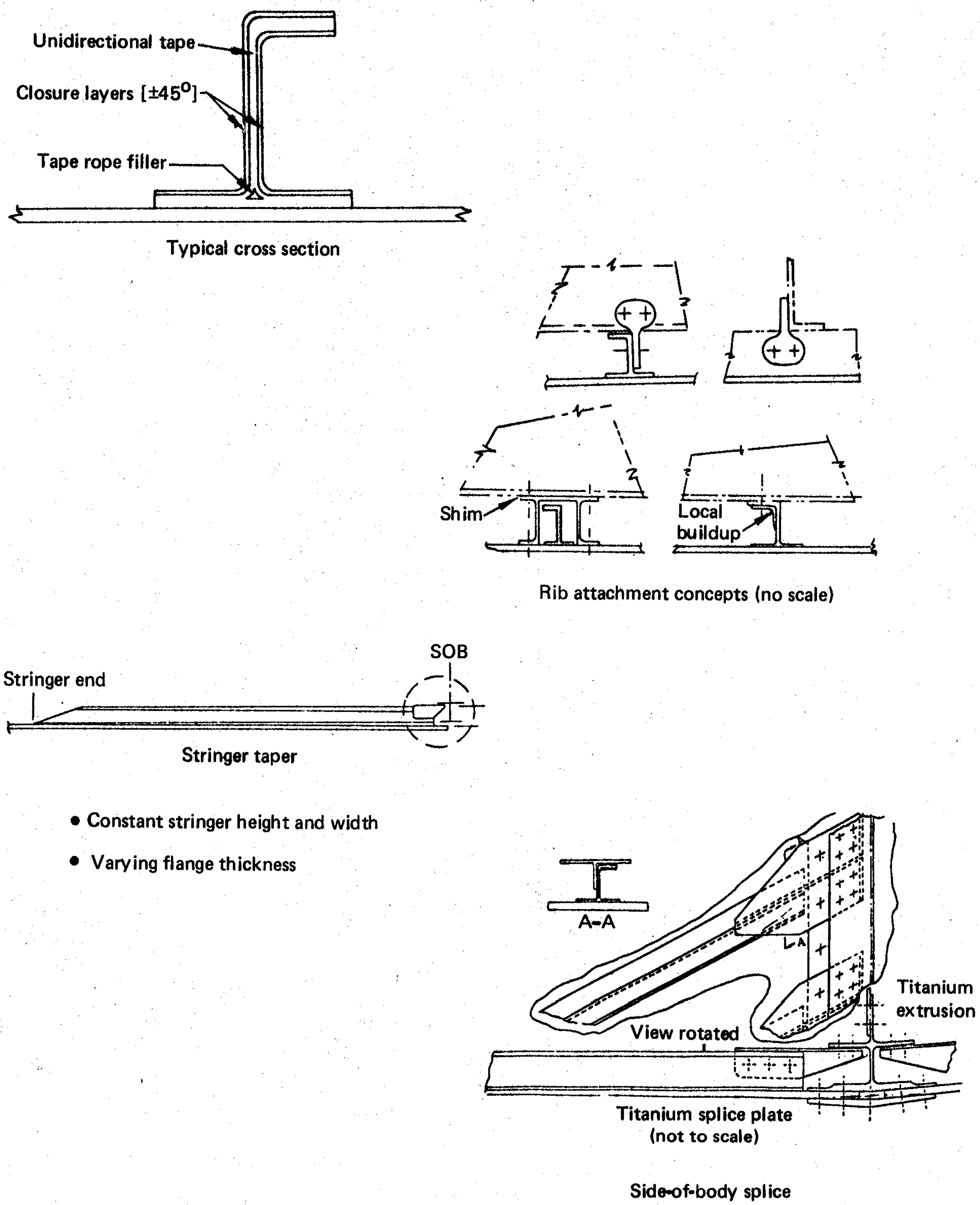
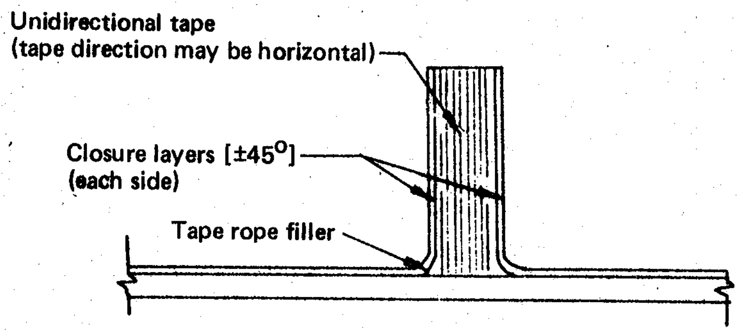
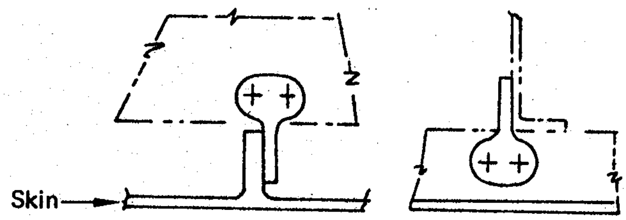


Figure 14. Durability and Damage Tolerance Level 2 Screening Open-Section Concept 3

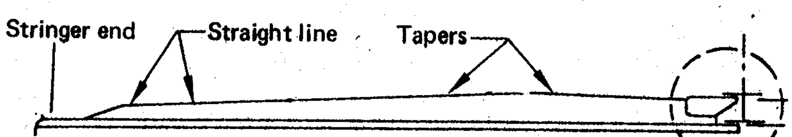


Typical cross section

See side-of-body splice
for discrete open section
concept 3



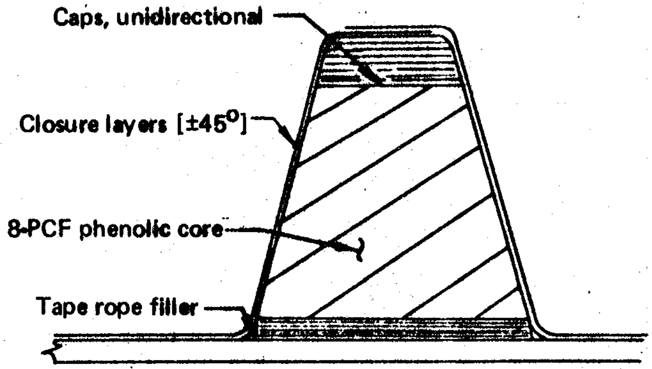
Rib attachment concepts (no scale)



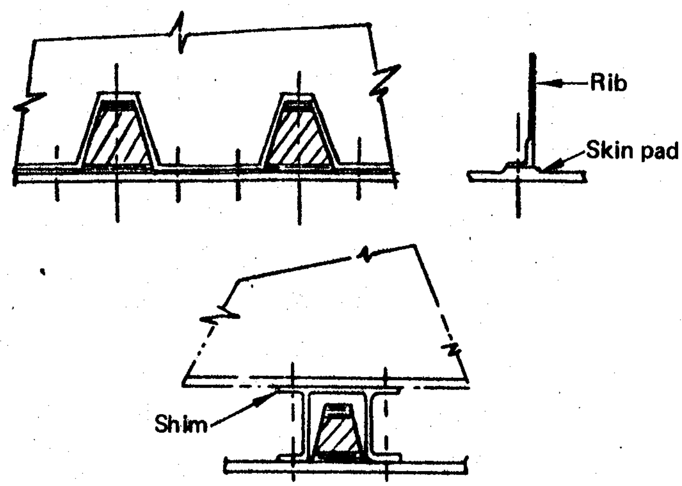
Stringer taper (typical)

- Constant stringer width
- Varying stringer height

Figure 15. Durability and Damage Tolerance Level 2 Screening Blade Concept 1

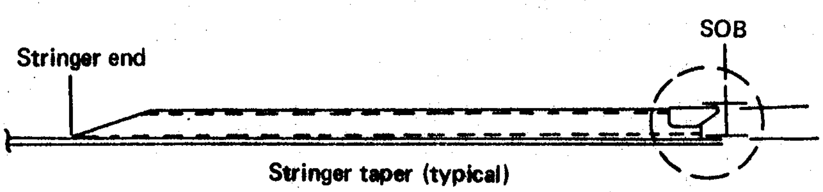


Typical cross section



Rib attachment concepts (no scale)

See side-of-body splice for modified blade concept 7



Stringer taper (typical)

- Constant stringer height and width
- Varying inner and outer cap thickness

Figure 16. Durability and Damage Tolerance Level 2 Screening Modified Concept 9

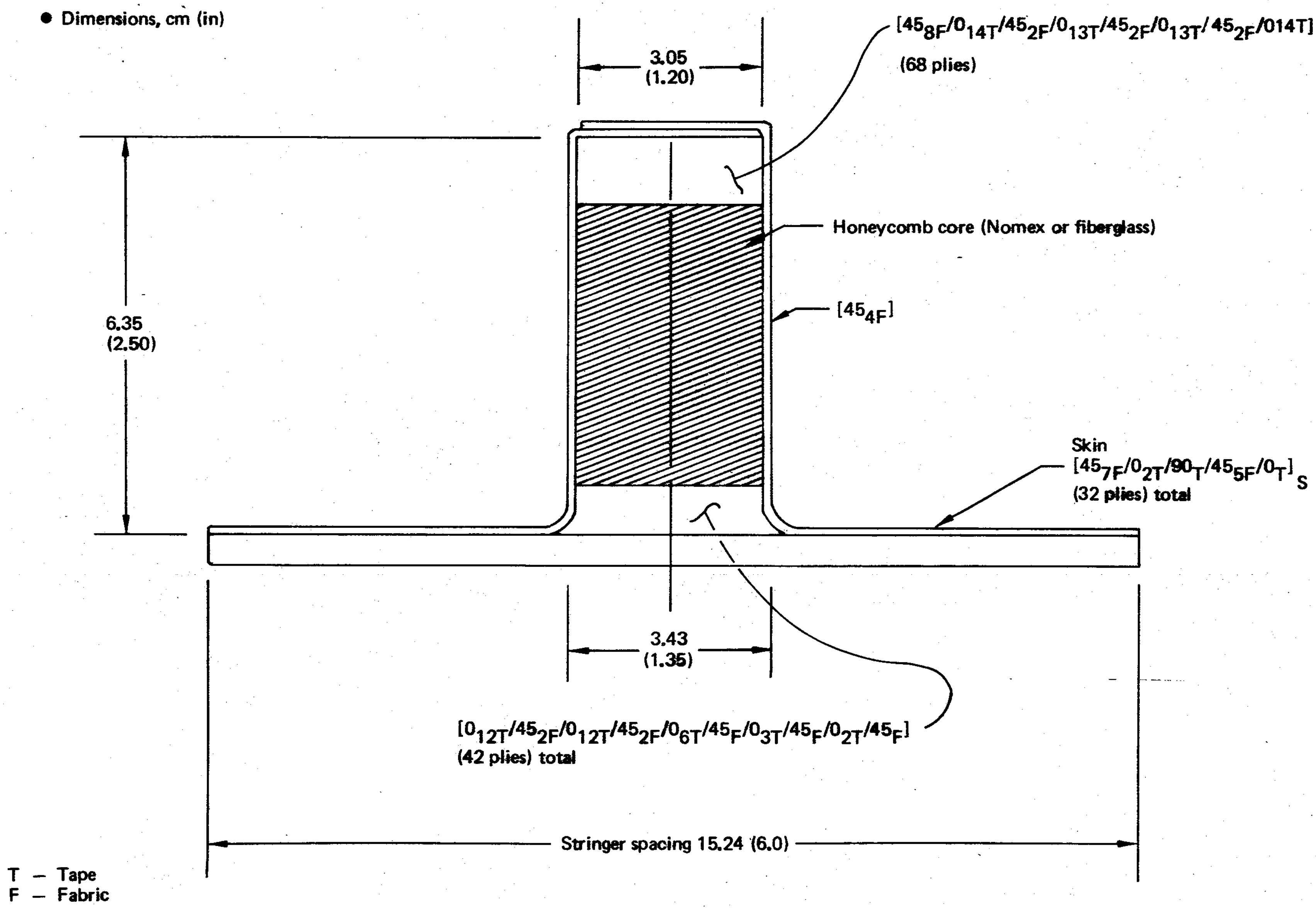


Figure 17. Upper Surface Stringer Configurations

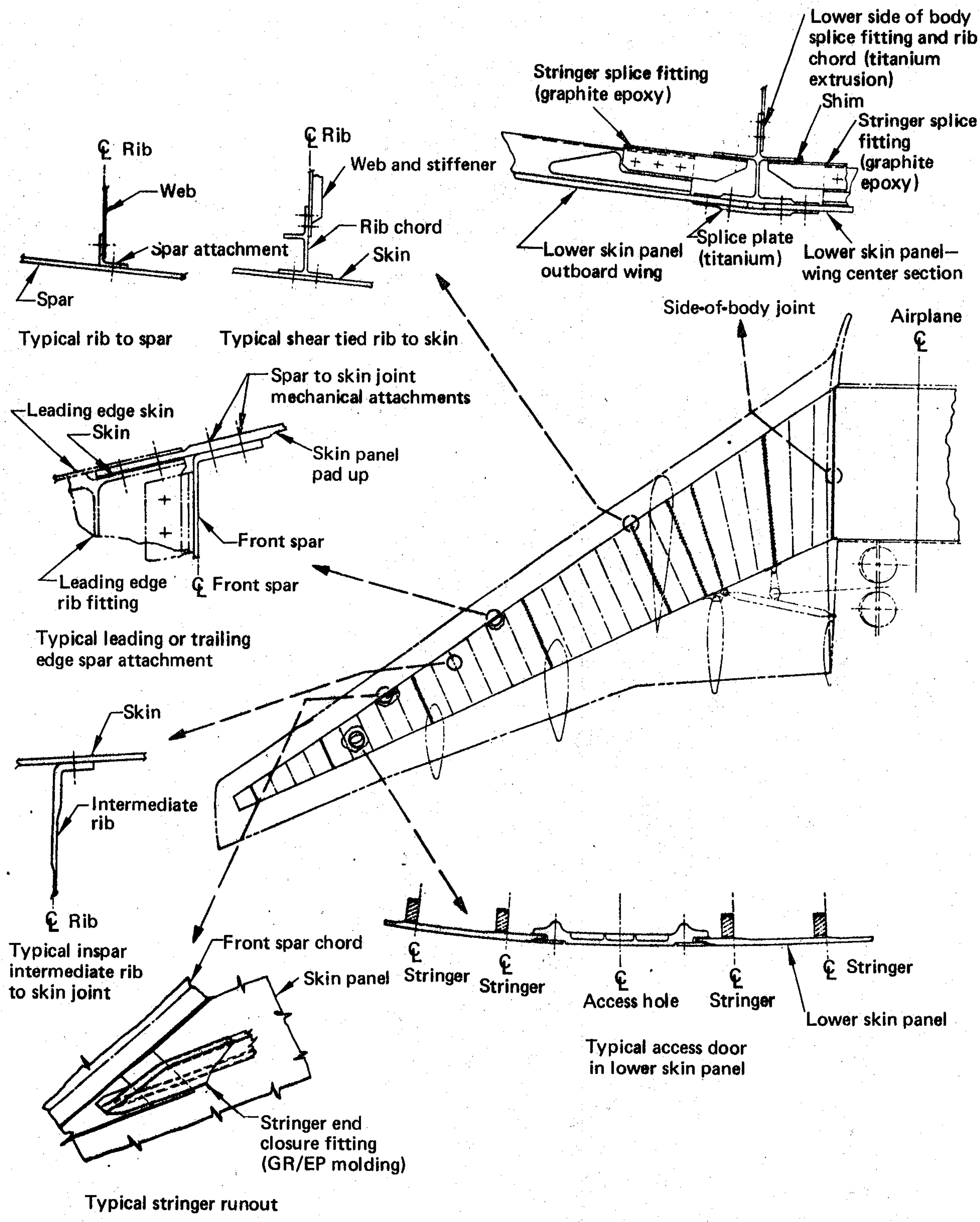


Figure 18. Composite Wing Concepts—NASA Damage Tolerance and Durability Study

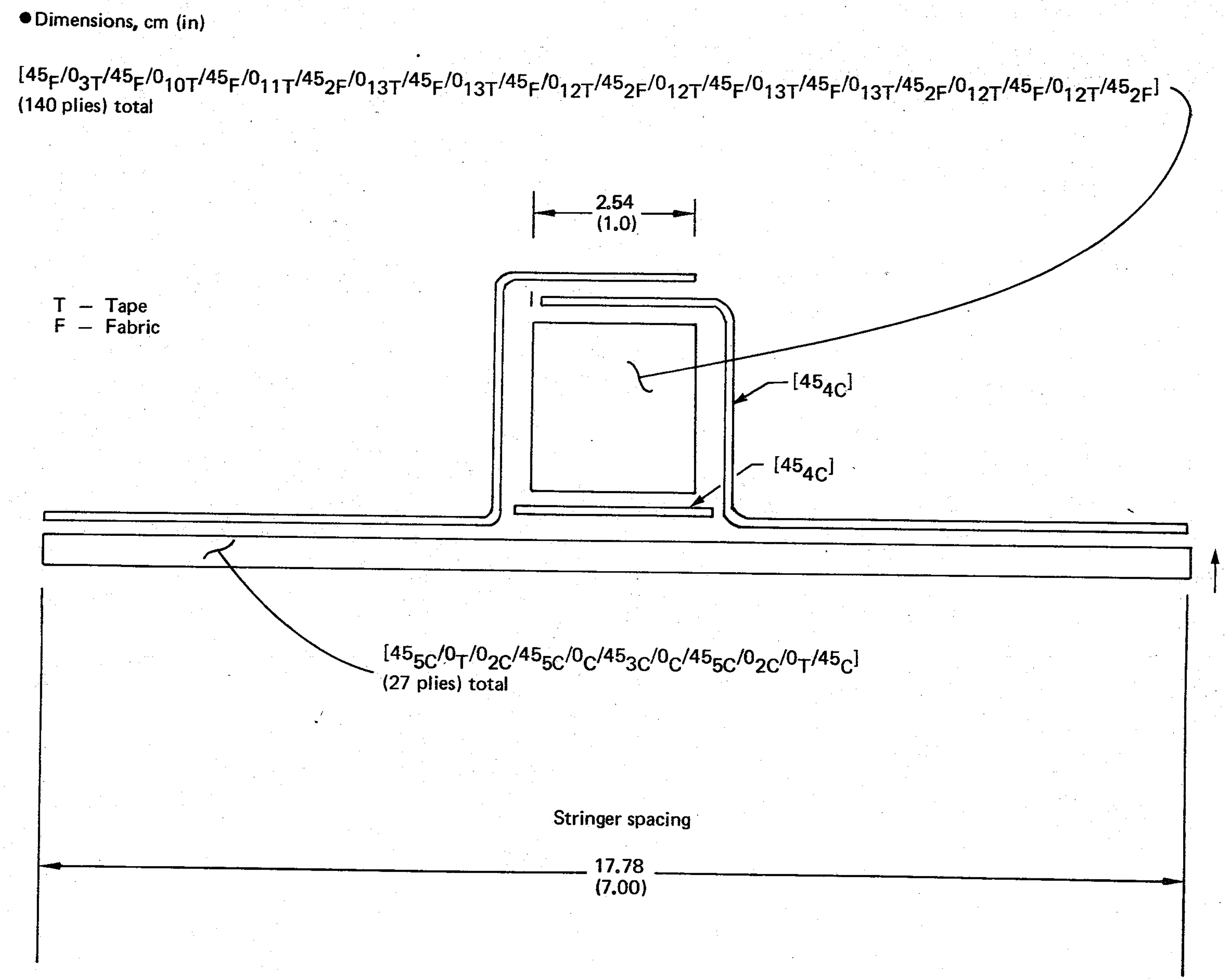


Figure 19. Lower Surface Stringer Configuration

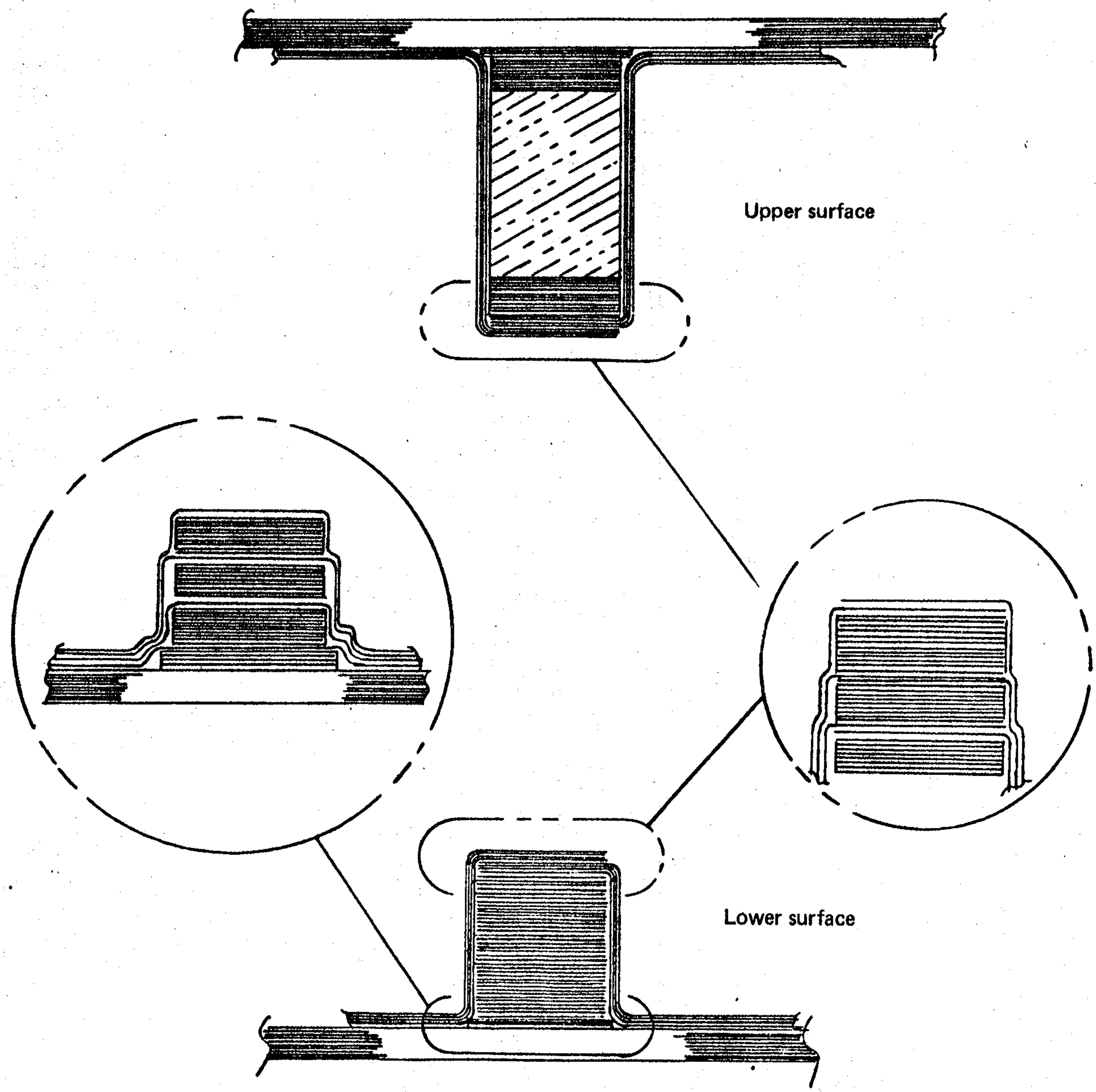


Figure 20. Alternate Layup Schemes

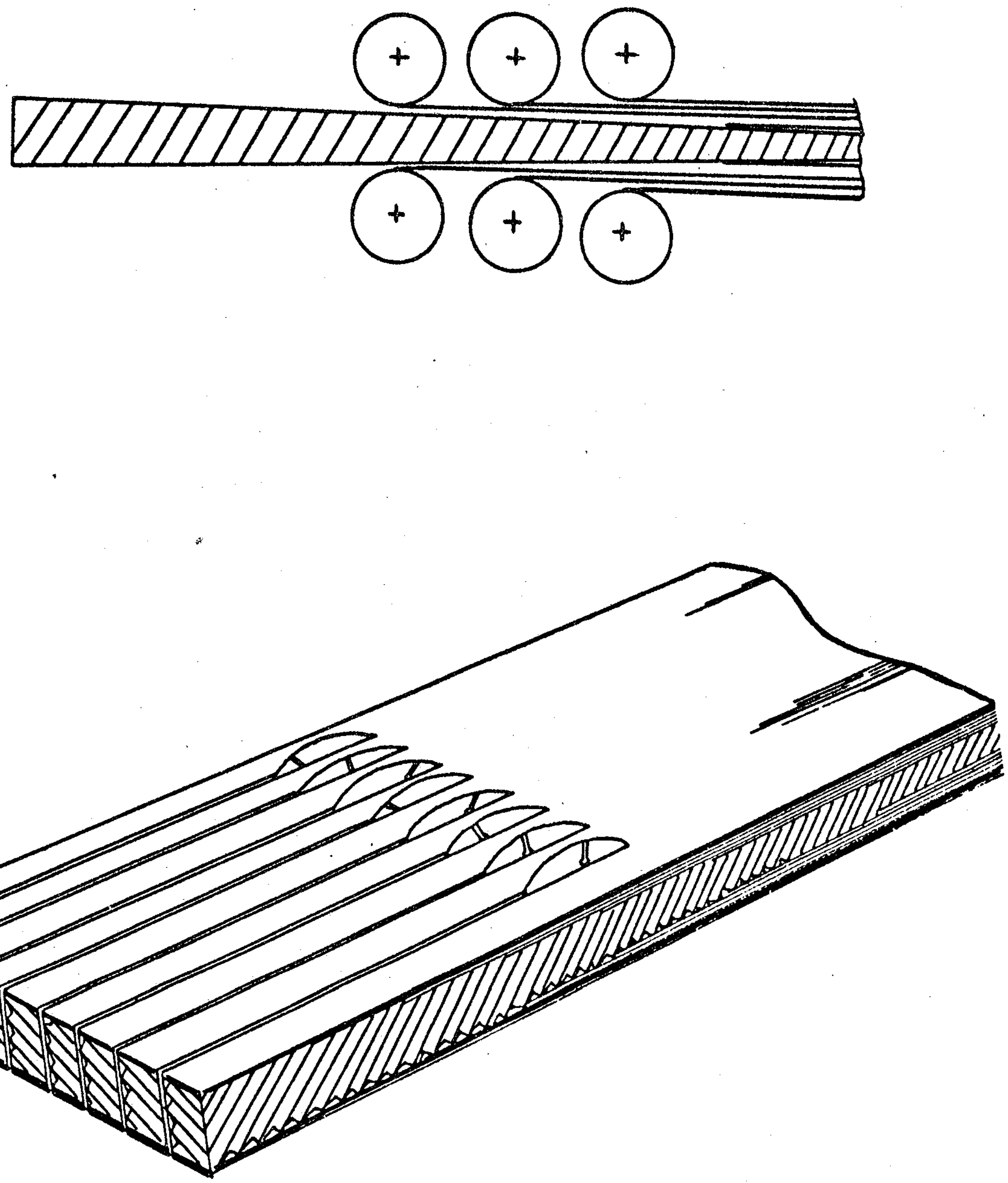


Figure 21. 1985 Stiffener Pultrusion Concept

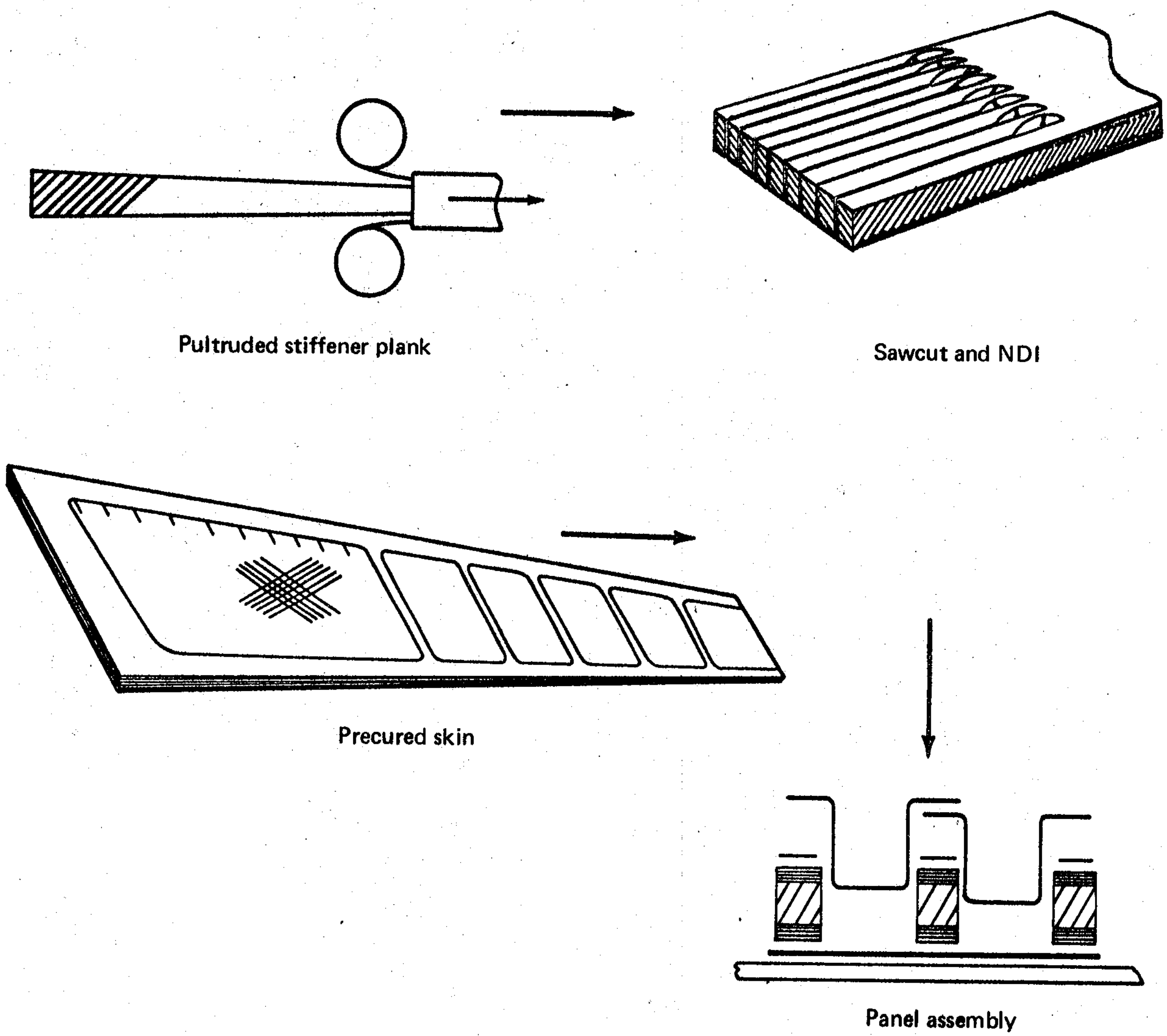


Figure 22. Panel Assembly Concepts

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16. Abstract

The application of graphite/epoxy to commercial aircraft wing structure will require significant technology development. This particular study addresses the design and selection of practical and producible wing panel configurations. A number of stiffened panel configurations were studied. Each configuration was assessed in terms of structural efficiency, practical design features. Structural efficiency (panel weight) analysis was based on compression end loads of 2.63 Mn/m (15,000 lb/in) and allowable ultimate strain of 4000 μ in/in. The structural efficiency of all the panel configurations evaluated is nearly identical. Weight savings of the composite wing panel designs are approximately 25% greater than structurally equivalent aluminum designs.

The final selected panel designs for the upper and lower wing surface are hat and blade-stiffened respectively. Selections of these designs are strongly influenced by manufacturing feasibility. A cost saving of 20% over aluminum wing cover panels is predicted, based on 200 aircraft and a projected automated composites manufacturing capability.

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