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DESIGN OF A COMPOSITE WING EXTENSION FOR A GENERAL AVIATION AIRCRAFT

FINAL REPORT

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DESIGN OF A COMPOSITE WING EXTENSION

FOR A GENERAL AVIATION AIRCRAFT

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SUMMARY

A composite wing extension was designed for a typical general aviation aircraft to improve lift-curve slope, dihedral effect, and lift-to-drag ratio. Advanced composite materials were used in the design to evaluate their use as primary structural components in general aviation aircraft. Extensive wind tunnel tests, conducted at the Texas A&M University Low-Speed Wind Tunnel, were used to evaluate six extension shapes. The extension shape chosen as the best choice was 28 inches long with a total area of 17.17 square feet. Subsequent flight tests, performed by engineers from the Cessna Aircraft Company, showed the wing extension's predicted aerodynamic improvements to be correct. The structural design of the wing extension consisted of a hybrid laminate - carbon core with outer layers of Kevlar - layed up over a foam interior which acted as an internal support. The laminate skin of the wing extension was designed from strength requirements, and the foam core was included to prevent skin buckling. A joint lap was recommended to attach the wing extension to the main wing structure. Narrow layers of Kevlar could be adhered to the composite wing extension and fastened to the aluminum wing. Problems associated with lightning and corrosion were also incorporated into the final design.

TABLE OF CONTENTS

NOMENCLATURE

A = Area of Wing Extension Cross-Section

a = unloaded plate edge

b = plate edge where buckling load is applied

E = Young's Modulus for an Isotropic Material

E_x = Longitudinal Young's Modulus

E_y = Transverse Young's Modulus

G_{xy} = Shear Modulus

I_{xx} = Moment of Inertia about z-axis

I_{xz} = Moment of Inertia about x and z axis.

I_{zz} = Moment of Inertia about x-axis.

M_x = Resultant Moment about x-axis.

M_y = Resultant Moment about y-axis.

M_z = Resultant Moment about z-axis.

M = half waves in a buckled plate in x-direction

N_x = Longitudinal Force Resultant

N_y = Transverse Force Resultant

N_{xy} = Shear Force Resultant

n = Number of half waves in a buckled plate in y-direction

P_i = force in a longitudinal stringer

q = shear flow

t = skin thickness

V_x = Resultant shear in x-direction

V_z = Resultant shear in z-direction.

x = Horizontal coorinate in cross-section

z = Vertical coorinate in cross-section.

σ_{yy} = Bending stress in the wing cross-section

σ_{xy} = Shearing stress in the wing cross-section

ε_x = Longitudinal strain

ε_y = Transverse strain

γ_{xy} = Shear strain

ν = Poisson's Ratio for an isotropic material

ν_{xy} = Major Poisson's Ratio

LIST OF FIGURES

LIST OF TABLES

1.0 INTRODUCTION

Advanced composite materials are utilized safety and effectively in a great number of aircraft applications today. Composite materials are used for secondary structural components in general aviation aircraft but rarely for primary structural components. The principal objective of this investigation was to design a wing extension for a typical general aviation aircraft and to demonstrate the possible benefits arising from the use of an advanced composite material in it.

Several composite materials and design concepts were evaluated for the preliminary structural design. The optimum design choice had to be weight effective, meet certain structural requirements and be the optimum design for application to general aviation aircraft, given the chemical and physical limitations of composite materials, and the in-service environment of the aircraft.

The external geometry of the wing extension was established from aerodynamic considerations. Wind-tunnel researchers tested five candidate configurations in the Texas A&M University 7ft. by 10 ft. Low-Speed wind tunnel. A 1/7 scale model of a general aviation aircraft was used to test the wind extensions at a range of wind tunnel conditions, providing force and moment data to evaluate the cruise, climb and glide performance of each wing extension.

The NASA Langley Research Center sponsored this work through a research grant (NAG-1-184, "Research on Composite Wing Extensions for General Aviation Aircraft') with the Texas A&M Research Foundation over the period of

May 1, 1981 to August 31, 1983. Investigators from both Texas A&M University and the Cessna Aircraft Company participated in the research effort. The initial phase of the study consisted of an aerodynamic investigation to determine the wing extension's best external configuration for the gen-aviation aircraft chosen for this investigation. Once the wing extension aerodynamic geometry had been established, a structural design and analysis effort was conducted to investigate the possible use of an aluminum alloy and several candidate advanced composite materials for the construction of the wing extensions. The final phase of the project was to have been the fabrication ground testing and flight testing of both the aluminum wing extension design and the composite material design. Two versions (both aluminum) of the wing extension have been manufactured and flight tested. Fabrication and testing of the composite material design will be performed at a later date.

The results of the earlier wind tunnel studies have been reported previously by Mr. Oran Nicks, Director of the Texas A&M Wind Tunnel Facility, in the 1982 Texas A&M Research Foundation Report TR-8203. The report contained here summarizes the previously reported wind tunnel study and a detailed account of the structural design and analysis phase of the project. Thus it will be submitted as a final report of the findings associated with NASA Grant NAG-1-184.

1.1 BACKGROUND

Significant progress has been made in the development of advanced composite materials and of design methods applicable to aircraft construction. The NASA sponsored Aircraft Energy Efficiency Program (ACEE) is a major effort to encourage the application of composites and related technologies to commercial transports, so that significant increases in fuel efficiency can be achieved. There has been no similar program program directed to the general aviation application of composite materials. However, NASA research in the area and many of the ACEE results should be applicable to general aviation needs.

Composite materials offer a direct benefit in the design of structural components as the properties of appropriately designed composite materials have superior strength-to-weight ratios, and produce lower part counts than materials currently in common use. A subtle, but equally important benefit accrues from obtaining aerodynamically desired shapes and surface contours conductive to reducing friction drag. Many manufacturing experts believe that in the long run manufacturing costs may be reduced by the increased volume use of composites due to lower part counts, and the ease of manufacturing complex shapes. At present, however, manufacturing procedures, FAA certification, and long-life requirements imposed on composite components can be expensive and often overshaw these potential gains.

1.2 PURPOSE

The investigation was designed to parallel the work being conducted on the NASA sponsored ACEE program. Advanced composite materials were investigated to determine and demonstrate their possible use as primary structures on general aviation aircraft. The wing of an existing typical general aviation aircraft was redesigned by extending the wing to provide for improved stall characteristics and dihedral effect. Aerodynamic testing was performed in the Texas A&M University Low-Speed wind tunnel using a 1/7 scale model aircraft provided by the Cessna Corporation. Various wing extension shapes were tested to obtain the lift, drag and stability information required to assess the aerodynamic gain for each particular geometry. Once the best aerodynamic geometry was established through win tunnel studies, the merits of a composite material extension was investigated by designing the extension for three candidate composite materials. The composite design appearing to be the most promising plus an aluminum extension were to have been fabricated and tested in both ground tests and flight tests, but thus far only an aluminum extension has been fabricated and flight tested. The preliminary results of those flight tests indicated that the wing geometry developed during the wind tunnel phase of the program did produce significant improvements in aircraft performance over the production wing extension.

2.0 SUMMARY OF WIND TUNNEL TESTS

Texas A&M researchers tested six candidate wing extension geometries to establish a configuration capable of improving the aircraft's lift-to-drag ratio and its stall characteristics. Based upon the results of these wind tunnel tests, the geometry for an effective wing extension was established which increased the wing's maximum lift coefficient, lift-curve slope, maximum lift-to-drag ratio, and overall wing efficiency. Wind tunnel tests were conducted on a 1/7 scale model of a single-engine high-wing monoplane during two wind tunnel entries. A total of 138 runs were made during the eighty hours of wind tunnel testing. Data were measured with the wing extension as a primary variable and a model buildup approach was used such that wing-body data could be obtained before adding tail surfaces. Second order characteristics attributable to wing extensions were studied by examining changes in flap and control settings. All aero dynamic data were reduced to coefficient form, providing comparative data for each wing extension.

Figures 2-1 through 2-7 contain a summary of the geometrical details of the orginal wing and the six candidate wing extensions. One of the wing extensions, designated as E1, was a simple rounded tip used as a baseline for comparison, and E5 was the production drooped wingtip configuration The other four extensions were designed to increase the wing area to provide improvements in the lift-drag ratio and the stall characteristics. Configurations E2 and E6 are similar with the exception that E2 has a spanwise dimension of 36 inches while E6 has a span of 28 inches. Wind tunnel tests for configuration E6 were performed at a separate test date after the preliminary tests on configuration E1 through E5. Stability improvements

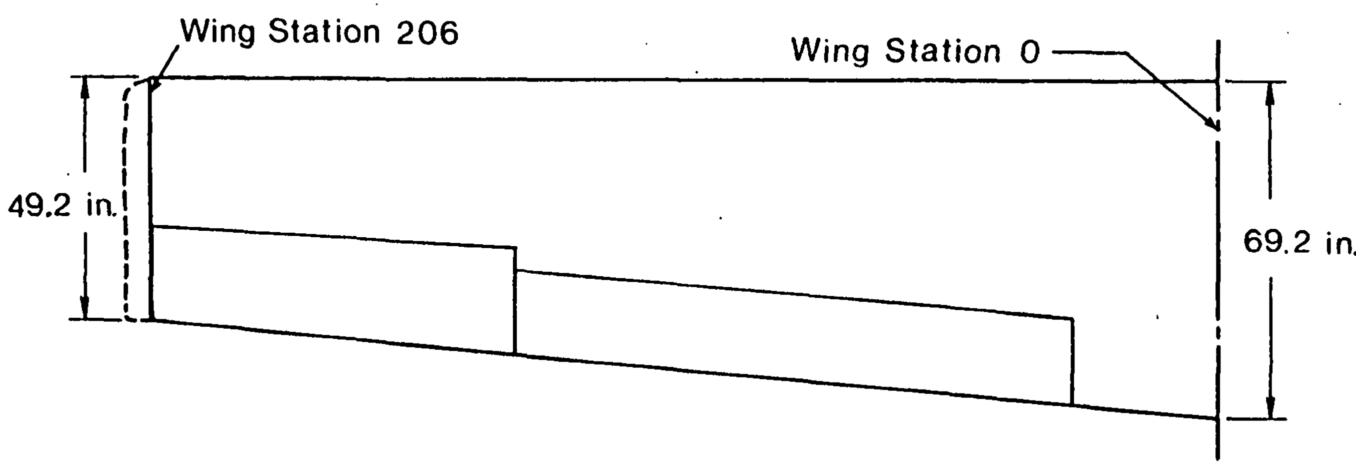

Figure 2-1. Basic Wing Geometry Chosen For Extension Design

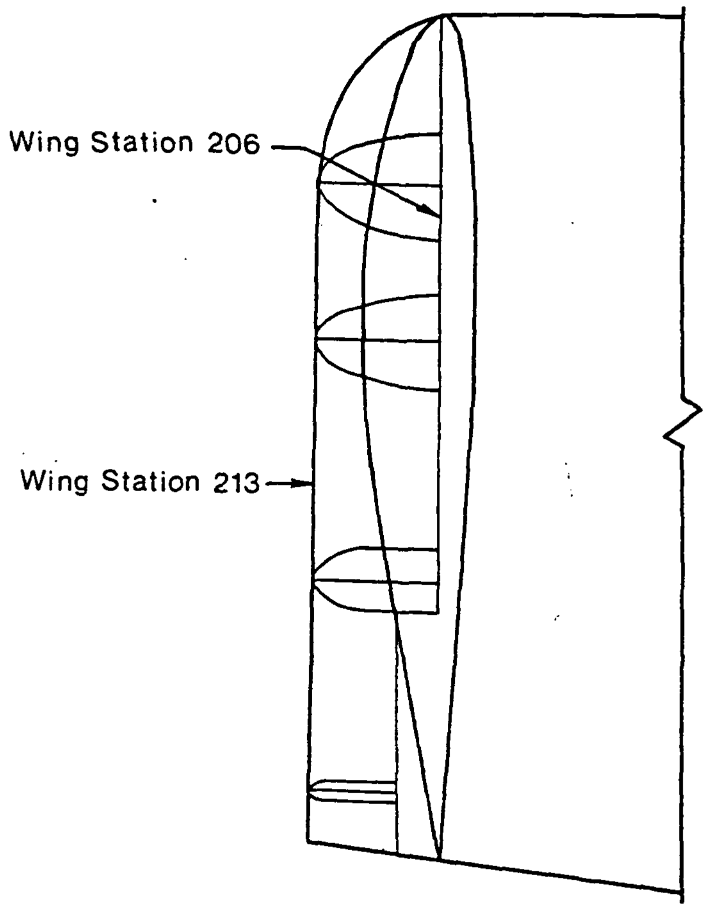

Figure 2-2. Planform of Wing Extension E1 - Wing Extension E1 has a total area of 5.72 ft.^{2} and increases the total wing area to 175.0 ft.^{2} with an aspect ratio of 7.20.

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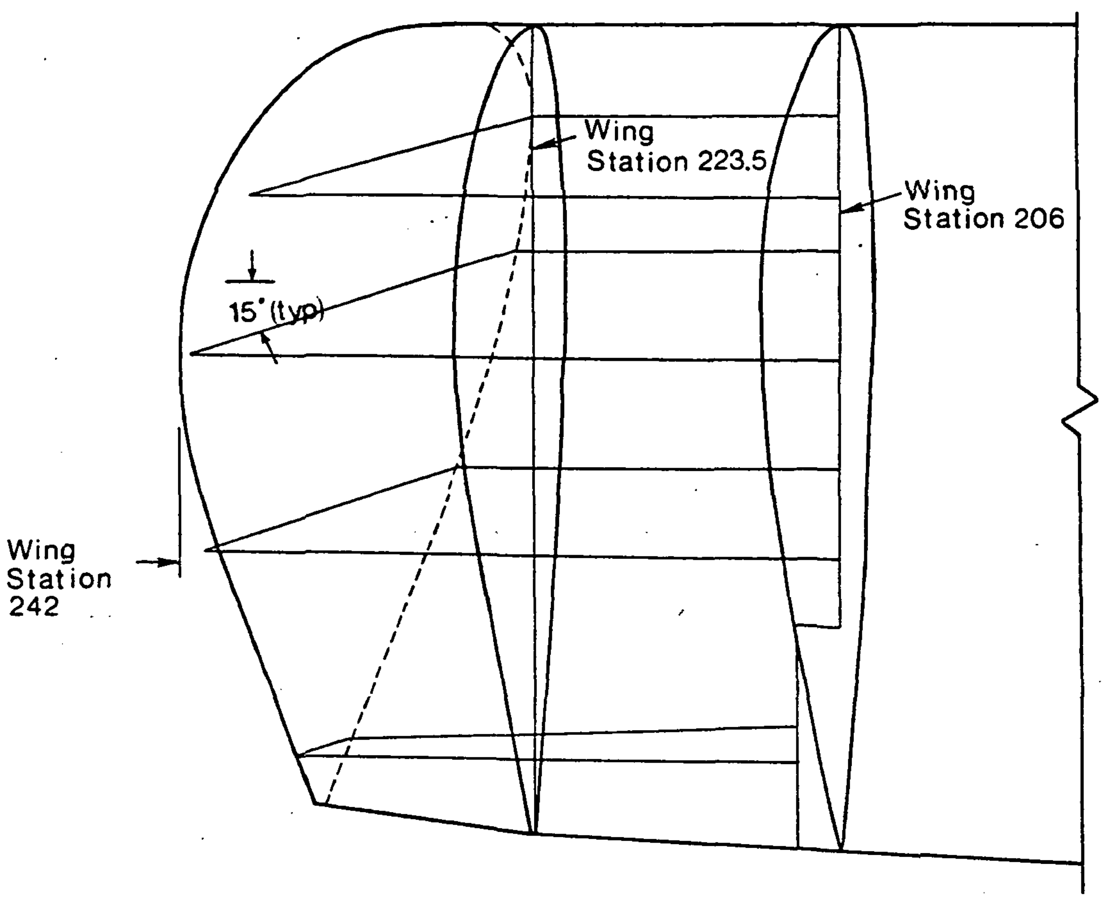

Figure 2-3. Planform of Wing Extension E2 - Wing Extension E2 has a total area of 23.13 ft.^{2} and increases the total wing area to 192.4 ft.^{2} with an aspect ratio of 8.44.

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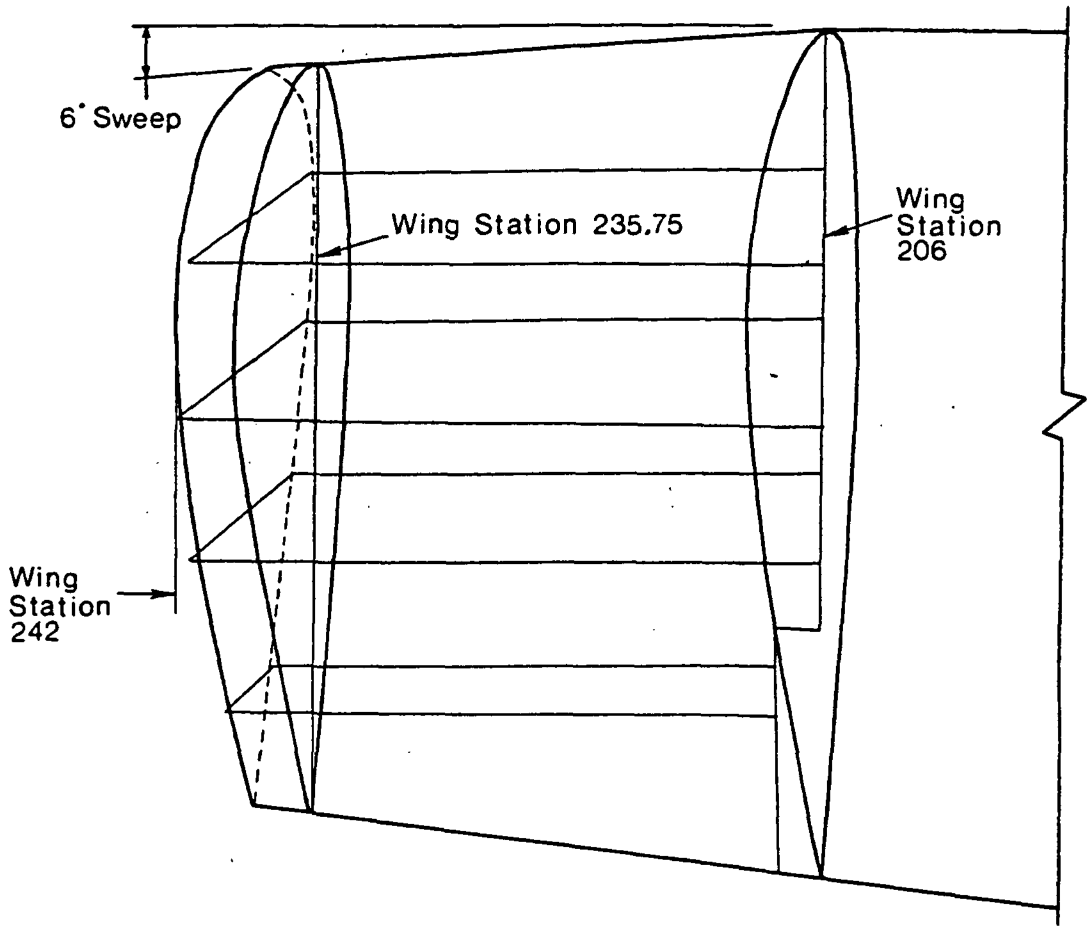

Figure 2-4. Planform of Wing Extension E3 - Wing Extension E3 has a total area of 22.15 ft.^{2} and increases the total wing area to 191.4 ft.^{2} with an aspect ratio of 8.50.

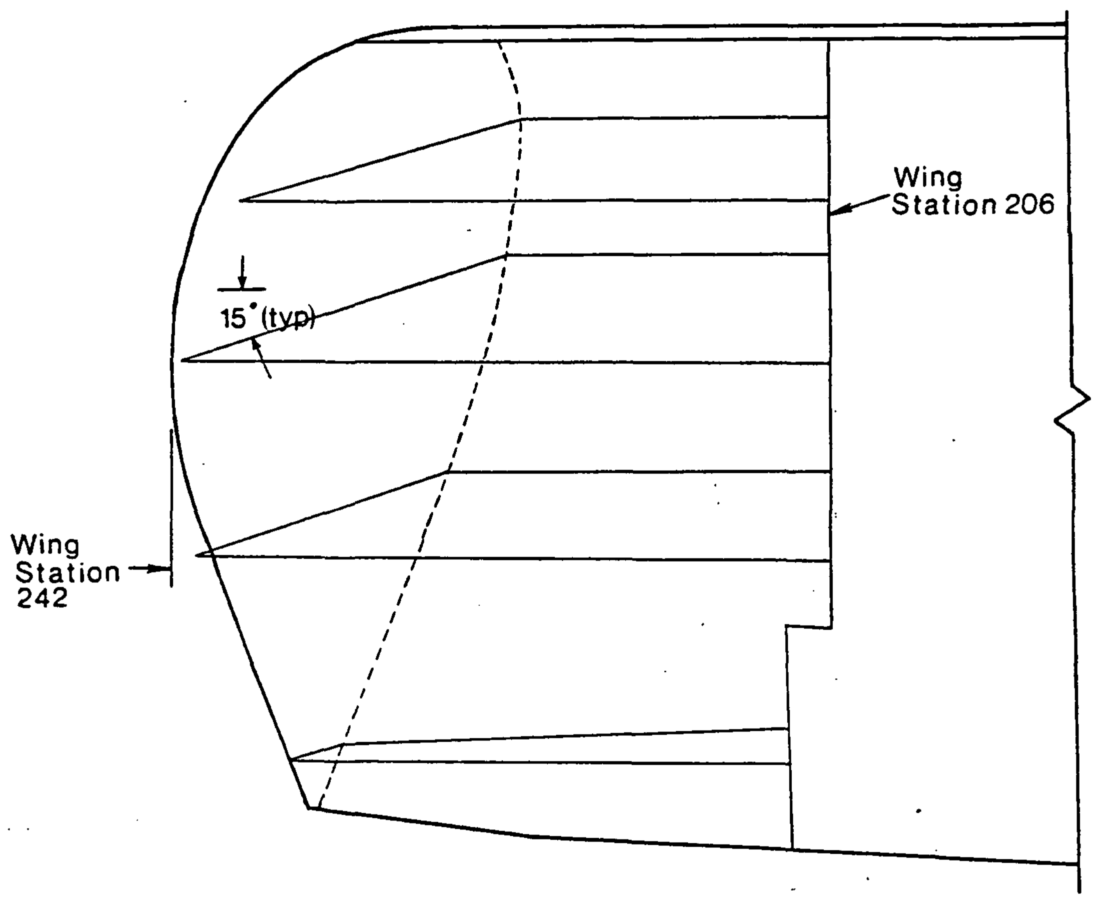

Figure 2-5. Planform of Wing Extension E4 - Wing Extension E4 has a total area of 25.46 ft.^{2} and increases the total wing area to 194.7 ft.^{2} with an aspect ratio of 8.34. The basic tip shape is the same as E2, but a leading edge modification has been added.

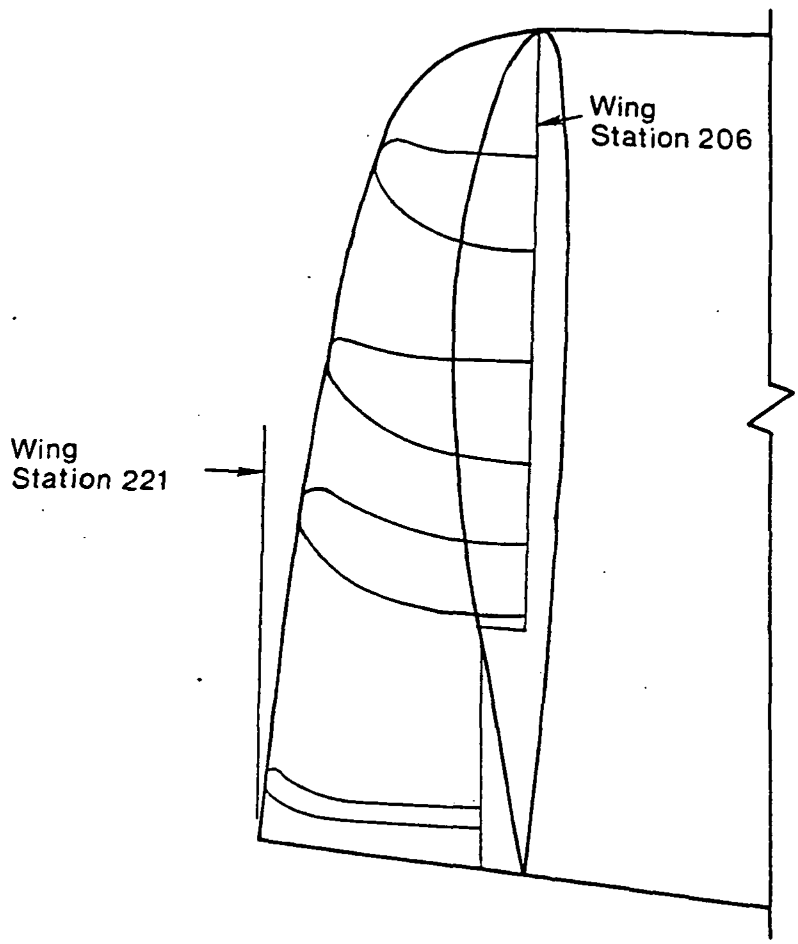

Figure 2-6. Planform of Wing Extension E5 - Wing Extension E5 has a total area of 7.36 ft.^{2} and increases the total wing area to 176.6 ft.^{2} with an aspect ratio of 7.67.

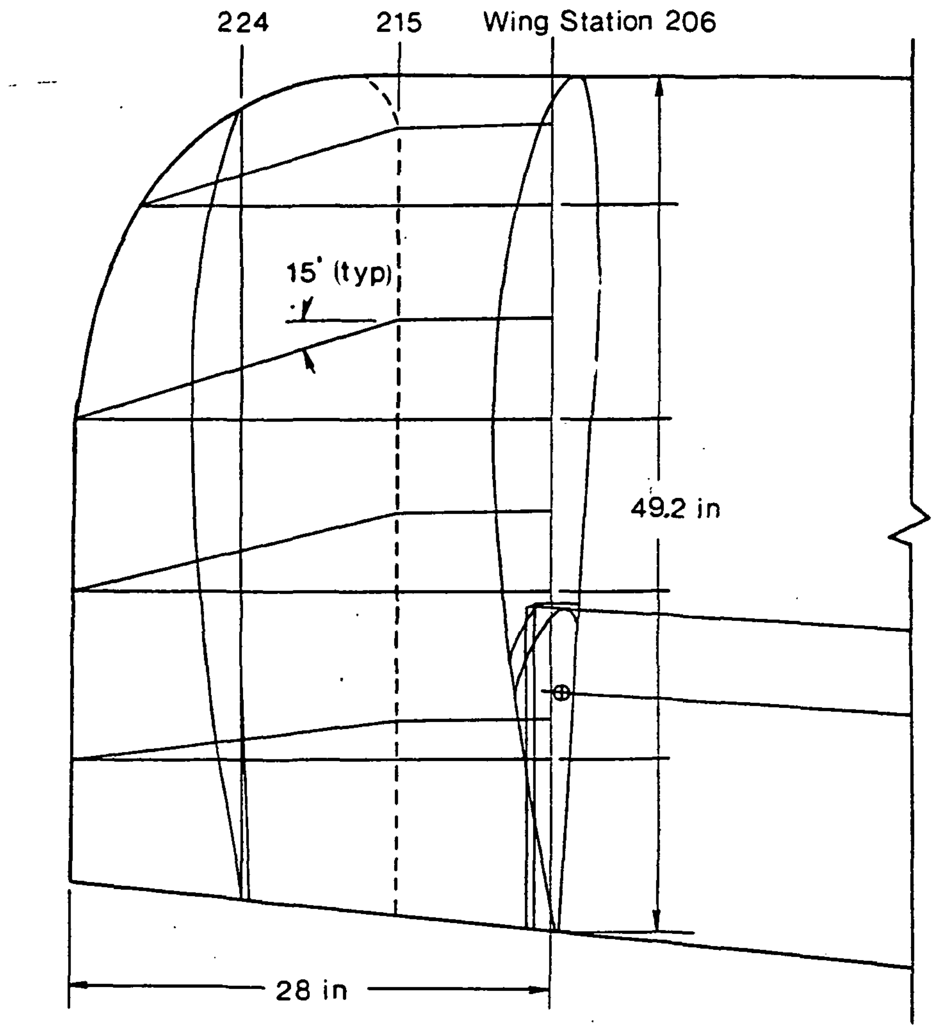

Figure 2-7. Planform of Wing Extension E6 - Wing Extension E6 has a total area of 17.17 ft.^{2} and increases the total wing area to 184.44 ft.^{2} with an aspect ratio of 8.16.

were achieved through an increase in the effective wing dihedral. In addition, a NASA leading edge modification, designed to extend the control range of the wing at high angles-of-attack, was incorporated into configuration E4 to determine its influence on the wing performance characteristics.

Measured improvements, relative to the E1 baseline configuration, were obtained for the four wing extensions, E2 through E4 and E6, in the maximum lift coefficient, lift curve slope, and maximum lift-to-drag ratio. In addition, as much as 5.5° of effective dihedral was obtained with some configurations. Improved stall characteristics were observed for the NASA leading edge modification of wing expansion E4.

It was apparent from the performance and flow visualization data that the wing extensions with sharp-edged tips provided the most favorable combinations of results. This was attributed to the sharp edge of the tip which prevented a great deal of the pressure leakage from the lower surface of the wing and insured vortex formation as far outboard as possible. Even though the sharp edge of the wing extension should increase the wing's parasite drag, the wind tunnel test results indicated a reduction in induced drag and thus a reduction in total drag as well as an increase in overall wing efficiency.

As mentioned earlier, only five configurations were tested orginally. After reviewing the orginal wind tunnel tests, Cessna engineers were convinced that the aerodynamic gains of configuration E2 could be accomplished with a shorter version. Thus, E6, was designed by maintaining the basic geometry of E2 but with a span of 28 inches rather than 36 inches.

The sharp edge tip was produced by slicing upward from the lower surface of the wing parallel to the chord line at a 15° angle with the lower surface, allowing the plan-view shape of the tip to be defined by the contour of the upper surface where the plane passes through. Based upon the comparison of the wind-tunnel data associated with the six configurations tested, wing extension E6 was selected for the structural design and flight test phases of the program.

3.0 TECHNICAL APPROACH TO STRUCTURAL DESIGN

A useful structural design of a composite wing extension depends on factors such as the design airloads and an accurate prediction of the in-service environment of the aircraft. For a final evaluation of the design, precise information concerning the airloads on a production wing with a wing extension would also be required. Due to the nature of composite materials, the designer has a great deal of flexibility in the design of not only the actual structure but the material as well. The first step in the design process was to design a composite laminate satisfying the strength requirements imposed on the wing extension. For this step, a preliminary stress analysis was necessary. Secondly, three structural design concepts were evaluated in terms of strength, stability, and weight. These consisted of a simple shell construction, a shell with a foam core for internal support, and a shell with longitudinal stringers as internal support.

3.1 DESIGN LOADS

As indicated, the final aerodynamic geometry for the wing extension was that of model E6 summarized in the previous section of the report. Exact information was not available for the loads on a production wing with the extended wingtip. The design aerodynamic loads were interpolated from the loading on a production P210 aircraft with the drooped wing extension of the E5 configuration. Figure 3-1 contains a summary of the aerodynamic loads at the attachment section of the wing extension.

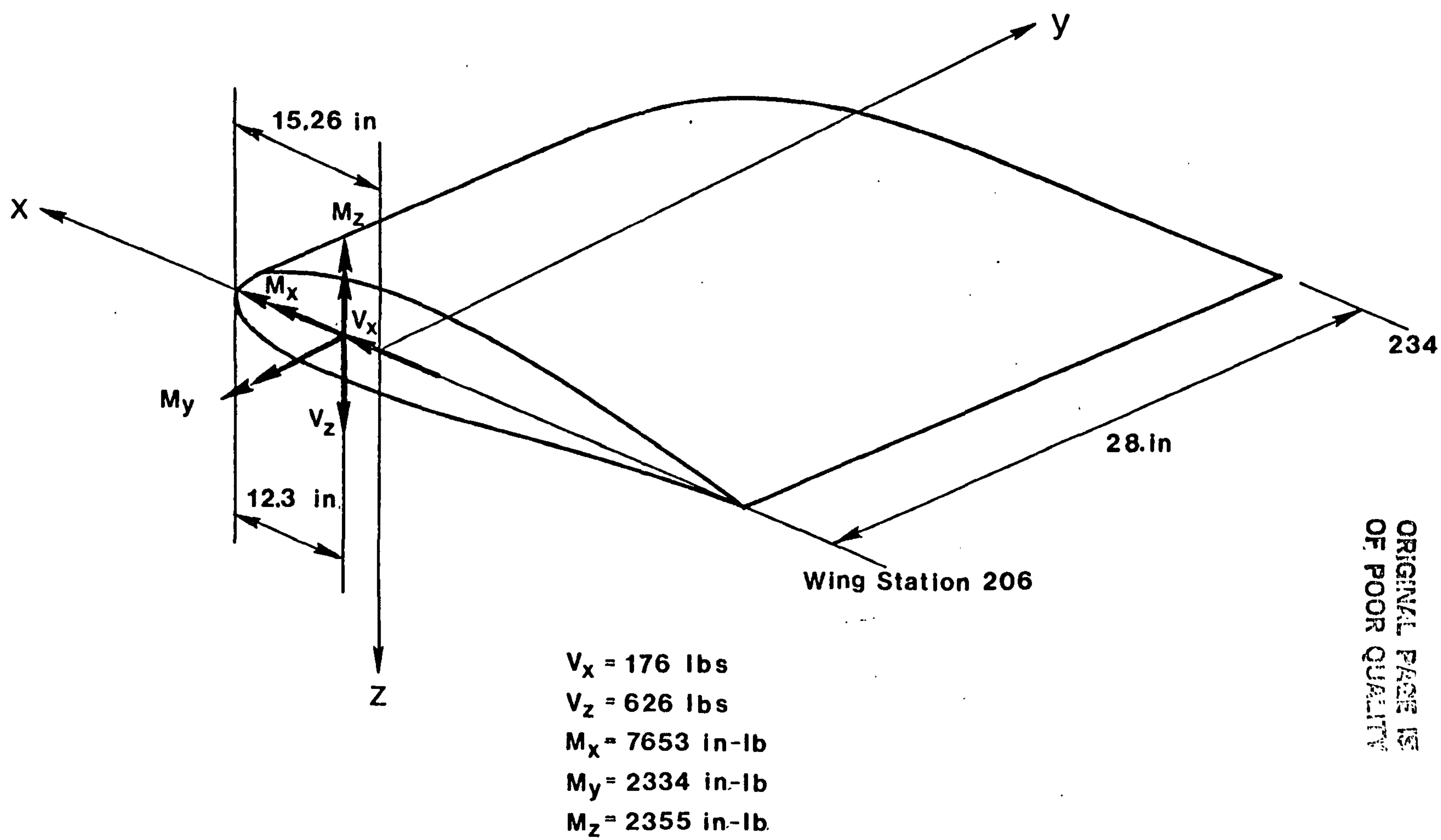

Figure 3-1. Attachment Point Design Airloads.

All forces and moments acting on the interface of the attachment point were assumed to be acting through the aerodynamic center. In addition, the wing extension was assumed to be an elastic structure with the elastic axis coincident with the line joining the shear centers of the various cross sections.

3.2 STRESS ANALYSIS

The critical aspect of the wing extension design was the buckling stability of the skin. Because of the low airloads at the attachment station, initial structural strength estimates indicated the wing extension would buckle long before it yielded.

3.2.1 BENDING STRESS ANALYSIS

The entire wingtip structure was designed to satisfy the internal force conditions that exist at the attachment point. The wing was analytically treated as a beam (length large compared to cross-sectional dimensions) and the wing extension was treated as a section of the elastic beam. For a given pressure distribution over the wing, internal stress resultants can be found at any cross-section along the length of the wing. An equation for the bending stress can be found from the Bernoulli-Euler theory of bending. In this theory it is assumed that cross-sectional planes of the beam remain plane and normal to the axis of the beam as it deforms. This is equivalent to postulating a linear strain distribution over the cross section. The linear strain assumption agrees with lamination theory [Ref 1]. Additional assumptions were that all stress components other than σ_{yy}

were negligibly small compared to σ_{yy} and that the material was linearly elastic. It was assumed also that the beam was homogenous and there were no axial forces of temperature gradients [Ref 2]. Given these assumptions, the equation for the bending stress in a monocoque section is:

$$\sigma_{yy} = \frac{(M_z I_{xx} + M_x I_{xz})\, x + (M_x I_{zz} - M_z I_{xz})\, z}{I_{xx} I_{zz} - I_{xz}^2}$$

The compressive stress resultant on the laminate is found by integrating the bending stress through the thickness of the laminate yielding:

$$N_y = \int_{-t/2}^{t/2} \sigma_{yy}\, dz$$

3.2.2 SHEARING STRESS ANALYSIS

It was assumed that a wing extension with a monocoque structure may be treated as a thin-walled hollow section. In a thin-walled section, the resultant shearing stresses must be in the direction of the tangents to the boundary at the inner and outer boundaries of the thin wall. Since these directions are very nearly parallel in a thin walled section, it is assumed that the resultant shearing stresses are constant throughout the wall and are in a direction tangent to the median line drawn through the middle of the wall thickness.

With these assumptions, the shear flow, q, at a point in the cross-section can be defined as the product of the shearing stress and the wall thickness, by the following equation:

$$q = \sigma_{xy} t$$

The applied torque on the thin-walled closed section can be related to the area of the cross-section and the shear flow in the cross-section by the application of the equilibrium equations yielding:

$$\sum M_y = 2Aq$$

When shear resultants are introduced into the cross-section the equilibrium moment equation for the section can be written as:

$$M_y - V_x z_c + V_z x_c + 2\sum A_{ij} q_{ij}^{(0)} + 2\sum A_i q_i^{(1)} = 0$$

These terms are represented in Figure 3-2. The shear flow, $q_{ij}^{(0)}$, arising from the load carried by the longitudinal stringers can be found from a fluid flow analogy where:

$$q_{i,i+1}^{(0)} = q_{i-1,i}^{(0)} - \frac{\Delta P_i}{\Delta y}$$

The load in a longitudinal stringer and the resulting shear flow is pictured in Figure 3-3. The actual shear flow in the section, q, is found by adding the results of the previous two equations:

$$q = q_{ij}^{(0)} + q_i^{(1)}$$

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Finally, the resultant shear force acting on the laminate is found by integrating the laminae stresses through the thickness of the laminate yielding [Ref 2]:

$$N_{xy} = \int_{-t/2}^{t/2} \sigma_{xy}\, dz$$

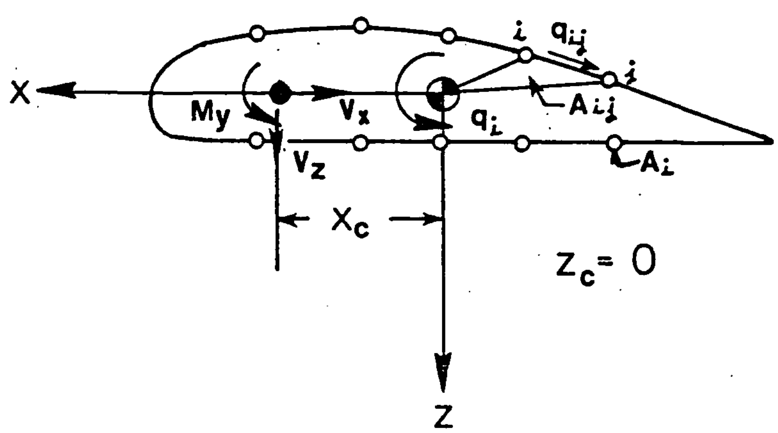

Figure 3-2. Cross-Section Shear Flow.

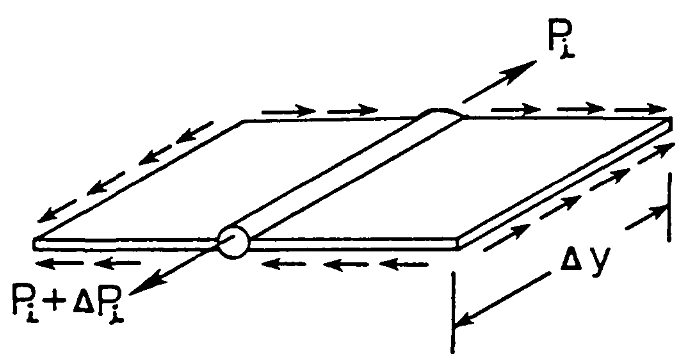

Figure 3-3. Shear Flow Created By Load in a Longitudinal Stringer.

3.3 MATERIAL EVALUATION

Before establishing the structural design of the wing extension, three materials - glass, Kevlar, and carbon - were screened for their use in the design of a composite laminate suitable for the wing extension. Glass was considered, because it is compatible with aluminum parts, is less expensive than the other materials and would possibly be adequate for the low stiffness requirement of the wing extension. Like fiberglass, Kevlar is compatible with aluminum parts, but Kevlar has a much higher strength-to-weight ratio than glass. Carbon has a considerably higher stiffness than Kevlar or glass but is also more expensive, and reacts galvanically with aluminum and thus requires consideration of the corrosion problem.

Woven fabrics were chosen over pre-pregged tapes for the design of the wing extension, because they are easier to handle and use in the fabrication of surface parts. The aerodynamic loads on the wing section are anticipated to be small; therefore, the stiffness degradation of the fabric relative to the tape should present no significant problem.

Material data for the glass, Kevlar, and carbon fabrics chosen were supplied by the Hexcel Corporation. Each candidate material considered had a common F155 epoxy resin matrix. The properties of the F155 epoxy resin are given in Table 3-1. The dry glass transition temperature of the resin is 250^{o}F, well above any temperature in the predicted service environment of a general aviation aircraft. Other favorable

Table 3-1

F155 Epoxy Resin Properties

T_{G} DRY	121°C (250°F)
Equilibrium Moisture Absorption	9.4%
Tensile Strength	.080 GPa (11.6 Ksi)
Tensile Modulus	3.25 GPa (0.47 Msi)
Tensile Strain	5.2%
Fracture Toughness, Kic	.263 MPa m(1.50 Ksi in)
Remarks	High Laminate Strengths Good Sandwich Panel And Metal To Metal Bonding Characteristics

properties include a relatively low moisture absorption and good bonding characteristics. The cure cycle of composite materials with a F155 epoxy resin matrix is shown in Figure 3-4.

The material properties of the three fabric/epoxy composites are presented in Table 3-2. Of the three materials considered, Kevlar has a significantly lower density than either glass or carbon; while the carbon fabric has a higher strength-to-weight ratio than the glass and Kevlar fabrics. Because no statistical meaning could be applied to the material data, the values given in Table 3-2 represent a 20% degradation of material properties to account for the variations common in composite materials.

3.4 LAMINATE DESIGN FOR FIRST PLY FAILURE

The first step in the design process was to find a laminate for each candidate material that could withstand the design aerodynamic loads on the wing extension. A laminate analysis code developed by the General Dynamics Corporation using a maximum strain failure criterion was used to evaluate the laminate design. Program input consisted of material properties and applied loads. The applied loads were multiplied by a safety factor of 1.5. During this phase of the design procedure the skin of the wing extension was designed for strength requirements, with no consideration given to buckling stability.

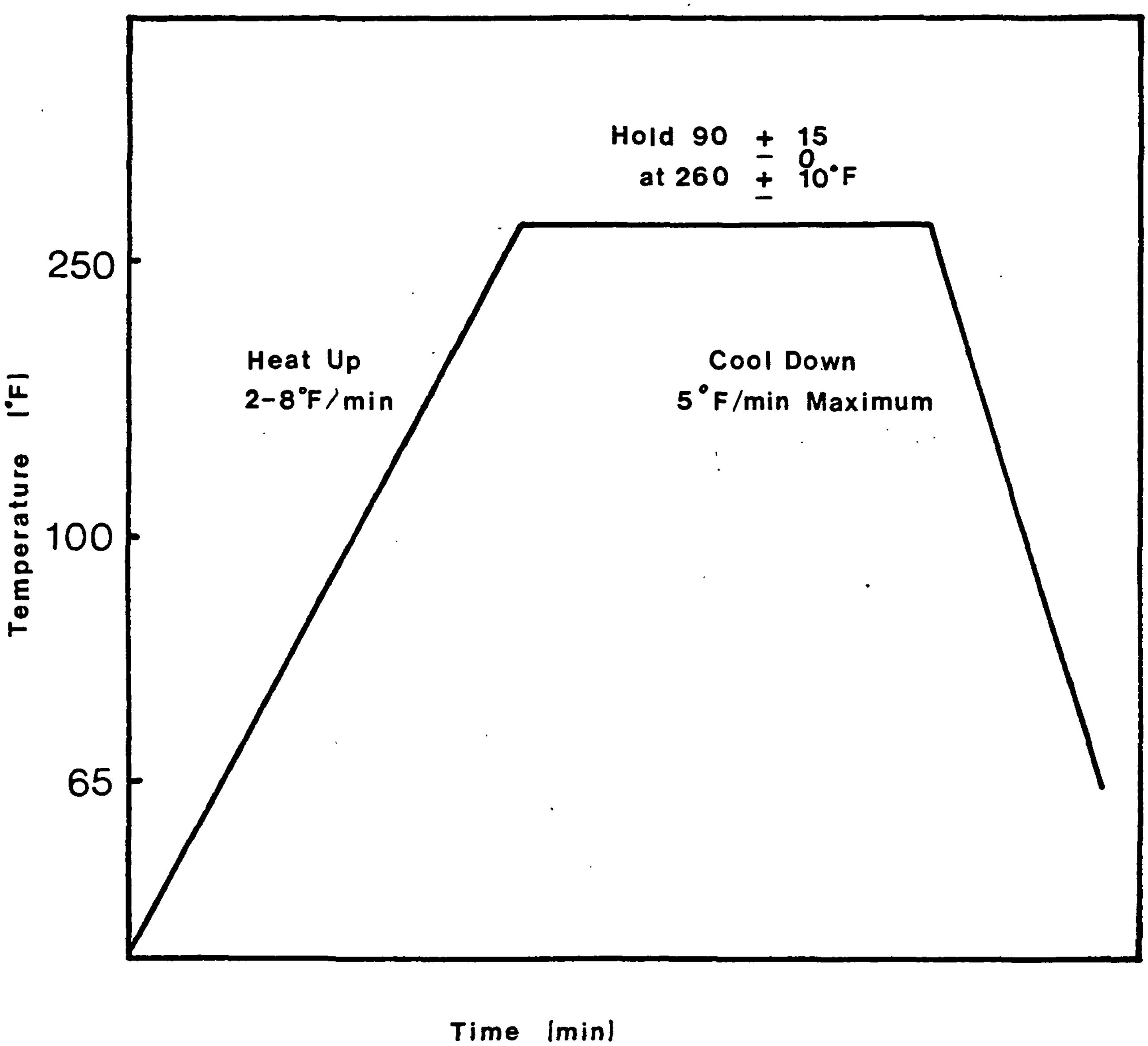

Figure 3-4. Cure Cycle for F155 Epoxy Resin.

Table 3-2

Candidate Material Properties

Geometric Properties

Material	Weave	% Fiber Volume	Ply Thickness mm (in)	Areal Weight N/m^2 (lb/ft^2)
Glass	Satin	45	.203 (.008)	2.97 (.062)
Kevlar	Plain	42	.114 (.0045)	.598 (.012)
Carbon	4-Harness Satin	51	.203 (.008)	1.825 (.038)

Modulus and Strength Properties

Material	Tensile Modulus GPa (Msi)	Tensile Strength GPa (Msi)	Compression Modulus GPa (Msi)	Compression Strength GPa (Msi)
Glass	18.2 (2.64)	.342 (49.6)	19.3 (2.80)	.386 (56)
Kevlar	21.5 (3.12)	.342 (49.6)	21.5 (3.12)	.177 (25.6)
Carbon	47.4 (6.88)	.463 (67.2)	47.4 (6.88)	.469 (68)

Strength-To-Weight Characteristics

Material	Tensile Strength / Areal Weight ($\times 10^9$)	Compressive Strength / Areal Weight ($\times 10^9$)
Glass	.115	.130
Kevlar	.572	.296
Carbon	2.537	.257

The laminates for each candidate material were symmetric and composed of (0°,90°) and ($\pm$ 45°) ply orientations, although each layup was chosen to reduce the angle of orientation between plies. Interlaminar shear stresses are reduced significantly when $\pm\theta$ layers are interspersed between 0° and 90° layers. Delamination is a direct result of interlaminar stresses, and becomes even more critical in cases of compression and shear loadings where stability is the major concern. The magnitude of the interlaminar stresses is related to the magnitude of the mismatch in Poisson's Ratio, elastic modulus, and shear modulus between the plies, and the stacking sequence of the laminate. Reducing the angle of orientation between each ply will therefore reduce interlaminar shear stress [Ref 3].

Based upon the results of the laminate analysis, presented in Table 3-3 and Figure 3-5, a glass fabric laminate could be fabricated to satisfy the strength requirement, but the weight would be prohibitive. Kevlar laminates have a low weight but do not satisfy the minimum margin of safety. The remaining laminates - carbon (3 plies), carbon and Kevlar (4 plies), glass and carbon (3 plies) - all satisfy the minimum margin of safety. The hybrid laminate of carbon and Kevlar, however, weighs less than the other three laminates.

Table 3-3

Results of Laminate Analysis

Material	Layup	E_x Gpa (Msi)	E_y GPa (Msi)	ν_{xy}	G_{xy} GPa (Msi)
		Average Elastic Laminate Constants			
Glass	[0°, 45°, 0°] 3 plies	15.03 (2.18)	15.72 (2.28)	.424	3.56 (.517)
Kevlar	$[45°, 0°]_s$ 4 plies	13.58 (1.97)	13.58 (1.97)	.583	4.29 (.622)
Kevlar	$[0°, 45°, 0°]_s$ 6 plies	16.53 (2.39)	16.53 (2.39)	.493	3.04 (.441)
Carbon	[45°, 0°, 45°] 3 plies	23.92 (3.47)	23.92 (3.47)	.647	12.75 (1.85)
Glass and Carbon	G C G [45°, 0°, 45°] 3 pliers	21.37 (3.10)	21.37 (3.10)	.474	5.36 (.778)
Kevlar and Carbon	K C C K [45°,0°,0°,45°] 4 pliers	33.09 (4.80)	33.09 (4.80)	.392	4.10 (.595)

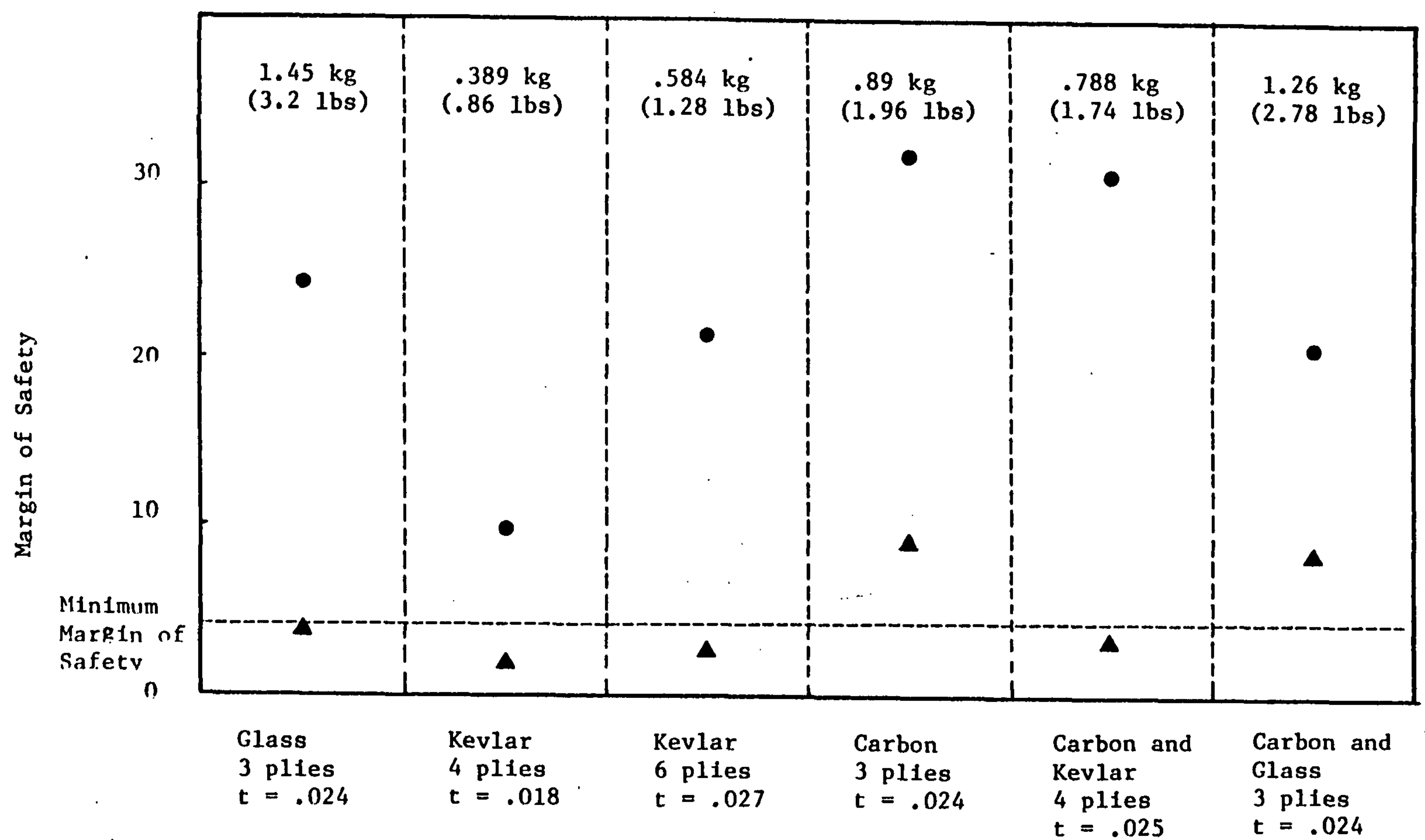

Figure 3-5. Range in Margin of Safety For Wing Extension Materials.

3.5 STABILITY ANALYSIS

The critical buckling load was found for the wing extension by modeling the upper surface as a thin flat plate with all edges simply supported. For plates of this type, the buckling consists of a bulging displacement in the central region of the plate. It is a conservative estimate to say that all edges of the plate are simply supported. A more accurate estimate of the buckling load lies somewhere between the buckling load values for clamped and simply supported edges [Ref 4].

The laminates under consideration are specially orthotropic plates since neither bending-extension coupling nor shear or twist coupling exists. In a specially orthotropic laminate the resultant forces depend only on the in-surface strains, and resultant moments depend only on the curvature of the laminate. These relationships are given as:

$$\begin{Bmatrix} N_x \\ N_y \\ N_{xy} \end{Bmatrix} = \begin{bmatrix} A_{11} & A_{12} & 0 \\ A_{12} & A_{22} & 0 \\ 0 & 0 & A_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_x \\ \varepsilon_y \\ \gamma_{xy} \end{Bmatrix}$$

$$\begin{Bmatrix} M_x \\ M_y \\ M_{xy} \end{Bmatrix} = \begin{bmatrix} D_{11} & D_{12} & 0 \\ D_{12} & D_{22} & 0 \\ 0 & 0 & D_{66} \end{bmatrix} \begin{Bmatrix} K_x \\ K_y \\ K_{xy} \end{Bmatrix}$$

For specially orthotropic plates the compressive buckling load, N_y, is given in Jones [Ref 5] as:

$$N_y = \pi^2 \left[D_{11}\left(\frac{m}{a}\right)^2 + 2\left(D_{12} + 2D_{66}\right)\frac{1}{b^2} + D_{22}\frac{1}{b^4}\left(\frac{a}{m}\right)^2 \right]$$

The compressive buckling curve for the four laminates under consideration is presented in Figure 3-6. The laminate performing the best is three plies of carbon fabric. However, the laminate of Kevlar and carbon was chosen for the wing extension as it weighs less, and the Kevlar plies will help prevent galvanic corrosion between the aluminum wing and the carbon in the wing extension.

The shear buckling load for a simply supported isotropic plate is given in Timoshenko [Ref 6]. Shear buckling resultant, N_{xy}, is related to longitudinal stringer spacing, b, by the expression:

$$N_{xy} = \frac{5.7\,\pi^2}{12\,b^2}\left[\frac{E\,t^3}{(1-\nu^2)}\right]$$

The shear buckling curve for the Kevlar and carbon hybrid laminate is shown in Figure 3-7. Because the spacing required between each longitudinal stringer to prevent shear buckling is unfeasible, the skin thickness was increased from .025 inches to .05 inches to improve the buckling characteristics. Both the compressive and shear buckling curves are shown in Figure 3-8 for a skin thickness of 0.05.

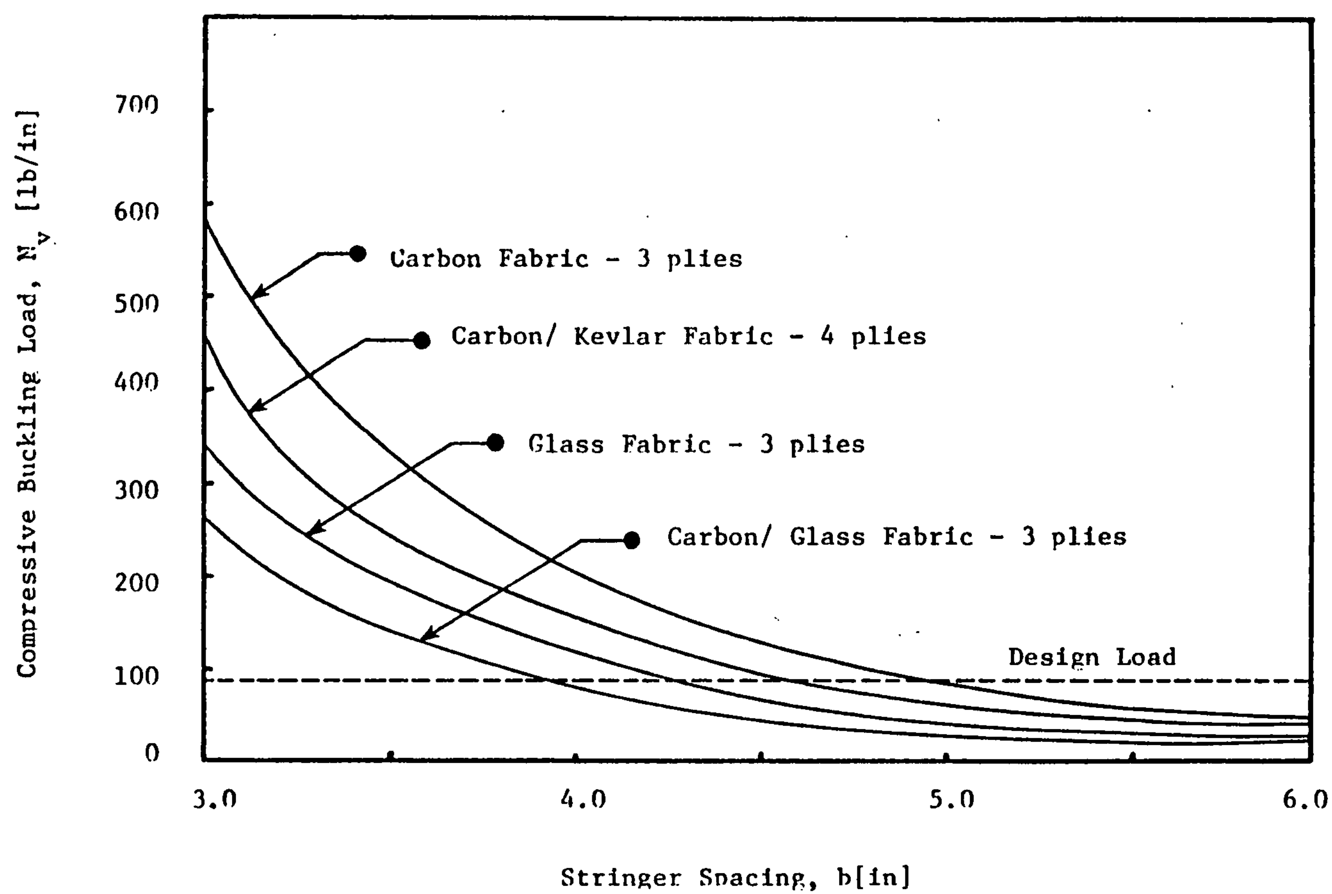

Figure 3-6. Compressive Buckling Curve of Each Laminate Under Consideration For the Wing Extension.

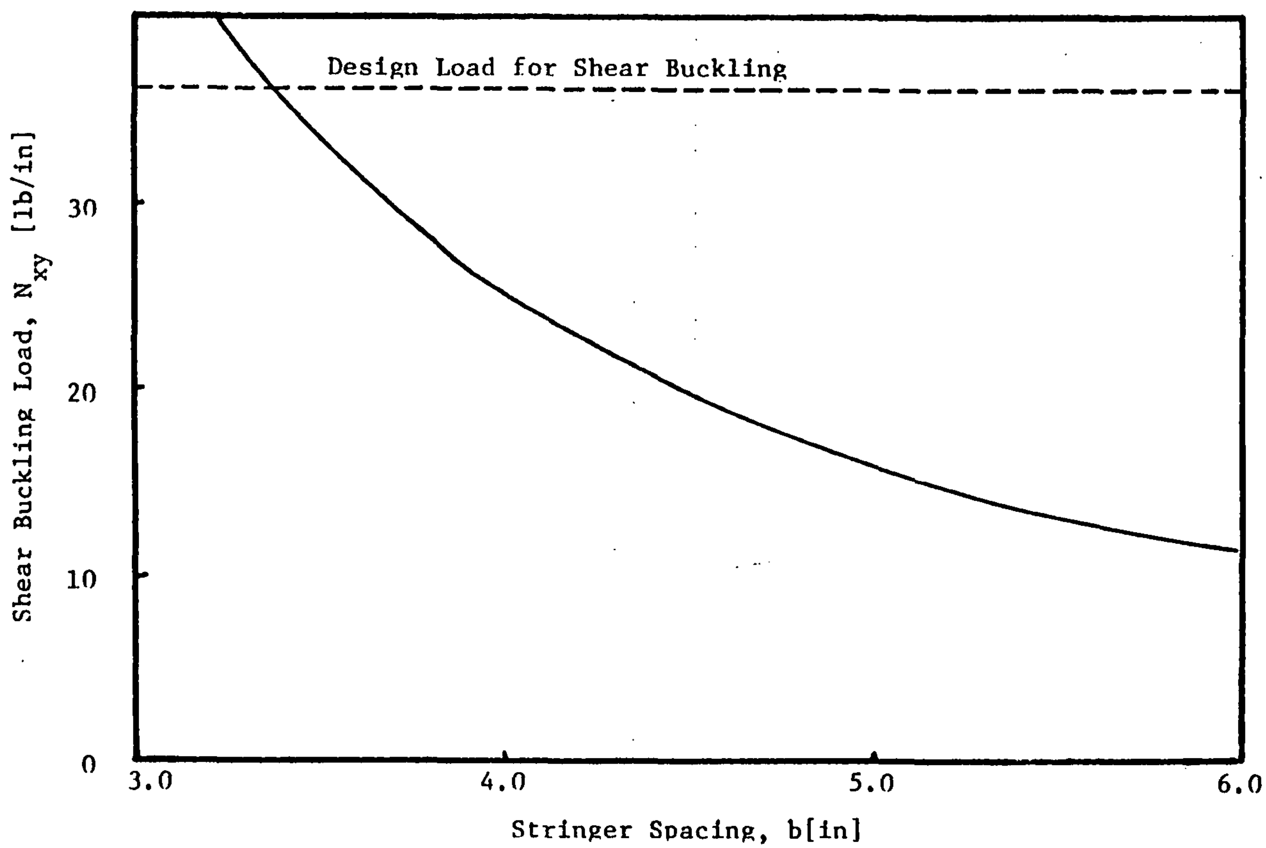

Figure 3-7. Shear Buckling Curve For a Kevlar/Carbon Hybrid Laminate With a Skin Thickness of .025 inches.

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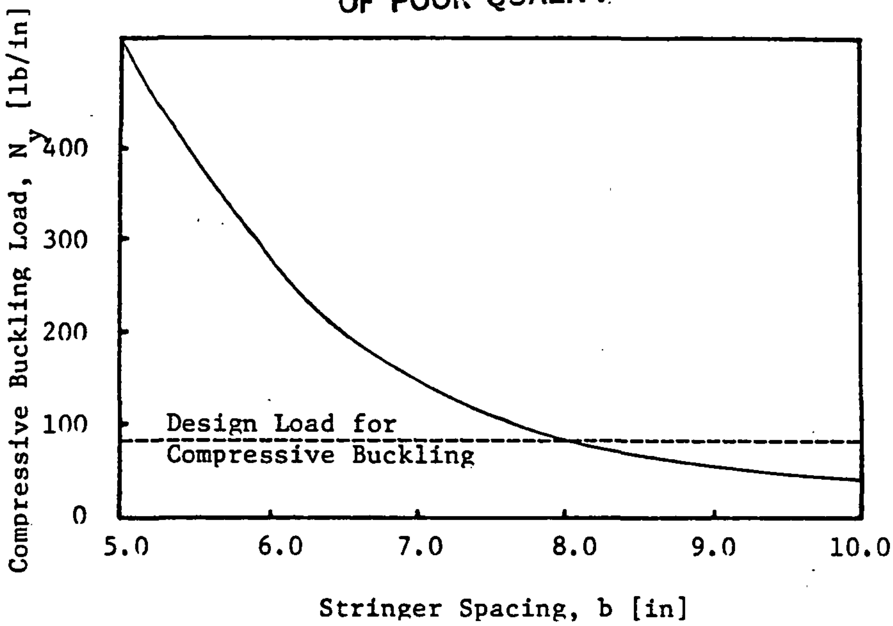
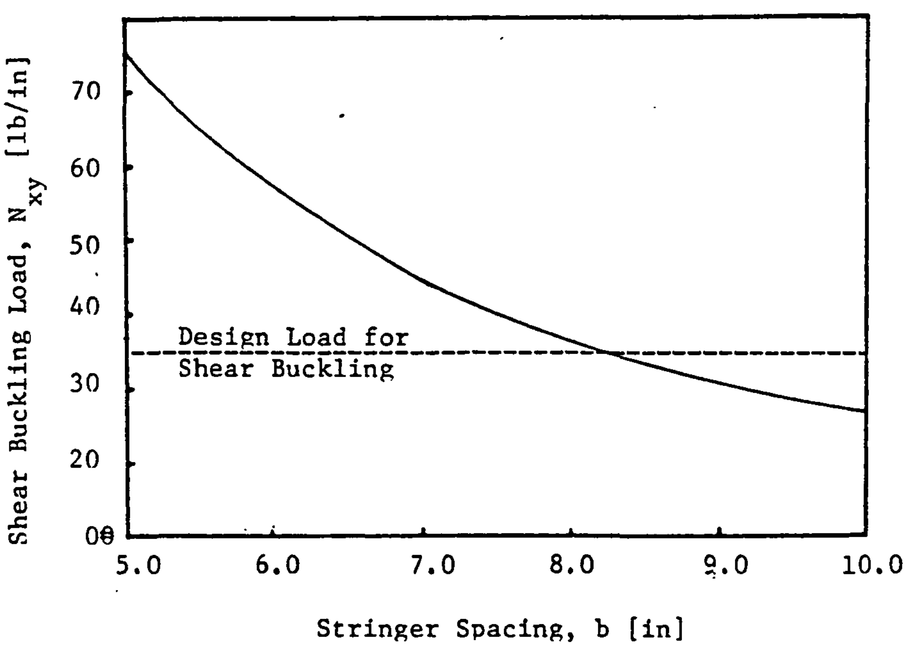


(a)



(b)

Figure 3-8. (a) Compressive and (b) Shear Buckling Curve For a Kevlar/Carbon Laminate With a Skin Thickness of .05 inches.

3.6 STRUCTURAL DESIGN CONCEPTS

Three structural concepts were evaluated for the composite wing extension. A laminate had already been designed to satisfy a first ply failure criterion arising from the bending and shearing stress; however, the overall design of the wing extension would need to prevent buckling of the skin and supporting structure.

Design I was a monocoque structure with a skin thickness large enough to prevent buckling. Design II was composed of a monocoque skin filled with foam. The foam would support the shell-like skin and prevent skin buckling. Design III was a semi-monocoque structure with longitudinal stringer supports to prevent skin buckling. Each of these structural concepts are illustrated in Figure 3-9. Preliminary weight estimates for each design concept are presented in Table 3-4.

Design I would weigh approximately 24.33 pounds which is greater than the estimated weight of an aluminum wing extension. The skin thickness was sized to prevent skin buckling.

Design II has a preliminary weight estimate of 4.24 pounds. This design would be easier and cheaper to manufacture than the other designs. The foam could be molded in the shape of the wing extension, and the laminate skin could be layed up over the foam core. A large percentage of the weight in this design is the foam core which weighs 2.5 pounds.

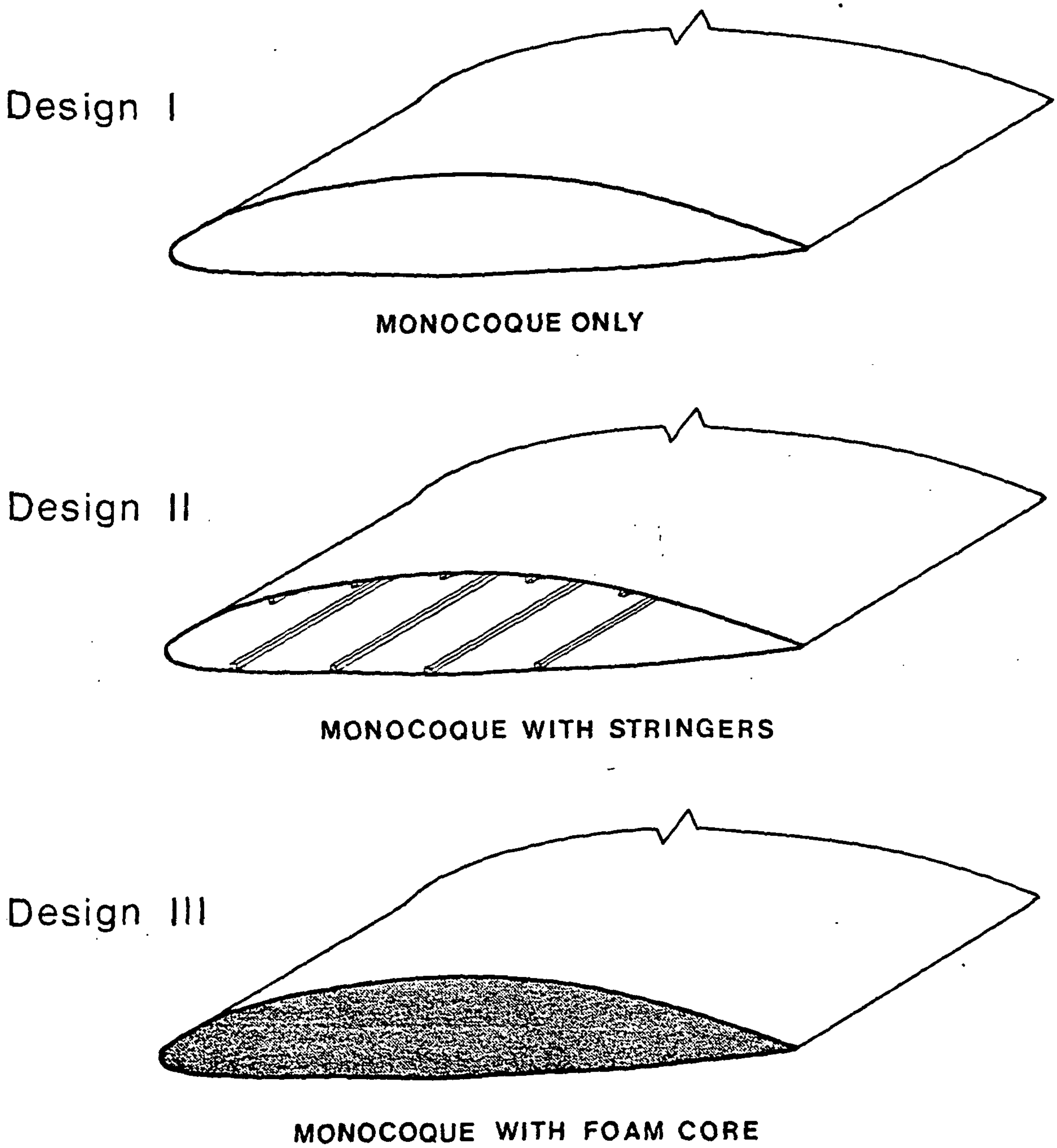


Figure 3-9. Basic Design Concepts Considered For the Wing Extension.

Table 3-4

Preliminary Weight Estimates

of the Wing Extension

Design	E_x GPa (Msi)	Skin Thickness mm (in)	Weight N (lbs)
I. Monocoque	33.10 (4.8)	8.89 (0.35)	108.22 (24.33)
II. Monocoque + foam*	33.10 (4.8)	0.635 (.025)	18.85 (4.24)
III. Monocoque + Stringers+	33.10 (4.8)	1.27 (.05)	18.33 (4.12)
IV. Aluminum (Skin + aluminum stringers)	72.4 (10.5)	0.795 (.031)	58.42 (13.13)

*Foam weighs 11.12 N (2.5 lbs.)
+ Eight stringers were used in this design. Each stringer weighs 3.77 N (.08 lbs.)

The last concept, Design III, weighs 4.12 pounds and uses eight longitudinal stringers bonded to the laminate skin to prevent buckling. The stringers were designed to prevent local buckling, using methods similar to the stability analysis mentioned earlier. Stringer details are shown in Figure 3-10. The laminate for both the stringer and skin has a thickness of 0.05 inches and ply orientations of [45^o_2 (Kevlar), 0^o_4 (Carbon), 45^o_2 (Kevlar)]. Unlike Design II, Design III would have a relatively high part count and would be labor intensive due to the time involved in the layup, compaction process, and autoclaving of each stringer [Ref 7].

Although Design III has the lowest preliminary weight, Design II is recommended for the wing extension because it would be easier to manufacture.

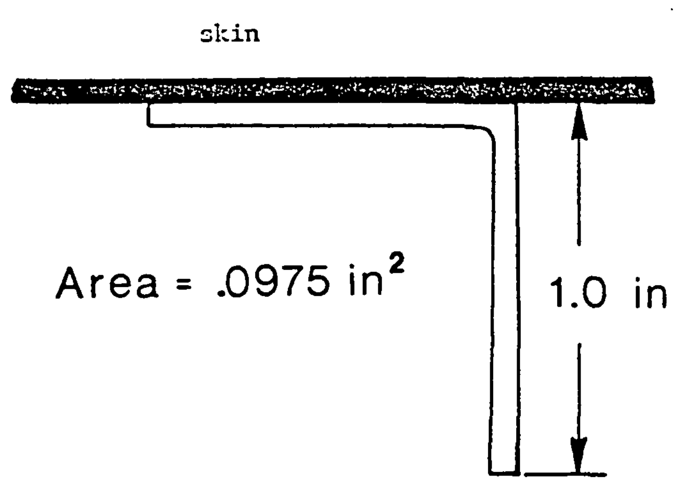


Figure 3-10. Wing Extension Longitudinal Stringer Detail.

4.0 DESIGN CONSIDERATIONS

Lightning, corrosion, and joint design account for major design problems in composite structures. These factors are often minimal design considerations in aluminum aircraft structure, but become problem areas in composite structures due to the physical nature of the materials.

4.1 LIGHTNING

Because composite materials are poor conductors, a system designed to provide electrical paths around the entire perimeter of the structure should be included in the final design. Clark [Ref 8] suggests a 120-mesh aluminum integrated into the laminate to provide an electrical path for lightning. The main advantage of the aluminum mesh is that it can be integrally bonded to the laminate during the cure cycle. The weight of the aluminum mesh (including adhesive and resin required for installation) for one wing extension would be 6.58 N (1.48 lbs). The areal weight of the mesh is 4.12 N/m^2 (.083 lb/ft^2).

4.2 CORROSION

Due to the highly corrosive nature of any contact between carbon-epoxy surfaces and aluminum, special consideration must be given to corrosion protection systems. The layers of Kevlar in the laminate protect the carbon-epoxy from contact with aluminum. To prevent corrosion, it is necessary to prime and enamel all carbon surfaces within three inches of an aluminum surface.

4.3 JOINT DESIGN

The joint design is crucial as it determines the degree of access to the wing extension. Mechanical fasteners rather than bonded joints would make the extension simpler to replace or inspect. Narrow strips of Kevlar could be chemically bonded to the composite extension and then mechanically fastened to the aluminum structure of the wing. Titanium fasteners would be required to prevent corrosion. The buildup of Kevlar layers should be enough to have stiffness matching at the joint. In addition, the layers should have a gradual buildup to prevent eccentricities in the joint. Figure 4-1 illustrates these recommendations. Further definition of joint particulars need only be considered in a detailed design.

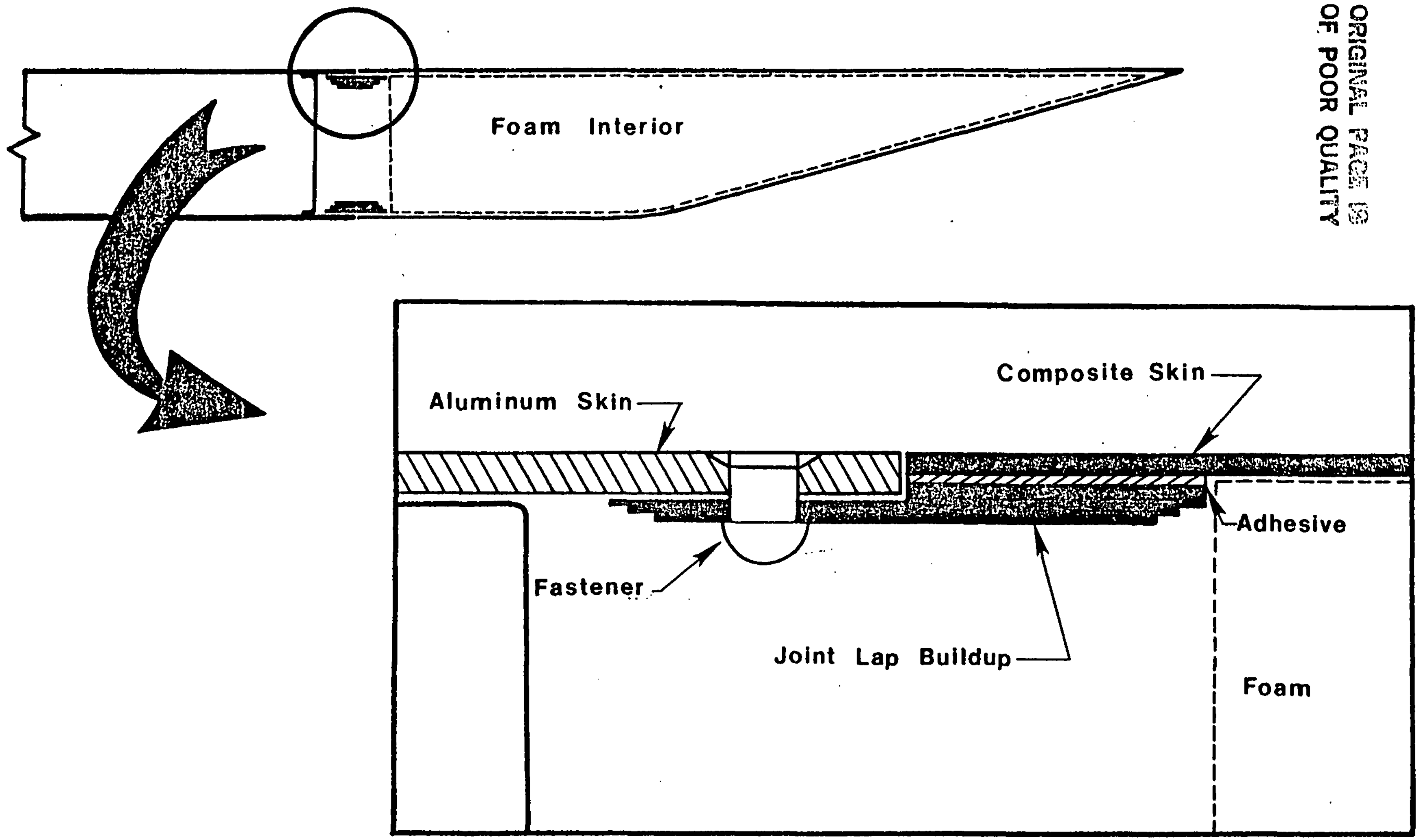


Figure 4-1. Recommendation for Joint Design

5.0 CONCLUSIONS

Prelinimary weight estimates showed that a composite wing extension would weigh at least half as much as an aluminum extension. Table 5-1 gives a final weight breakdown for the composite wing extension. The items include the weight of the hybrid laminate skin, the foam core, aluminum mesh, and the Kevlar buildup at the joint. The weight of the titanium fasteners and paint for the extension are unknown.

Although the composite wing extension provides a greater benefit in weight, additional benefits from the use of composite materials exist, but are more subtle. A composite wing extension would have a smooth surface increasing aerodynamic gains. In addition, there is little material wasted when composite components are fabricated as opposed to the manufacture of aluminum aircraft components.

The final and most important benefit would arise from being able to observe the performance of a composite aircraft component on a small scale and perhaps judge better the adequacy of components for general aviation aircraft.

Table 5-1

Weight Breakdown of the Composite

Wing Extension

Laminate Skin [45° (Kevlar), $0°_2$ (Carbon), 45° (Kevlar)]	.773 N (1.74 lbs)
Foam Core	11.12 N(2.5 lbs)
Aluminum Mesh (for lightning protection)	6.58 N (1.48 lbs)
Kevlar buildup at joint (t = .1035)	1.67 N (3.77 lbs)
Titanium Fasteners	unknown
Surface Paint	unknown
Approximate Total Weight	20.14 N (6.10 lbs)

6.0 REFERENCES

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