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POTENTIAL STRUCTURAL MATERIALS AND DESIGN CONCEPTS FOR LIGHT AIRCRAFT

Prepared by
SAN DIEGO AIRCRAFT ENGINEERING, INC.
San Diego, Calif.
for NASA Headquarters
Mission Analysis Division
Moffet Field, Calif.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • MARCH 1969

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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PREFACE

San Diego Aircraft Engineering, Inc., was responsible for conducting a NASA study of potential structural materials and design concepts for light aircraft, and to summarize the results of the study in a report which would be useful in guiding future structural designs of this class of aircraft.

These tasks were performed under contract NAS 2-4423 for NASA's Mission Analysis Division located at Ames Research Center, Moffett Field, California.

Ladislao Pazmany, Chief Design Engineer of San Diego Aircraft Engineering, managed the study program. He reported directly to Mr. G.D. McVicker, Chief Engineer and Executive Vice President of San Diego Aircraft Engineering, and to Mr. Frank Fink, President of the company. Assisting him were the following staff members:

Aerodynamics:	Larry Frohlich and Gary Johnson
Design & Weights:	Charles Waterman
Costs & Statistics:	Fred Tietge
Fatigue:	Fred Jones
Fasteners:	John O'Husky
Structures:	Hillyer Prentice

T.L. Galloway of NASA served as project monitor, coordinating the many objectives of this study in all its phases, as well as providing effective liaison between personnel of the Mission Analysis Division of NASA and San Diego Aircraft Engineering, Inc.

Acknowledgment is extended to the many people in the fields of education, government, and industry who gave freely of their time and supplied much valuable information.

Aircraft Owners and Pilots Association
Aluminum Company of America
American Aviation Corporation
Beech Aircraft Corporation
Bell Helicopter Company
Bellanca Aircraft Engineering Corp.
Boeing Aircraft Company
Bölkow GMBH
Brantley Helicopter Corporation
Cessna Aircraft Company
General Dynamics Corporation/Convair
Crescent Mold Engineering Corporation
Department of Transportation(FAA&CAB)
McDonnell-Douglas Aircraft Co.
E.I. DuPont de Nemours & Company
Experimental Aircraft Association
Fiberite Corporation
Flight Safety Foundation
Gibbs Flying Service
Goodyear Aerospace Corporation
Haveg Industries, Inc.
Heath Tecna Corporation
HITCO
Hughes Tool Company, Aircraft Division
Leach Industries
Lockheed California Company
M.C.W., Inc.
North American/Rockwell - Columbus Div.
Owens-Corning Fiberglas Corporation
Piper Aircraft Corporation
Pixie Mold and Tool Corporation
Ryan Aeronautical Company
Swedlow, Incorporated
Union Carbide Corporation
Whittaker Corporation, Narmco Research and Development Division

CONTENTS

PAGE

ILLUSTRATIONS

FIGURE		PAGE

TABLES

INTRODUCTION

The expansion and competitive position of general aviation in the field of transportation depends upon improving the safety and utility of light aircraft while, simultaneously, reducing their cost. Toward this end, the Mission Analysis Division of NASA is investigating various areas associated with the design of light aircraft and has sponsored this study on structural materials and concepts.

The primary objectives of this two-phase study, accomplished by San Diego Aircraft Engineering, Inc., was

(1) to make a comparative evaluation of a wide variety of materials and structural concepts, presently and potentially available for application to light aircraft, by investigating the affect of design, manufacturing, operational, and material requirements on the cost of this class of aircraft.

(2) to apply the more promising materials and structural concepts to the conceptual design of light aircraft.

(3) to identify key problem areas where additional research may increase the potential of promising materials or concepts.

A secondary objective was to prepare this report summarizing the results of the comparative evaluation and showing how these results may be applied to the structural design studies of light aircraft. This report is a sequel to the Final and Summary Reports which were prepared at the conclusion of the study.

Initially this report describes several pertinent cost considerations representative of this class of aircraft to establish a cost base for the study. The following section tabulates the properties of a variety of metallic and non-metallic materials that are promising candidates for application to future aircraft designs. And, the remaining sections, discuss in more detail the evaluation of these materials, their areas of application, fatigue consideration, and fastening techniques.

SYMBOLS, ABBREVIATIONS, AND CONVERSION FACTORS

A = Area, in.^{2}, ft^2
AR = Aspect ratio
a = Area of individual element
b = Width, in. or span, ft
C = Restraint coefficient
C_f = Fabrication cost/lb., $/lb.
C_{fb} = Baseline material fabrication cost/lb., $/lb.
C_{fn} = Candidate material fabrication cost/lb., $/lb.
C_i = Installation cost/lb., $/lb.
C_m = Material cost/lb., $/lb.
C_{mb} = Baseline material cost/lb., $/lb.
C_{mn} = Candidate material cost/lb., $/lb.
c_r = Root chord
c_t = Tip chord
C_w = Dollars worth of a pound of material saved
D = Diameter, in.
E = Modulus of elasticity in tension, psi
E_c = Modulus of elasticity in compression, psi
E_t = Tangent modulus, psi
e = Elongation in percent
F = Allowable stress or Fahrenheit
FAA = Federal Aviation Agency
FAR = Federal Air Regulations
F_b = Allowable bending stress, psi
F_c = Allowable compressive primary buckling stress, psi
F_{cc} = ultimate allowable crippling strength, psi
F_{cr} = Allowable compressive crippling stress, psi
F_{cu} = Ultimate allowable compressive stress, psi
F_{cy} = Yield allowable compressive stress, psi
F_{su} = Ultimate allowable shear stress, psi
F_{tu} = Ultimate allowable tensile stress, psi
F_{ty} = Yield allowable tensile stress, psi
f = Internal (calculated) stress, psi
f_{cc} = Ultimate crippling stress of element, psi
G.A.G. = Ground-air-ground, fatigue spectrum
HP = Horsepower
K = Factor
K_d = 33% markup factor for distributor/dealer
K_p = 10% profit factor for manufacturer
K_t = Theoretical stress-concentration factor
ksi = One thousand pounds per sq. in.
L = Length, in.
MAC = Mean aerodynamic chord
MIL-HDBK-5 = Military Handbook - Metallic Materials and Elements for Aerospace Vehicle Structures
N = Cycles to failure, fatigue
N_x = Compressive load per unit width, lb./in.
N_{xy} = Shear flow, lb./in.
n = Exponent, subscript
P = Applied load, lb. or power
P_f = Fabrication cost, $.
P_i = Installation cost, $.
P_{ib} = Baseline material installation cost, $.
P_{in} = Candidate material installation cost, $.
P_m = Material cost, $.
q' = Shear flow, lb./in.
R = Ratio of minimum to maximum stress, fatigue
S = Structural efficiency or wing area
$\$_{Savings}$ = Net overall savings realized, $.
S_b = Baseline material structural efficiency
S.L. = Sea level
S-N = Stress vs. cycles to failure, fatigue
S_n = Candidate material structural efficiency

t = Thickness, in. Also indicates tension when suscript

$\bar{t}$ = Cross-sectional area per unit width

t_c = Core thickness, in.

V_A = Design maneuvering speed (knots)

V_C = Design cruise speed (knots)

V_D = Design dive speed (knots)

VG = Positive and negative accelerations vs. air speed

V_{NE} = Design never-exceed speed (knots)

V-n = Refers to diagram plotting limit load factor vs. indicated airspeed

W = Weight, lb.

W_b = Baseline material weight, lb.

W_n = Candidate material weight, lb.

w = Density, lb./in.^{3}

α = Thermal coefficient of expansion, in./in./^oF.

Γ = Dihedral

ΔP = Difference in installation cost, \$.

$\Delta\$_{OC}$ = Difference in operating cost, \$.

$\Delta\$_{PP}$ = Change in purchase price of airplane

ΔW = Difference in weight, lb.

ε = Efficiency factor (materials)

Λ = Sweep

λ = Taper ratio

$\bar{\eta}$ = Plasticity reduction factor

τ_{cr} = Shear buckling stress, psi

CONVERSION FACTORS FOR INTERNATIONAL SYSTEM OF UNITS
(Ref. 39)

w	g/cm^3	= .03613 lb/in^3
P	kg	= 2.205 lb
$\frac{K_{eff}\ W}{L}$	kg/cm	= 5.602 lb/in
E_c	kg/cm^2	= 14.22 psi
F, f, $\frac{P}{L^2}$, $\frac{N_x}{L}$, $\frac{N_x}{b}$, $\frac{N_{xy}}{b}$	kg/cm^2	= 14.22 psi
	kg/cm^2	= 14.22(10^3) ksi
$\frac{WC}{L^3}$, $\frac{W}{bL^2}$, $\frac{W}{b^2L}$, $\frac{WK^{1/3}}{b^2a}$	kg/cm^3	= 36.13 lb/in^3
α	cm/cm/^{o}c	= .555 in/in/^{o}F
	km/hr	= 0.6214 mph
		= 0.5396 knots

COST CONSIDERATIONS

Evaluation of any material or structural concept is ultimately, if not initially, performed in terms of price or cost. This section discusses several parameters that are associated with or influenced by cost, i.e.:

Dollar value and price trends
Cost as a function of speed
Cost as a function of empty weight
Cost by component
Cost breakdown
Effect of labor savings (i.e.,mass production) on consumer price

Dollar Value and Price Trends

When comparing or evaluating anything in terms of dollars (or any currency) over a period of time, the effects of currency value fluctuation must always be considered. Otherwise, a change in price or cost due to some technical reason could be artificially magnified, diminished, or compensated by dollar value fluctuation -- thus camouflaging the particular cost or price effect being evaluated. This currency value fluctuation (usually inflation) is measured and described in terms of a consumer price index and is compared to any convenient point in time. The U.S. Government publishes a running tabulation of this index (based on price of representative goods, products, and services) in the STATISTICAL ABSTRACT OF THE UNITED STATES (ref. 1) which is published yearly. Price index values are plotted versus calendar year in Figure 1 for the period 1935 to 1985. The data from reference 1

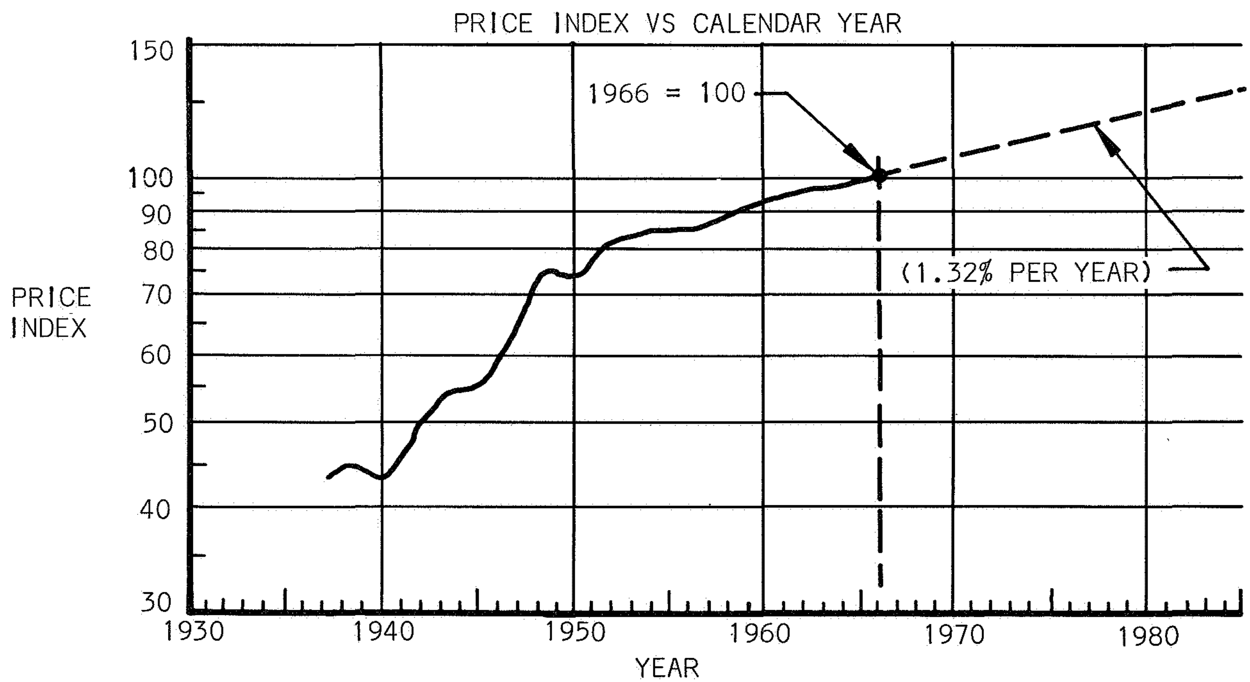


Figure 1

is based on 1958 equalling 100. The plot in Figure 1 is adjusted so that 1966 equals 100. So that any constant rate of inflation (i.e., a constant percentage increase per year) could be depicted as a straight line, the data is plotted on a semi-logarithmic graph. The constant inflation rate of 1.32% per year, apparent since about 1951, is extended to 1985. Therefore, in order to eliminate the effect of dollar value fluctuation, all dollars discussed hereafter will be 1966 dollars. Dollars of any particular year on the graph are converted to 1966 dollars by dividing the dollar value in question by the price index for that year.

The price trends of several typical General Aviation aircraft are illustrated in Figure 2. From the graph, three price categories are apparent. The low-price category includes those aircraft priced below $12,500.00 and are characterized by fixed landing gear, four-cylinder engines (180 hp max.) and a fixed pitch propeller. The middle-price category aircraft are priced approximately between $12,500 and $20,000 and are characterized by six-cylinder engines (up to 300 hp) and include some with retractable landing gear. The high-price category aircraft are priced above $20,000.00 and are characterized by six-cylinder engines (up to 400 HP), retractable landing gear, and constant speed propeller. The very high-price aircraft, i.e. twins, executive, and air taxi type, are not included since they are beyond the scope of the study.

GENERAL AVIATION AIRCRAFT CONSUMER PRICE TRENDS
(in 1966 dollars)

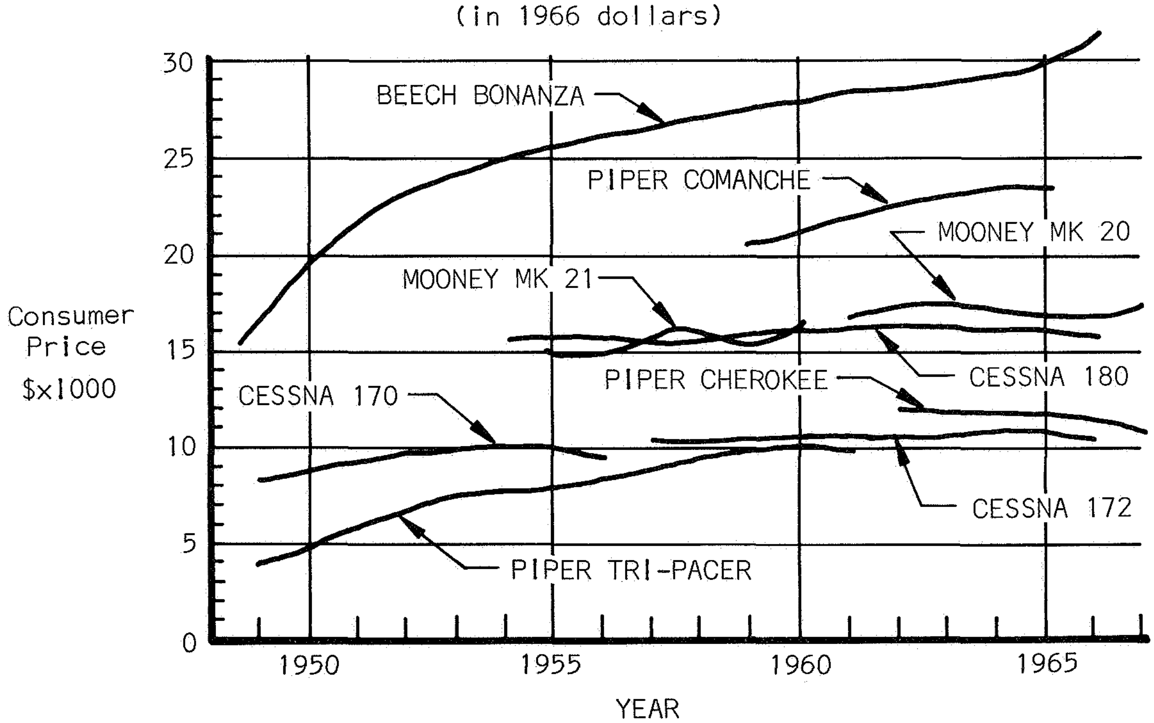


Figure 2

The following observations have been made from these price trends:

(1) Price of low-price aircraft in this decade is fairly constant to declining.

(2) Price of middle-price aircraft is fairly constant to rising.

(3) Price of high-price aircraft is generally rising.

The price per pound (empty) of aircraft is plotted in Figure 3 and illustrates, with only three exceptions, that not only is the price of airplanes rising, but consumers are paying a little more for each pound of aircraft.

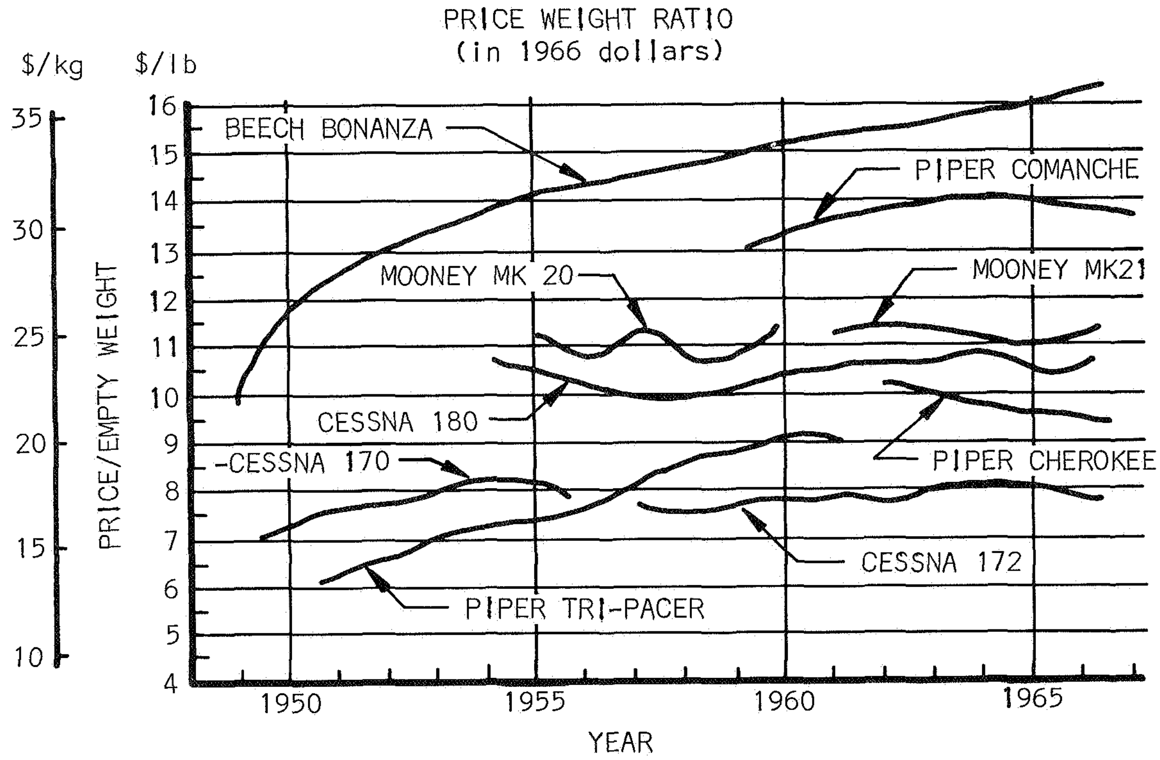


Figure 3

The increase of cost per pound is probably due to the 6% per year increase of U.S. aluminum and aircraft industry wages. No doubt, the following enhancements are contributory to the higher consumer prices:

Aerodynamic cleanness - More sophisticated instruments
Safety features - Comfort items - Luxurious interiors
Style changes - Engine refinements - Propeller advancements
Accommodations for accessories and non-standard equipment

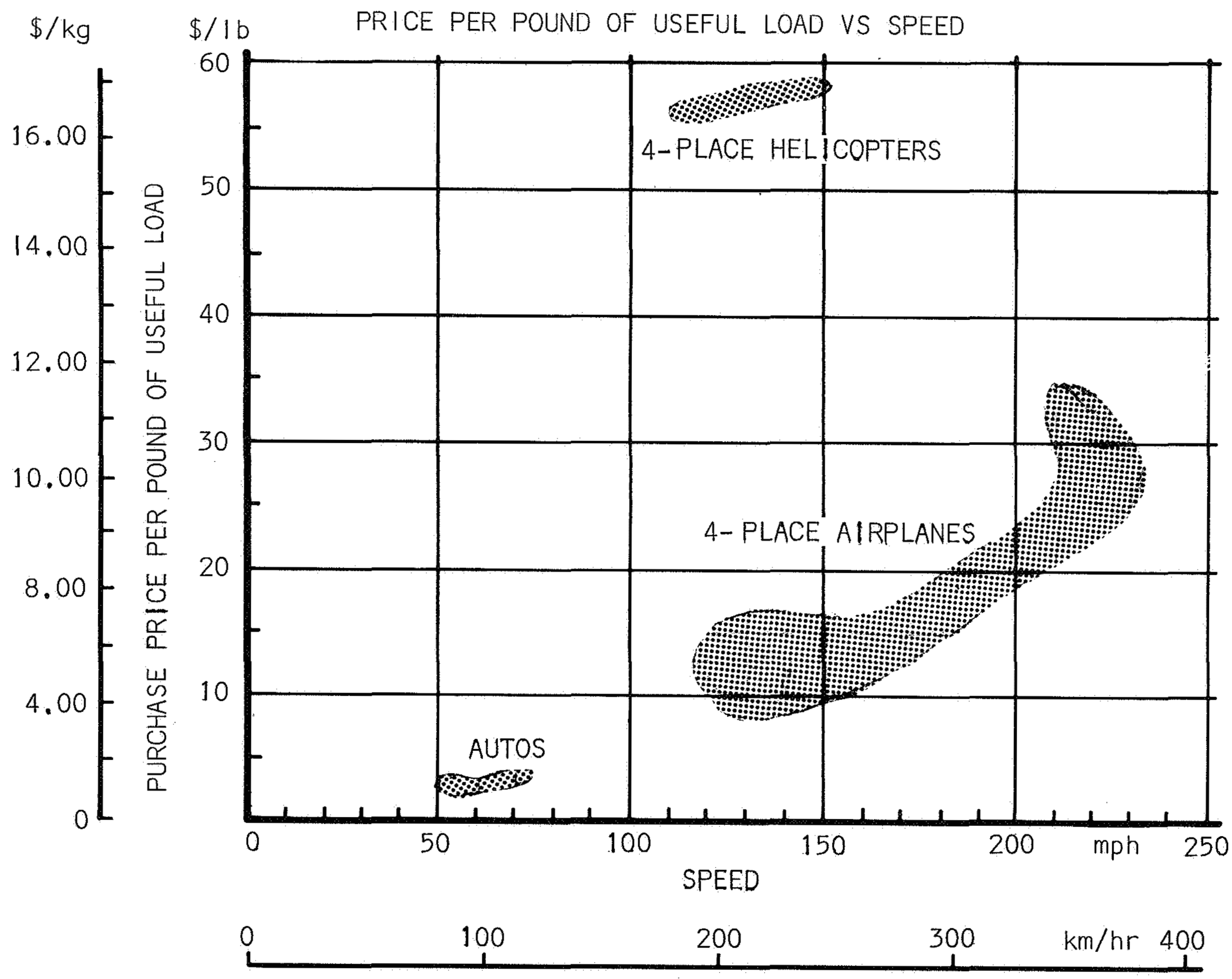
Cost as a Function of Speed and Empty Weight

As a comparative measure of the capital outlay required to transport a pound of payload (people) in four-place (minimum) vehicles at various speeds, Figure 4 shows that:

(1) It costs from \$2.50 to \$4.00 per pound to travel at 50 - 70 miles per hour in an automobile.

(2) It costs from \$8.50 to \$34.50 per pound to travel at 115 - 230 miles per hour in a General Aviation light, four-place airplane.

(3) It costs from \$56.00 to \$58.00 per pound to travel at 110 - 145 miles per hour in a four-place helicopter.



NOTE: Useful load includes all persons on board, fuel, oil and baggage.

Figure 4

Figure 5 illustrates the trend in helicopter prices per pound. The only conclusions that can be drawn are: (1) that reciprocating engine powered helicopters cost between $23.00 and $33.00 per pound empty; (2) that turbine powered helicopters cost between $60.00 and $75.00 per pound empty; and (3) that the cost per pound empty of helicopters is apparently <u>not</u> a function of empty weight.

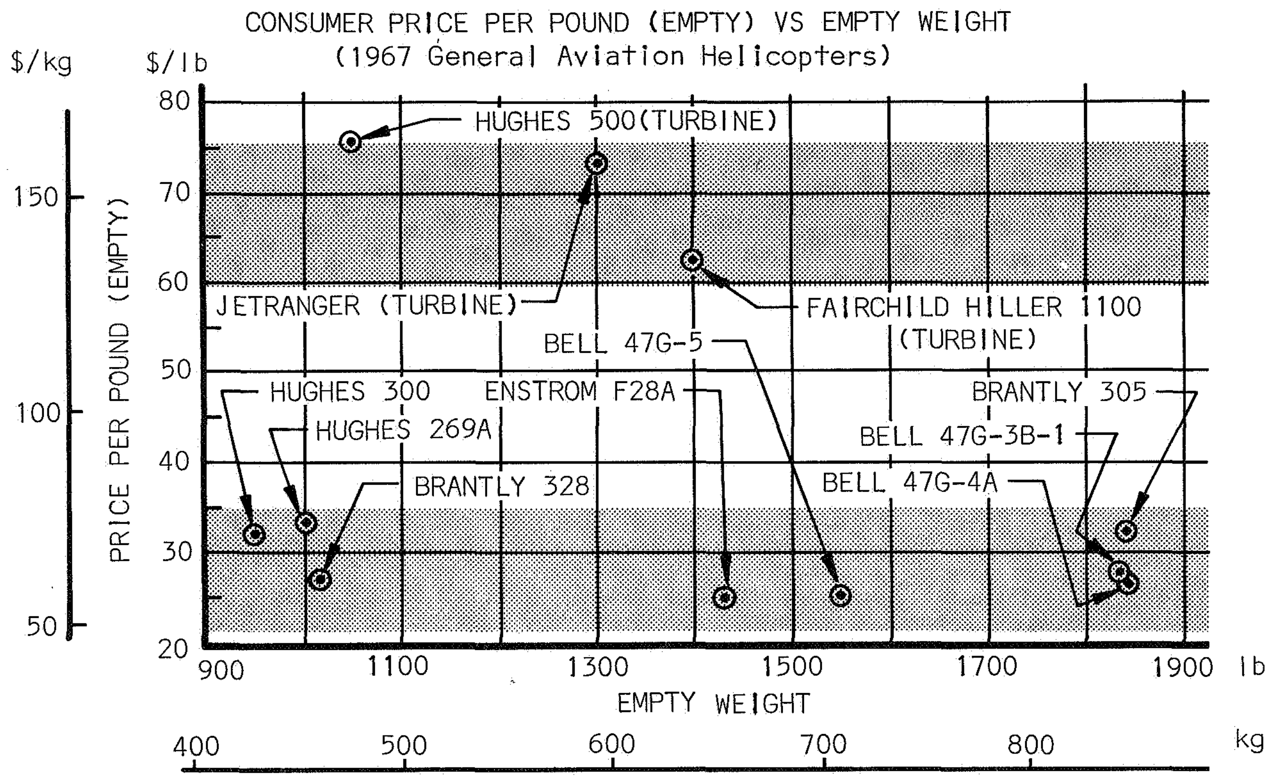


Figure 5

The cost per pound of empty weight for most of the light airplanes in U.S. production is plotted against empty weight in Figure 6; it varies from about $8.00/lb to about $27.00/lb.

The cost per pound of empty weight of most of the light airplanes in U.S. production is plotted against maximum speed in Figure 7. The cost varies from about $6.00/lb at 115 mph. to $27.00/lb at 300 mph.

The cost per pound of airframe for some representative light airplanes in U.S. production is plotted against empty weight in Figure 8.

The cost per pound of airframe for some representative light airplanes in U.S. production is plotted against maximum speed in Figure 9. It varies from $3.90/lb to $9.25/lb.

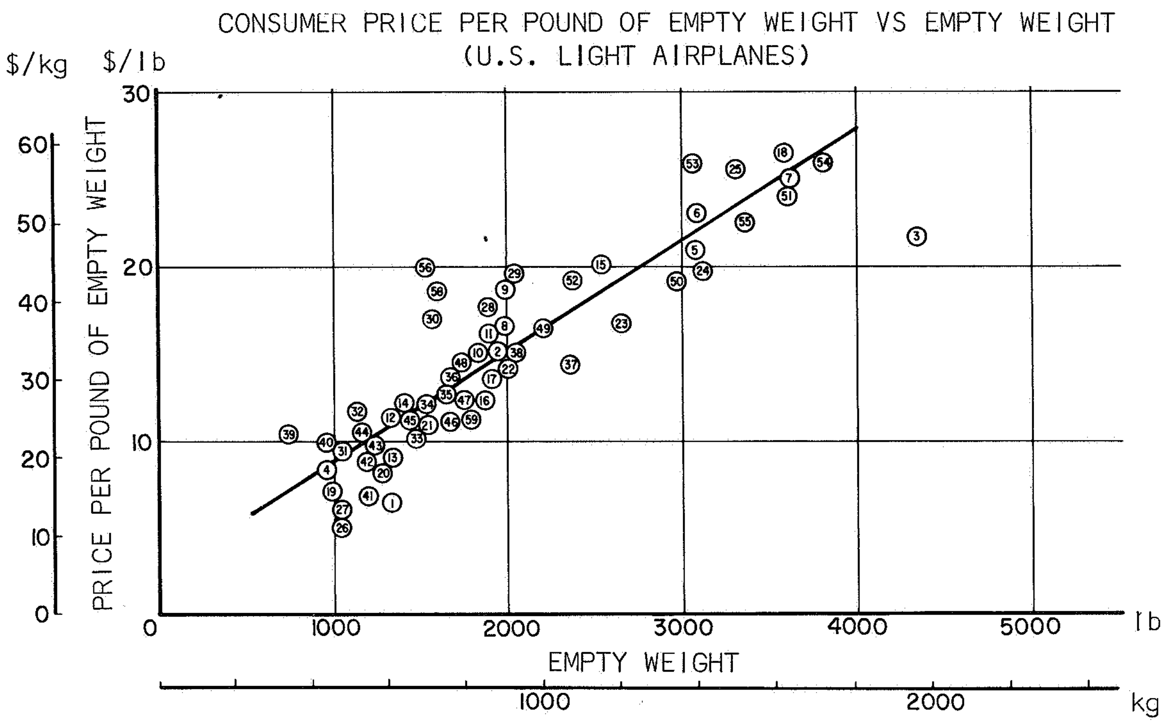


Figure 6

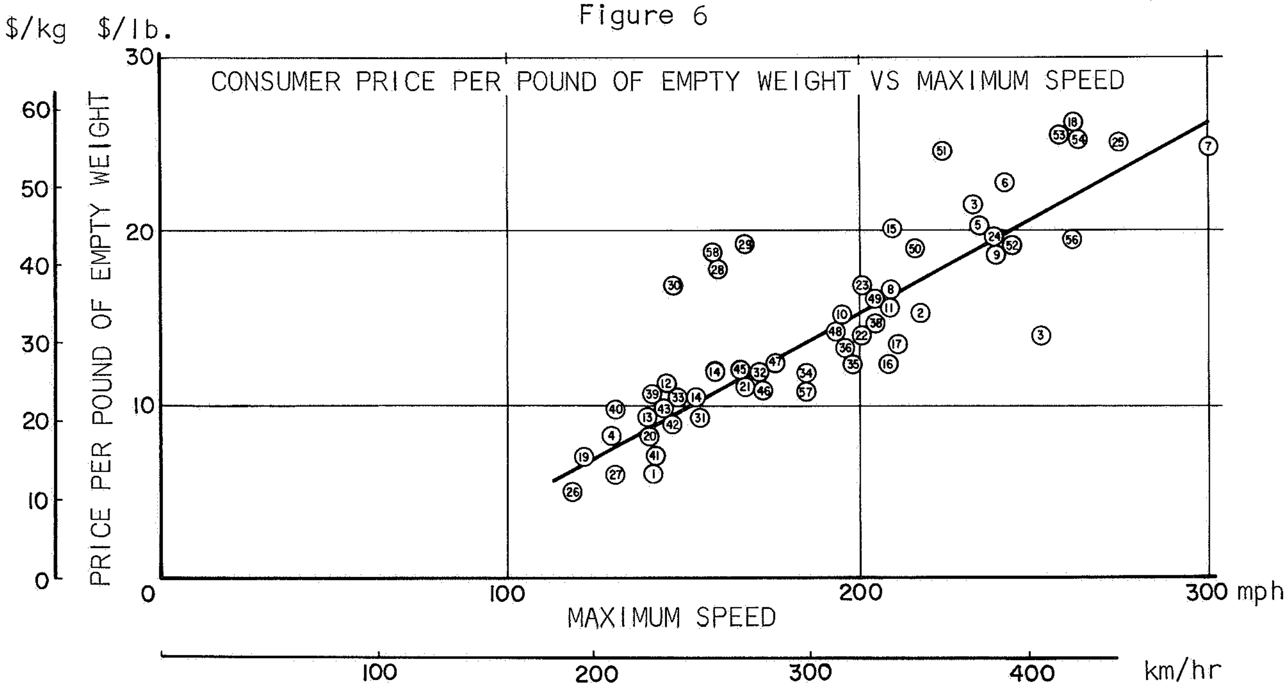


Figure 7

1. AERO COMMANDER 100
2. AERO COMMANDER 200
3. AERO COMMANDER 500 U
4. ALON A2 AIRCOUPE
5. BEECH BARON B55
6. BEECH BARON C55
7. BEECH BARON 56TC
8. BEECH BONANZA V35
9. BEECH BONANZA V35TC
10. BEECH DEBONAIR C33
11. BEECH DEBONAIR C33A
12. BEECH MUSKETEER CUSTOM III
13. BEECH MUSKETEER SPORT III
14. BEECH MUSKETEER SUPER III
15. BEECH TRAVEL AIR D95A
16. BELLANCA 260C
17. BELLANCA VIKING 300
18. CESSNA 401
19. CESSNA 150
20. CESSNA 172
21. CESSNA 182
22. CESSNA 210
23. CESSNA SUPER SKYMASTER
24. CESSNA 310L
25. CESSNA SKYNIGHT
26. CHAMPION 7ECA
27. CHAMPION 7GCA-A
28. HELIO COURIER
29. HELIO SUPER COURIER
30. LAKE LA-4
31. MAULE M-4JETASEN
32. MAULE M-4ROCKET
33. MOONEY MASTER
34. MOONEY MARK 21
35. MOONEY SUPER 21
36. MOONEY EXECUTIVE 21
37. MOONEY MUSTANG
38. NAVION MODEL H
39. BOLKOW JUNIOR
40. PIPER PA 18
41. PIPER CHEROKEE 140
42. PIPER CHEROKEE 150
43. PIPER CHEROKEE 160
44. PIPER CHEROKEE C180
45. PIPER CHEROKEE 235B
46. PIPER CHEROKEE SIX
47. PIPER CHEROKEE 300-SIX
48. PIPER COMANCHE B
49. PIPER TWIN COMANCHE B
50. PIPER AZTEC C
51. PIPER NAVAJO
52. TURBO TWIN COMANCHE B
53. PIPER TURBO AZTEC C
54. PIPER TURBO NAVAJO
55. RILEY TURBO ROCKET
56. WACO TS250
57. WACO S220
58. WREN 460

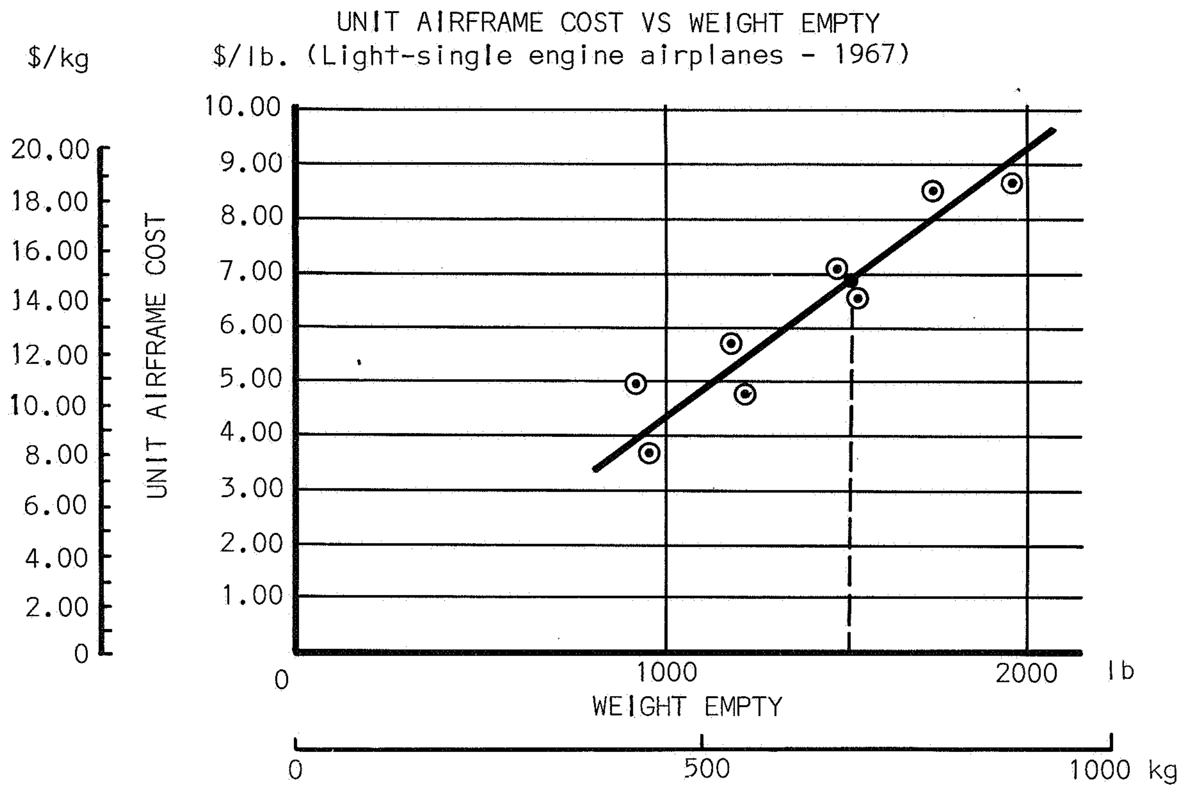


Figure 8

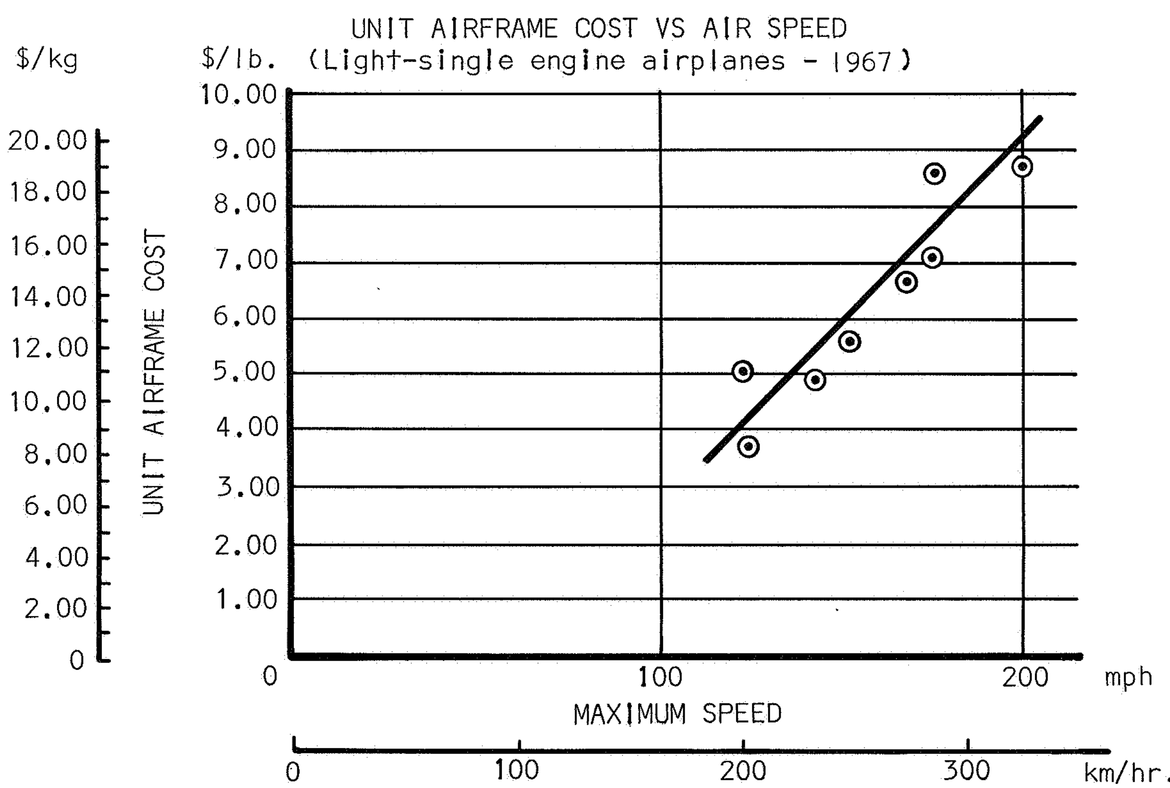


Figure 9

Cost by Component

Based on manufacturer's suggested retail prices and on catalog wholesale prices, the airframe (structure) cost of the various main components of a typical light airplane has been determined. Figure 10 illustrates the cost per pound of structure for: the wing, tail group, fuselage, and landing gear.

TYPICAL COST OF STRUCTURE
(in dollars per pound)

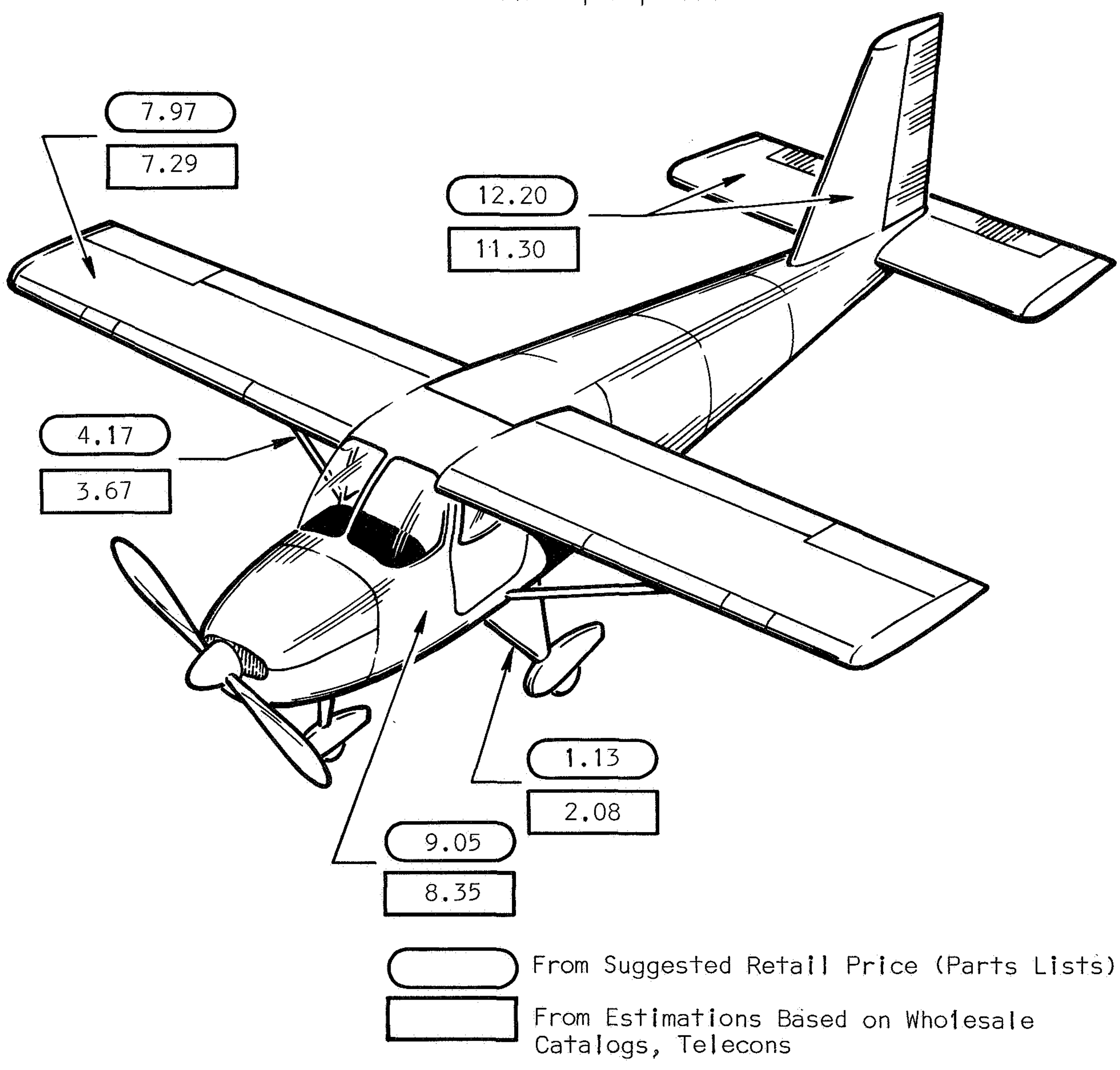


Figure 10

Cost Breakdown

The cost of a typical four-place airplane (approximately $17,000) is broken down both by dollar and by precentage of the total cost in Table I. The airframe fabrication cost represents approximately 36% of the consumer price. Dividing the airframe fabrication cost by its AMPR (*), or airframe weight, yields a unit airframe cost of $6.75 per pound.

TABLE I

COST BREAKDOWN OF A TYPICAL LIGHT AIRPLANE

Item	Dollars	Percent Total
Direct Labor - 630 hours (@ $2.70/hr)	$ 1,700.00	10.0
Overhead (130% of $1,700.00)	2,210.00	13.0
Material - Airframe	765.00	4.5
Equipment ($2420 Engine; $375 Prop.; $1305 Other)	4,100.00	24.2
Sub-Total	$ 8,775.00	51.7
Direct, Sales, and General Administrative Expenses (32% of $8,775.00)	2,810.00	16.5
Sub-Total (Manufacturing Cost)	$ 11,585.00	68.2
Factory Profit (10% of $11,585.00)	1,159.00	6.8
Total Dealer's Cost	$ 12,744.00	75.0
Distributor and Dealer Mark-up (33% of $12,744.00)	4,256.00	25.0
Total Cost to Customer	$ 17,000.00	100.0

AIRFRAME FABRICATION COST ANALYSIS

Airframe Labor (80% of Direct Labor)	$ 1,360.00
Airframe share of Overhead (80% of $2,210 + $2,810)	4,015.00
Raw Materials	765.00
Airframe Fabrication Cost	$ 6,140.00

* AMPR Weight is assumed to be 910 pounds.

Unit Airframe Cost: $\frac{\$\ 6,140.00}{910\ \text{lbs}} = \$\ 6.75/\text{lb}$

* AMPR weight includes Empty Weight less the following items: wheels, brakes and tires, engine (incl. carb. air box), starter, propeller and spinner, instruments, navigation equipment, battery and generator, electronics, cabin heat and vent.

Figure 11 illustrates this same breakdown. It should be noted that, although airframe labor and raw material represent only 12.5% of the consumer price of typical four-place, single-engine airplanes, this has a much farther-reaching effect on the total price of the airplane; i.e., dealer's mark-up, manufacturer's mark-up, and overall burden (the sum of which represents 61.3% of total price) are all functions of airframe cost. These effects are described quantitatively in the next paragraphs.

TYPICAL CONSUMER PRICE PERCENTAGE BREAKDOWN
OF A FOUR-PLACE SINGLE ENGINE AIRPLANE

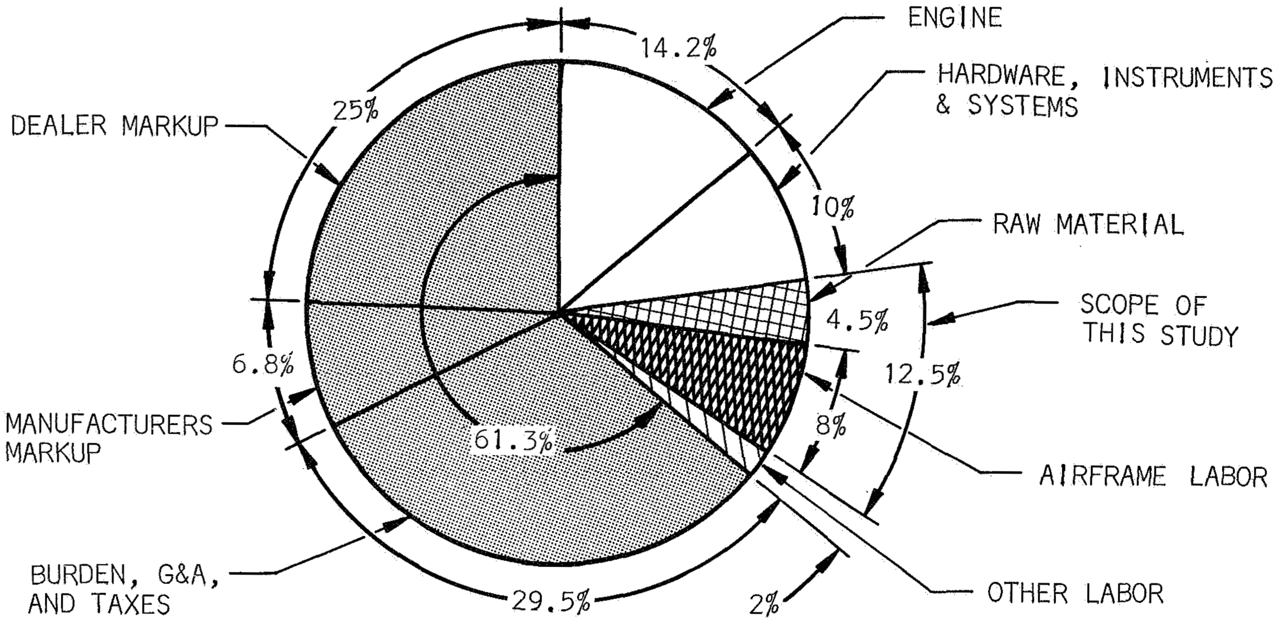


Figure 11

Effect of Labor Savings

As indicated previously, the cost of labor involved in manufacturing a light airplane (or any product for that matter) affects other portions of the total price. The change in consumer price, resulting from reductions in airframe fabrication labor, has been calculated and is illustrated in Figure 12. This plot was based on the following three assumptions:

(1) That manufacturer and dealer mark-ups would remain a constant percentage of consumer price (i.e., 6.8% + 25% = 31.8%).

(2) That raw materials and purchased hardware cost would remain constant regardless of labor savings.

(3) That overall burden (i.e., overhead, sales, and G&A expense) is 2.95 times labor.

NOTES: a. The 2.95 is derived from data in Table I, i.e.,

$$\frac{\$2,210 + \$2,810}{\$1,700} = 2.95$$

b. General formula used was:

$$CP_n = (L_n + 2.95\ L_n + M + E) + .318\ CP_n$$

Substituting:

$$CP_n = \frac{3.95\ L_n + 4865}{.682}$$

$$CP_n = 5.8\ L_n + 7140$$

Then converting to percentages:

$$\frac{CP_n}{CP_o} = \frac{5.8\ L_n}{CP_o} + \frac{7140}{CP_o}$$

$$\frac{CP_n}{CP_o} = \frac{5.8\ L_n}{10\ L_o} + \frac{7140}{CP_o}$$

Calling: $\frac{L_n}{L_o} = x$ and $\frac{CP_n}{CP_o} = y$

Then: $y = .58x + .42$

Where:

CP_n = Consumer Price - new

CP_o = Consumer Price - original = $10\ L_o$ = \$17,000

L_n = Labor - new

L_o = Labor - original

M = Materials = \$765

E = Equipment = \$4,100

Thus, as labor approaches zero, the resulting consumer price approaches a minimum of 42%. Obviously, the 100% savings in labor can only be approached through automation.

PRICE EFFECT OF LABOR SAVING

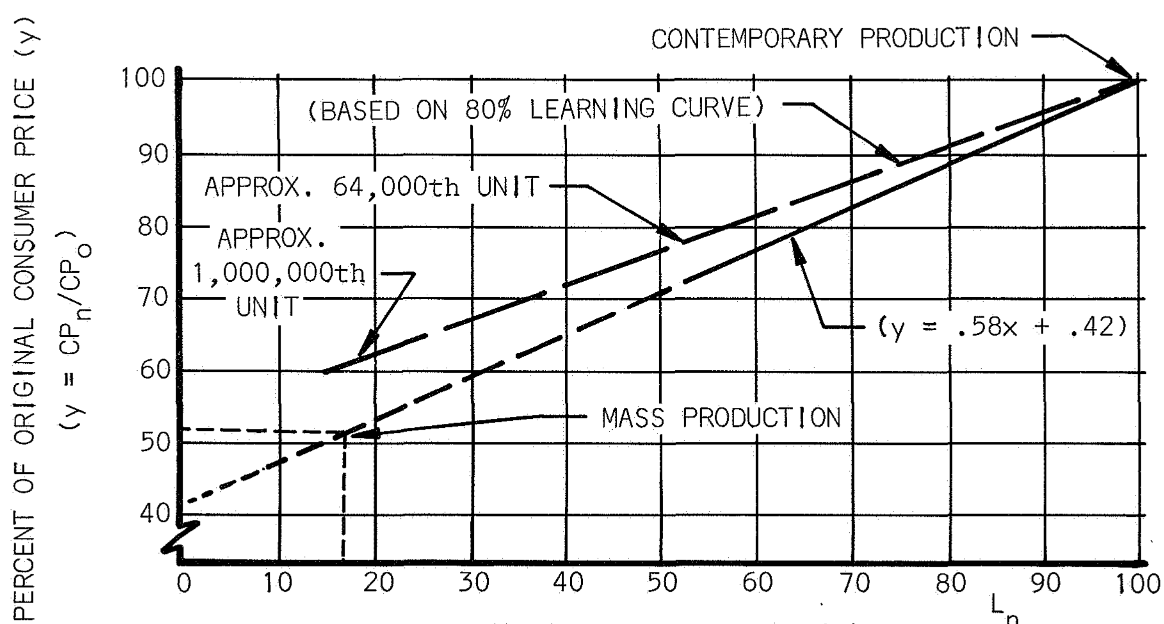


Figure 12

The following method of estimating potential price reductions resulting from very high labor savings (i.e., as the result of mass production), approximates the above estimate of 42%. The General Aviation single-engine, four-place light aircraft is really no more complicated or sophisticated than today's automobile. As an example, there is nothing on a light, General Aviation aircraft that is any more complicated than an automobile automatic transmission or a power brake unit. Some aircraft instruments are quite complicated and sophisticated, but mass production has proven itself in comparable sophisticated domestic products, such as remote control automatic tuning color television (e.g., consumer price of color television has been reduced by mass production from $1,500/$2,000 to less than $300.00).

Therefore, on the reasonable assumption that General Aviation light aircraft and automobiles are transportation vehicles of comparable complexity, the following dimensionless relationship has been generated to equate the two:

$$\frac{\$/\text{lb}_{\text{auto}}}{\$/\text{lb}_{\text{steel}}} \cong K \frac{\$/\text{lb}_{\text{aircraft}}}{\$/\text{lb}_{\text{aluminum}}}$$

This equation indicates that, for vehicles of comparable complexity, the ratio of raw material specific cost, (ingot), to finished product specific cost should be equal or similar for both vehicles, except as affected by production rate. This production rate or "mass production factor" is given as K in the equation.

The equation is solved for K with the following data:

$\$/\text{lb}_{\text{auto}}$ = $ .70 (from Reference 2)

$\$/\text{lb}_{\text{aircraft}}$ = $10.50 (from Figure 5)

$\$/\text{lb}_{\text{steel}}$ = $ .04 (from Reference 3)

$\$/\text{lb}_{\text{aluminum}}$ = $ .31 (from Reference 3)

solving: $$\frac{.70}{.04} \cong K \frac{10.50}{.31}$$

$$K = \frac{17.5}{33.9} = .52 \text{ or } 52\%$$

In other words, the cost per pound of a typical four-place, single-engine General Aviation aircraft could be expected to be reduced to 52% of today's cost if mass produced. This amounts to a practical consumer price reduction of 48%, which approximates the 58% limit price reduction determined in Figure 12. Obviously, the potential savings attainable through labor savings are well worth striving for. Consequently, relative fabrication costs should play a significant role when selecting candidate materials listed in the sub-section after next

POTENTIAL STRUCTURAL MATERIALS

This chapter concerns the investigation of a wide variety of structural materials applicable in the design of light aircraft (including helicopters) during the next 5 to 15 years. Materials available in five years are classified near-term. Those available fifteen years from now are considered far-term. High-priced near-term materials are also considered as far-term, anticipating cost reductions during the next 15 years.

The objective of this investigation was to determine from the initial compilation, a list of promising candidate materials based on parameters involving strength, stiffness, weight, and raw material cost.

Candidate materials will be further evaluated in subsequent chapters against such parameters as design-concept compatibility, method of joining, fatigue, formability, and costs relating to fabrication.

Materials were first selected from the broad spectrum of the various types available. In the beginning, an effort was made to pick representative examples from each type, basing the selection on one or more of the following characteristics:

(1) Accepted use in present-day aircraft construction
(2) Low density
(3) Low material cost

Not always an important factor because fabrication costs can be far more significant.

(4) High stiffness

Many areas of light aircraft and helicopter structures are designed for stiffness. This takes precedence on static strength requirements

(5) High strength

(6) Weldability, Brazability, Bondability

Inasmuch as present-day fabrication methods such as riveting contribute considerably to the overall cost of the finished product, a number of potential materials lending themselves to welding, brazing, and or bonding were included

(7) Minimum maintenance

(8) Materials exhibiting good corrosion resistance to atmospheric environments were considered.

Tables II and III tabulate the initial selection of materials, together with their pertinent properties.

In evaluating the initial selection of materials, structural efficiencies were determined for comparison purposes. These structural efficiencies are:

$$\text{Tension} = \frac{F_{tu}}{w} \qquad \text{Column} = \frac{\sqrt{E_c}}{w} \qquad \text{Shear Buckling} = \frac{\sqrt[3]{E_c}}{w}$$

Each structural efficiency was also divided by the material cost to obtain additional comparisons. In the case of far-term materials (to be used 15 years from now), the projected cost 15 years from now will be used. Comparative structural efficiencies are also presented in Tables II and III.

<u>Material Costs</u>

Materials costs, in dollars per pound, were determined by using price information obtained from the following companies:

Steel	- Ryerson & Sons, Los Angeles, California Republic Steel, Los Angeles, California
Aluminum	- Aluminum Company of America, San Diego, California
Magnesium	- The Dow Chemical Company, Los Angeles, California
Titanium	- Reactive Metals, Inc., Los Angeles, California
Beryllium	- Beryllium Metals & Chemicals Corp., New York, New York
Plastics (Reinforced)	- Whittaker Corp. (Narmco Division), San Diego, California Owens-Corning Fiberglas Corporation, New York, New York General Dynamics/Convair, San Diego, California Goodyear Aerospace Corporation, Akron, Ohio
Plastics (Unreinforced)	- Whittaker Corp. (Narmco Division), San Diego, California General Electric (Chemical Material Dept), Pittsfield, Mass. U.S. Rubber Company, Chicago, Illinois DuPont (Textile Fibers Dept), Wilmington, Delaware Borg-Warner (Marbon Chemical Div.), Washington, West Virginia Fibertite Corporation, Orange, California
Woods	- Niedermeyer-Martin Company, Portland, Oregon Gordon Plywood Company, Alhambra, California
Core Materials	- Hexcel Products, Inc., Los Angeles, California

<u>Promising Candidate Materials</u>

The selection of promising candidate materials was based primarily on an evaluation of the comparative structural efficiencies listed in Tables II and III for all initially selected materials. Additional considerations, such as ability to absorb energy, formability, fatigue, stress corrosion and atmos-

TABLE II

INITIAL SELECTION OF METALLIC MATERIALS AND COMPARATIVE STRUCTURAL EFFICIENCIES

Material	Availability	F_{tu}	F_{ty}	F_{cy}	E_c	w	Material Cost	Characteristics	$\frac{F_{tu}}{w}$	$\frac{F_{tu}}{w\ \$/lb}$	$\frac{\sqrt{E_c}}{w}$	$\frac{\sqrt{E_c}}{w\ \$/lb}$	$\frac{\sqrt[3]{E_c}}{w}$	$\frac{\sqrt[3]{E_c}}{w\ \$/lb}$	Ref.
	⑤	KSI	KSI	KSI	$\frac{PSI}{10^6}$	$\frac{LB}{in^3}$	$\frac{\$}{LB}$ ⑥		$\times10^{-3}$	$\times10^{-3}$	$\times10^{-3}$	$\times10^{-3}$	$\times10^{-2}$	$\times10^{-2}$	
									Tension		Column		Shear Buckling		
Alloy Steels															
1025 Tube	N	55	36	36	29	.284	0.50 ③	Low Cost, Weldable	194	388	19	38	-	-	4
4130 Norm.Tube	N	95	75	75	29	.283	0.92 ③	High Strength, Weldable	336	365	19	21	-	-	4
4130 (180HT) Bar	N	180	163	179	29	.283	0.13 ①	High Strength, Weldable	635	4900	19	146	-	-	4
4340 (260HT) Bar	N	260	217	242	29	.283	0.16 ①	Ultra High Strength, Weldable	919	5750	19	119	-	-	4
25Ni Maraging	N	319	284	-	24	.296	2.25 ①	Ultra High Strength, Weldable	1078	480	17	8	-	-	5
Stainless Steel															
301 (Full Hard)	N	185	140	179	28	.286	0.75	Corrosion Resistant, Weldable	645	860	18	24	11	15	4
PH15-7Mo (RH950)	N	225	200	210	30	.277	1.28	Ultra High Strength, Corrosion Resistant	813	635	20	16	11	9	4
Aluminum Alloys															
Sheet															
2024-T3	N	64	42	45	10.7	.100	0.65	Common use, Good Strength/Wgt.	640	985	33	50	22	34	4
2024-T3 (CLAD)	N	60	45	37	10.2	.100	0.66	Low Cost, High Energy Absorb.	600	910	32	48	22	34	4
2219-T87	N	62	50	50	10.8	.102	0.86	Weldable	610	710	32	37	22	25	4
5086-H32	N	40	28	26	10.4	.096	0.53	Weldable, Low Cost	417	787	34	64	23	43	4
5456-H343 ④	N	53	41	39	10.4	.096	0.60	High Welding Efficiency	552	920	34	57	23	38	4
6061-T6	N	42	36	35	10.1	.098	0.54	Low Cost, Corr.Resist, Weldable Formable, High Energy Absorb.	428	794	32	60	22	41	4
7005-T6	N	47	38	39	10.5	.101	0.65	Weldable, Low Distortion	465	716	32	49	22	33	6
7075-T6	N	76	66	67	10.5	.101	0.71	High Strength/Weight	752	1060	32	45	22	31	4
7178-T6	N	83	73	73	10.5	.102	0.71	High Strength/Weight	814	1145	32	45	21	30	4
Extrusions															
2014-T6	N	60	53	55	10.7	.101	0.97	Low Cost, Heavy Extrusions	590	608	32	33	-	-	4
2024-T4	N	60	44	39	10.7	.100	1.12	Common use, Good Str./Weight Low Cost, High Energy Absorb.	600	535	33	29	-	-	4
6061-T6	N	38	35	34	10.1	.098	0.44	Low Cost, Corr.Resist, Weldable Formable, High Energy Absorb.	388	1710	32	73	-	-	4
7075-T6	N	81	73	74	10.5	.101	1.39	High Strength/Weight	802	577	32	23	-	-	4
7075-T73	N	66	58	58	10.6	.101	1.42	Stress Corrosion Resistant	655	462	32	23	-	-	7
7178-T6	N	88	79	79	10.5	.102	1.49	High Strength/Weight	863	579	32	21	-	-	4
6061-T6 (Tube)	N	42	35	34	10.1	.098	0.70	Low Cost, Corr.Resist. Weldable Formable, High Energy Absorb.	428	612	32	46	-	-	4
Forgings															
2014-T6	N	65	55	55	10.7	.101	-	Common use	643	-	32	-	-	-	4
6151-T6	N	44	37	39	10.3	.098	-	High Forgeability, Low Cost	450	-	33	-	-	-	4
Castings															
356-T6	N	25	16.5	16.5	10.3	.097	-	Low Cost, Common use	258	-	33	-	-	-	4
A356-T61	N	38	28	28	10.5	.097	-	Premium Type	392	-	33	-	-	-	4
359-T61	N	45	34	34	10.7	.097	-	High strength	463	-	34	-	-	-	4
Magnesium Alloys															
Sheet															
AZ31B-H24	N	39	29	24	6.5	.064	1.10	High Stiff/Wt. Weld. Low Dens.	610	555	40	36	29	27	4
LA 141-T7	P	19	14	15	6.1	.048	25 (5) ②	Low Density	396	80	52	10	38	8	4, 8
Mg Yttrium-T5	P	55	50	50	6.5	.067	(6) ②	Good Strength/Weight, Weldable	820	137	38	6	28	5	9
Extrusions															
AZ31B-F	N	35	22	12	6.5	.064	1.20	High Stiff/Wt. Weld. Low Dens.	547	455	40	33	-	-	4
ZK60A-T5	N	45	36	30	6.5	.066	3.06	Good Strength/Weight & Stiffness/Weight	682	223	39	13	-	-	4
Castings															
ZK61A-T6	N	34	23	-	6.5	.065	-	Good Strength/Weight	523	-	39	-	-	-	10
ZE63A-T6	N	38	24	-	6.5	.065	-	Good Strength/Weight, Weldable	585	-	39	-	-	-	10
AZ91C-T6	N	27	14	14	6.5	.065	-	Ductile, Sound Castings	416	-	39	-	-	-	4
Titanium Alloys															
Bars															
Ti-6Al-4V ⑧	N	160	150	-	16.4	.160	4.33	High Strength, Weldable Corrosion Resistant	1000	231	25	6	-	-	4
Ti-13V-11Cr-3Al	N	170	160	162	15.5	.174	5.73		977	170	23	4	-	-	
Ti-6Al-4V Sheet ⑧	N	157	143	152	16.4	.160	13.65		980	72	25	2	16	1	4
Beryllium Alloys															
Sheet															
Unalloyed (Hot Pressed)	P	40	27	27	42.5	.067	-		597	-	97	-	-	-	4
Powder Sheet	P	70	50	50 ②	42.5	.067	275 (70) ②	High Stiffness/Weight	1045	15	97	1	52	1	11
Lockalloy ⑦	P	44	31	28	28	.076	290 (70) ②	Excellent for Compression	580	8	70	1	40	1	11
Extrusions															
Unalloyed	P	93	45	45 ②	42.5	.067	-		1390	-	97	-	-	-	11
Lockalloy ⑦	P	56.5	44.5	40 ②	28	.076	-		743	-	70	-	-	-	11

NOTES: ① Bar
② Estimated
③ 3/4" Diameter x .065" Wall
④ t = .050" Minimum Thickness
⑤ N = Near Term; P = Potential
⑥ Costs: t = .032" for Sheet; t = .125" for Extrusion; ( ) = 1982 Estimate
⑦ 62% Be - 38% Al
⑧ Solution Heat Treated and Aged

TABLE III

INITIAL SELECTION OF NON-METALLIC MATERIALS AND COMPARATIVE STRUCTURAL EFFICIENCIES

MATERIAL	AVAIL-ABILITY	F_{tu}	F_{ty}	F_{cu}	E_c	w	MATERIAL COST	CHARACTERISTICS	$\frac{F_{tu}}{w}$	$\frac{F_{tu}}{w\ \$/LB}$	$\frac{\sqrt{E_c}}{w}$	$\frac{\sqrt{E_c}}{w\ \$/LB}$	$\frac{\sqrt[3]{E_c}}{w}$	$\frac{\sqrt[3]{E_c}}{w\ \$/LB}$	REF.
		KSI	KSI	KSI	$\frac{PSI}{10^6}$	$\frac{LB}{in^3}$	$ / LB		$\times 10^{-3}$	$\times 10^{-3}$	$\times 10^{-3}$	$\times 10^{-3}$	$\times 10^{-2}$	$\times 10^{-2}$	
	②						③			③		③		③	
GLASS REINFORCED PLASTICS															
Chopped Fiber															
E-Glass/Polyester	N	20	-	26	1.99	.070	0.63	Corrosion Resistant, Formable	286	454	20	32	18	29	12
E-Glass/Nylon 6/10	N	20	-	18	1.0	.048	1.64(0.65)	Low Density, Formable	418	261 (643)	21	13 (22)	21	13 (32)	13
1" S-Glass/Epoxy	N	45	-	62	7.8	.060 ①	4.00(2.00)	High Strength & Stiff/Weight	750	190 (380)	46	12 (24)	33	8 (16)	14
Continuous Fiber															
181 Cloth E-Glass/Epoxy	N	45	-	45	3.3	.070	2.00(1.00)	Corrosion Resistant, Formable, High Strength/Weight	643	321 (643)	26	13 (26)	21	11 (21)	15
143 Cloth E-Glass/Epoxy	N	85	-	60	5.1	.070	2.00(1.00)		1210	605(1210)	32	16 (32)	25	12 (25)	15
181 Cloth S-Glass/Epoxy	N	94	-	65	4.2	.070	4.00(2.00)		1340	335 (770)	29	7 (14)	23	6 (12)	16
143 Cloth S-Glass/Epoxy	N	139	-	76	5.9	.070	4.00(2.00)		1980	495 (990)	35	9 (18)	26	6 (13)	16
Diallyl Phthalate (Prepreg)	N	49 ①	-	-	2.6	.070 ①	3.15	Low Curing Temp., Formable	700	223	23	7	20	6	17,18
FILAMENT REINFORCED PLASTICS/EPOXY MATRIX															
Unidirectional															
Boron	P	140	-	175	33	.071	700(10.00)	High Strength/Weight, Low Density, Corrosion Resistant	1970	(197)	81	(8)	45	(5)	19,30
Graphite	P	95.9	-	56.5	15.4	.051	600 (1.00)		1870	(1870)	77	(77)	49	(49)	19,30
E-Glass	P	150	-	85	6.9	.076	2.00(1.00)		1970	(1970)	35	(35)	25	(25)	19
S-Glass	P	210	-	120	7.6	.073	4.00(2.00)		2880	(1440)	38	(19)	27	(14)	19
Hollow Glass	P	80	-	80	4.5	.065	-		1230	-	33	-	25	-	19
Hi-Modulus Glass	P	210	-	120	9.2	.073	-		2880	-	42	-	29	-	19
Laminate (t=.016 in) ±45° Layers															
Boron	P	19.8	-	37.7	4.18	.071	700(10.00)	High Strength/Weight, Low Density, Corrosion Resistant	279	-	-	-	-	-	19,30
Graphite	P	5.8	-	31.6	2.10	.051	600 (1.00)		114	-	-	-	-	-	19,30
E-Glass	P	17.5	-	29.8	2.19	.076	2.00(1.00)		230	-	-	-	-	-	19
S-Glass	P	17.7	-	37.3	2.49	.073	4.00(2.00)		243	-	-	-	-	-	19
Hollow Glass	P	17.6	-	28.8	1.53	.065	-		271	-	-	-	-	-	19
Hi-Modulus Glass	P	17.7	-	37.3	2.98	.073	-		243	-	-	-	-	-	19
Laminate (t=.040 in) ±45°, 0°, 0°, 0° Layers															
Boron	P	91.9	-	120.1	21.9	.071	700(10.00)	High Strength/Weight, Low Density, Corrosion Resistant	1295	-	-	-	-	-	19,30
Graphite	P	59.5	-	46.5	10.2	.051	600 (1.00)		1175	-	-	-	-	-	19,30
E-Glass	P	97.0	-	62.9	5.0	.076	2.00(1.00)		1275	-	-	-	-	-	19
S-Glass	P	133.1	-	86.9	5.6	.073	4.00(2.00)		1825	-	-	-	-	-	19
Hollow Glass	P	55	-	59.5	3.3	.065	-		847	-	-	-	-	-	19
Hi-Modulus Glass	P	133.1	-	86.9	6.8	.073	-		1825	-	-	-	-	-	19
UNREINFORCED THERMOPLASTICS															
ABS (Sheet)	N	3.8	-	5.0	.190	.040	0.90	Low Density, Formable	95	105	11	12	14	16	20
ABS (High Strength)	N	7.3	-	10.4	.180	.039	0.46 ⑥		187	407	11	24	14	31	21
Polycarbonate	N	9.5	8.5	-	.345	.043	1.90		221	116	14	7	16	9	22
Nylon Yarn	N	22	-	-	.640	.049	5.10		450	88	16	3	18	3	23
Whittaker PBI-8	N	20	-	30	.700	.043	5.00		465	93	20	4	21	4	-
WOOD															
Hardwoods ④				F_{cy}											
White Ash	N	13.2	7.2	4.3	1.4	.022	5.80	Low Density, Presently used in some light aircraft	600	104	54	9	-	-	24
Yellow Birch	N	15.1	7.6	4.6	1.85	.025	6.60		603	92	54	8	-	-	24
Softwoods ④															
White Cedar	N	10.2	6.7	4.1	1.4	.016	2.10		638	303	74	35	-	-	24
Douglas Fir	N	10.9	5.9	4.2	1.5	.018	0.52		606	1170	68	131	-	-	24
Sitka Spruce	N	9.4	5.3	3.5	1.4	.015	0.67		626	935	79	118	-	-	24
Plywoods, 3-ply (.070 in thick) parallel to face grain															
Birch-Birch	N	8.6	-	2.7	1.2	.028	2.06		307	149	39	19	38	18	24
Poplar-Poplar	N	4.6	-	1.6	.8	.020	2.12		230	109	45	21	46	22	24
Mahogany-Poplar	N	6.7	-	2.6	.9	.020	2.05		340	166	48	23	48	23	24
Modified Woods, Staypak (parallel laminated)															
Birch, t=0.46	P	44.1	18.9	8.0	4.4	.049	-	Good Strength/Weight, Stabilized Wood	900	-	43	-	-	-	24
Spruce, t=0.32	P	35.8	25.9	4.3	4.7	.047	-		760	-	46	-	-	-	24

CORE MATERIALS	AVAIL-ABILITY	F_{su}(min)	F_{cu}(min)	w	MATERIAL COST	CHARACTERISTICS	REF.
	②	PSI	PSI	LB/FT³	$/LB		
Resin Coated Nylon 3/8 cell	N	45	140	2.0	22.90	Light Weight, Fireproof	25
3003 Aluminum 1/4 cell	N	44	92	2.3	4.17	Inexpensive, presently used in aircraft	25
5052 Aluminum 1/4 cell	N	52	112	2.3	4.84		25
2024 Aluminum 1/4 cell	N	138	300	2.8	11.62	High Strength/Weight	25
Nylon Phenolic 3/8 cell	N	56	160	2.5	14.10	Good Strength/Weight	25

NOTES: ① ESTIMATED ② N = NEAR TERM, P = POTENTIAL ③ ( ) = 1982 ESTIMATE ④ PARALLEL TO GRAIN ⑥ RESIN

⑤ MIL HDBK 17 material properties were used in this table if available. Otherwise, manufacturers published data were used.

pheric corrosion, low-quench sensitivity, loading intensity, and accepted usage in present-day aircraft, also influenced the choosing of candidates. Metallic material condidates are listed in Table IV, together with their structural efficiencies. Non-metallic material candidates are presented in Table V in a similar manner. Figures 13, 14, and 15 list the comparative structural efficiency of materials by decreasing order of magnitude.

Metallic Materials (Ref. Table IV)

TUBING - Two steels and one aluminum alloy were selected as tubing candidates. While the 6061-T6 aluminum alloy is superior from the standpoint of structural efficiencies, 1025 steel is still being used today in areas where low cost and ease of welding so dictate. The 4130 normalized steel tubing is used where column loading intensities are moderate-to-high and size limitations are present. The most likely areas of application for tubing are fuselage weldments and engine mounts.

BAR MATERIAL - Candidates are listed with the intent of showing materials of high strength for use in areas of landing-gear assemblies, rotor mechanisms, and primary structural fittings having space limitations. Although there are many types of high-strength materials available, the selection represents the lower and upper end of the chrome-alloy series (4130 and 4340), and also includes one of the newer types of maraging steels, 25 Ni. This steel, although 1.8 times as strong as 4130 (180 H.T.), is also seventeen times as costly ($2.25/lb vs. $0.13/lb). It is a high-quality steel with superior corrosion resistance and toughness over the commonly-used chrome-alloy series.

FORGINGS are occasionally used in helicopters and light aircraft. When used, 2014-T6 is the primary forging alloy, especially for miscellaneous low-stressed fittings where economy and increased corrosion performance predominate.

SHEET - A number of sheet materials are available for use in the construction of light aircraft and helicopters. Sheet stock is used mainly as a covering for the airframe. It is also bent and formed into frames, ribs, stringers, stiffeners, and various types of brackets.

The 2024-T3 alloy, especially the clad version, is by far the most commonly-used skin covering on present-day light aircraft. In addition to having high structural efficiencies, it is a good corrosion-resistant candidate, exhibiting superior qualities of fatigue, energy absorption, and formability when compared to most of the other sheet materials.

The 5XXX series aluminum sheet material is included because of its low-cost structural efficiencies. It also has good formability.

Type 6061-T6 is next in importance to 2024-T3 clad as a material candidate. Its low cost, coupled with its high corrosion resistance and high stress corrosion resistance, formability, and energy absorption characteristics, makes it extremely attractive.

TABLE IV

PROMISING CANDIDATE MATERIALS - METALLIC

MATERIAL	AVAIL-ABILITY	F_{tu}	F_{ty}	F_{cy}	F_{su}	E_c	e	w	CORROSION RESISTANT	MATERIAL COST	WELD-ABILITY	THERMAL CO-EFF. $\alpha/10^5$	COMPARATIVE STRUCTURAL EFFICIENCIES $\frac{F_{tu}}{w}$	$\frac{F_{tu}}{w\ \$/LB}$	$\frac{\sqrt{E_c}}{w}$	$\frac{\sqrt{E_c}}{w\ \$/LB}$	$\frac{\sqrt[3]{E_c}}{w}$	$\frac{\sqrt[3]{E_c}}{w\ \$/LB}$	REF.
	(6)	KSI	KSI	KSI	KSI	$\frac{PSI}{10^6}$	%	$\frac{LB}{in^3}$		\$ / LB		in/in/°F							
TUBING																			
1025 Steel	N	55	36	36	35	29	8-13	.284	POOR	0.50 (1)	EXCEL	.70	194	388	19	38	-	-	4
4130(Normalized)	N	95	75	75	55	29	12	.283	FAIR	0.92 (1)	GOOD	.63	336	365	19	21	-	-	4
6061-T6	N	42	35	34	27	10.1	12	.098	EXCEL	0.70 (1)	GOOD	1.30	428	612	32	46	-	-	4
BAR (t=1.00 in)																			
4130 (180HT)	N	180	163	179	109	29	12	.283	FAIR	0.13	GOOD	.63	635	4900	19	146	-	-	4
4340 (260HT)	N	260	217	242	149	29	10	.283	FAIR	0.16	FAIR	.63	919	5750	19	119	-	-	4
25 Ni (Maraging)	N	319	284	-	-	24	8	.296	GOOD	2.25	FAIR	.59 (5)	1078	480	17	8	-	-	5
FORGING																			
6181-T6	N	44	37	39	28	10.3	10	.098	EXCEL	-	-	1.28	450	-	33	-	-	-	4
2014-T6	N	65	55	55	39	10.7	7	.101	POOR	-	-	1.25	643	-	32	-	-	-	4
SHEET (t=.032 in)																			
2024-T3	N	64	42	45	40	10.7	15	.100	POOR	0.65	GOOD	1.29	640	985	33	50	22	34	4
2024-T3 CLAD	N	60	45	37	38	10.2	15	.100	GOOD	0.65	GOOD	1.29	600	910	32	48	22	33	4
5086-H32	N	40	28	26	24	10.4	6	.096	GOOD	0.53	EXCEL	1.32	417	787	34	64	23	43	4
5456-H343 (4)	N	53	41	39	31	10.4	6	.096	GOOD	0.60	EXCEL	1.33	552	920	34	57	23	38	4
6061-T6	N	42	36	35	27	10.1	10	.098	EXCEL	0.54	GOOD (2)	1.30	428	794	32	60	22	41	4
X7005-T6	N	47	38	39	-	10.5	-	.101	GOOD	0.65	GOOD	1.32	465	716	32	49	22	33	6
7075-T6	N	76	66	67	46	10.5	7	.101	POOR	0.71	GOOD	1.29	752	1060	32	45	22	31	4
7178-T6	N	83	73	73	50	10.5	7	.102	POOR	0.71	GOOD	1.30	814	1145	32	45	21	30	4
AZ 31B-H24	N	39	29	24	18	6.5	6	.064	POOR	1.10	GOOD	1.40	610	555	40	36	29	27	4
EXTRUSION(t≤.250)																			
2014-T6	N	60	53	55	35	10.7	7	.101	POOR	0.97	GOOD	1.25	590	608	32	33	-	-	4
2024-T4	N	60	44	39	32	10.7	12	.100	POOR	1.12	GOOD	1.29	600	535	33	29	-	-	4
6061-T6	N	38	35	34	24	10.1	10	.098	EXCEL	0.44	GOOD	1.30	388	1710	32	73	-	-	4
7075-T6	N	81	73	74	45	10.5	7	.101	POOR	1.39	GOOD (2)	1.29	802	577	32	23	-	-	4
7075-T73	N	66	58	58	-	10.6	-	.101	GOOD	1.42	GOOD	1.29	655	462	32	23	-	-	7
7178-T6	N	88	79	79	47	10.5	5	.102	POOR	1.49	GOOD	1.30	863	579	32	21	-	-	4
Mg Yttrium-T5	P	55	50	50	30	6.5	4	.067	POOR	(6.00) (3)	GOOD	1.40 (5)	820	(137)	38	(6)	-	-	9
CASTING																			
A356-T61	N	38	28	28	27	10.5	5	.097	GOOD	-	-	1.19	392	-	33	-	-	-	4
356-T6	N	25	16.5	16.5	25	10.3	3	.097	GOOD	-	-	1.19	258	-	33	-	-	-	4
359-T61	N	45	34	34	31	10.7	4	.097	GOOD	-	-	1.16	463	-	34	-	-	-	4
ZK 61A-T6	N	34	23	-	-	6.5	2	.065	FAIR	-	-	1.40 (5)	523	-	39	-	-	-	10
ZE 63A-T6	N	38	24	-	-	6.5	4	.065	FAIR	-	-	1.40 (5)	585	-	39	-	-	-	10
AZ 91C-T6	N	27	14	14	-	6.5	2	.065	FAIR	-	-	1.40	416	-	39	-	-	-	4

(1) 3/4 x .065 WALL (2) RESISTANCE WELDABILITY (3) ( ) = 1982 ESTIMATE (4) t = .051 (5) ESTIMATED
(6) N = NEAR TERM, P = POTENTIAL

Type X7005 aluminum alloy is one of the more recently developed materials. It can be easily brazed, soldered, or welded and still maintain its high properties without requiring solution heat treating afterwards. Its low-quench sensitivity, eliminating severe distortion during cooling after heat treatment, makes this alloy a material candidate.

Types 7075-T6 and 7178-T6 are included as they represent the highest strength aluminum alloys available today. While their corrosion and stress-corrosion resistance, formability, energy absorption, and quench sensitivity characteristics are inferior to some of the other aluminum alloys, they exhibit superior tensile structural efficiencies and will outperform other aluminum alloys when used in areas of high-load intensity.

AZ 31B-H24 magnesium alloy has superior column and shear buckling structural efficiencies and is, therefore, listed with the aluminum sheet material. Its higher cost and lower corrosion resistance make it a less likely candidate.

EXTRUSIONS are used mainly as flange material in beams and major bulkheads, stringer material in wide columns (fuselage semi-monocoque, wing-plate stringer), and stiffeners in high-loading intensity areas.

Type 2014-T6 is generally used for sections greater than 0.125-inch thick where its low cost, together with its high-yield strength, makes it a desirable candidate.

Type 2024-T4 extrusions are commonly found in light aircraft for sections under 0.125-inch thick. This alloy, in addition to having good structural efficiencies, exhibits superior fatigue and energy-absorption qualities.

Type 6061-T6 shows considerable promise for extrusions requiring thin sections and high corrosion resistance. The low cost, high energy absorption, and stress-corrosion resistance of this alloy make it an excellent candidate.

The 7075 and 7178 extrusions have the highest mechanical properties of the aluminum alloys. While the T6 tempers are relatively low in stress-corrosion resistance and energy-absorption capabilities, the T73 temper of 7075 is excellent in both respects and warrants consideration in the final selection of candidate materials.

Mg Yttrium-T5 is a new high-strength magnesium alloy. Its high compression yield strength (improving the compressive tangent modulus), coupled with its low density, makes it the most efficient of all the metallic candidates when used in compression critical structures. However, the projected cost of $6.00 per pound fifteen years from now reduces its chances of becoming a prime candidate.

CASTINGS are used mainly for rotor mechanisms, wheel hubs, pulleys, brackets, bellcranks, and various fittings.

A356-T61 and 359-T61 are premium-quality composite mold castings. Although they are in general use today, anticipated high production rates for light aircraft/helicopters make these alloys less likely candidates than a permanent mold or die-cast material.

Type 356-T6 is a permanent mold casting alloy in general use today, and it appears it will remain a likely candidate in the future.

AZ 91C-T6, available as a permanent mold casting, is one of the most common magnesium castings in use today.

CORE MATERIAL (Ref. Table III) is used in honeycomb-sandwich constructions. Type 3003 1/4-inch cell, 2.3 pounds per cubic foot aluminum honeycomb core is considered to be the most promising candidate. It is of adequate strength for light aircraft construction and is only a fraction of the cost of the expensive reinforced plastic honeycomb.

COMPARATIVE SHEAR CRIPPLING EFFICIENCIES

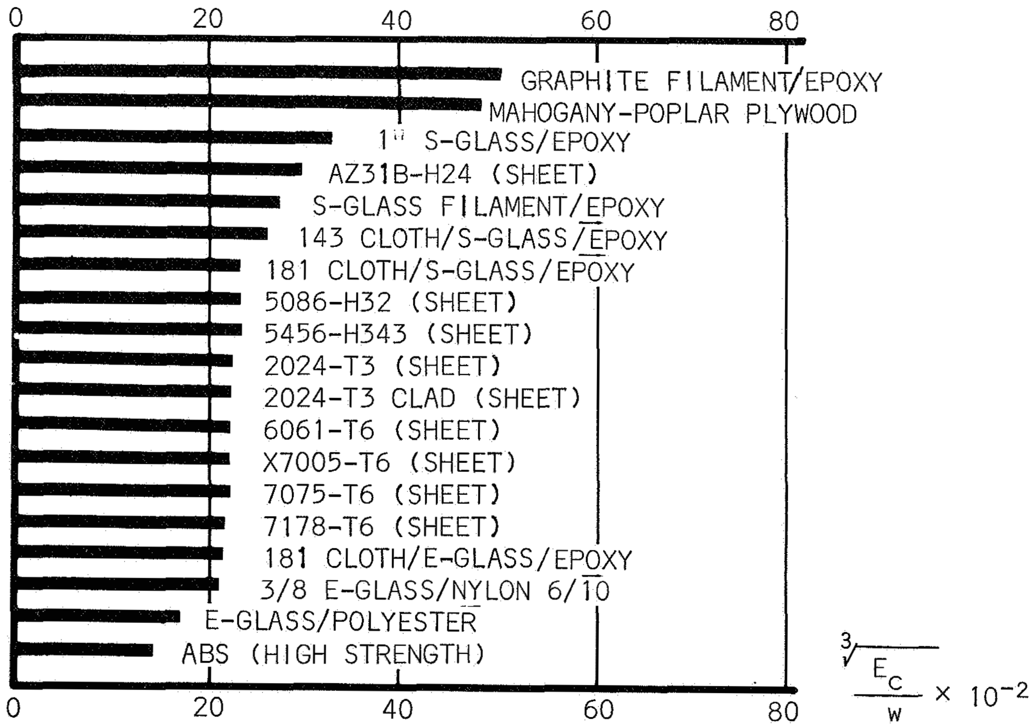


Figure 13

COMPARATIVE COLUMN EFFICIENCIES

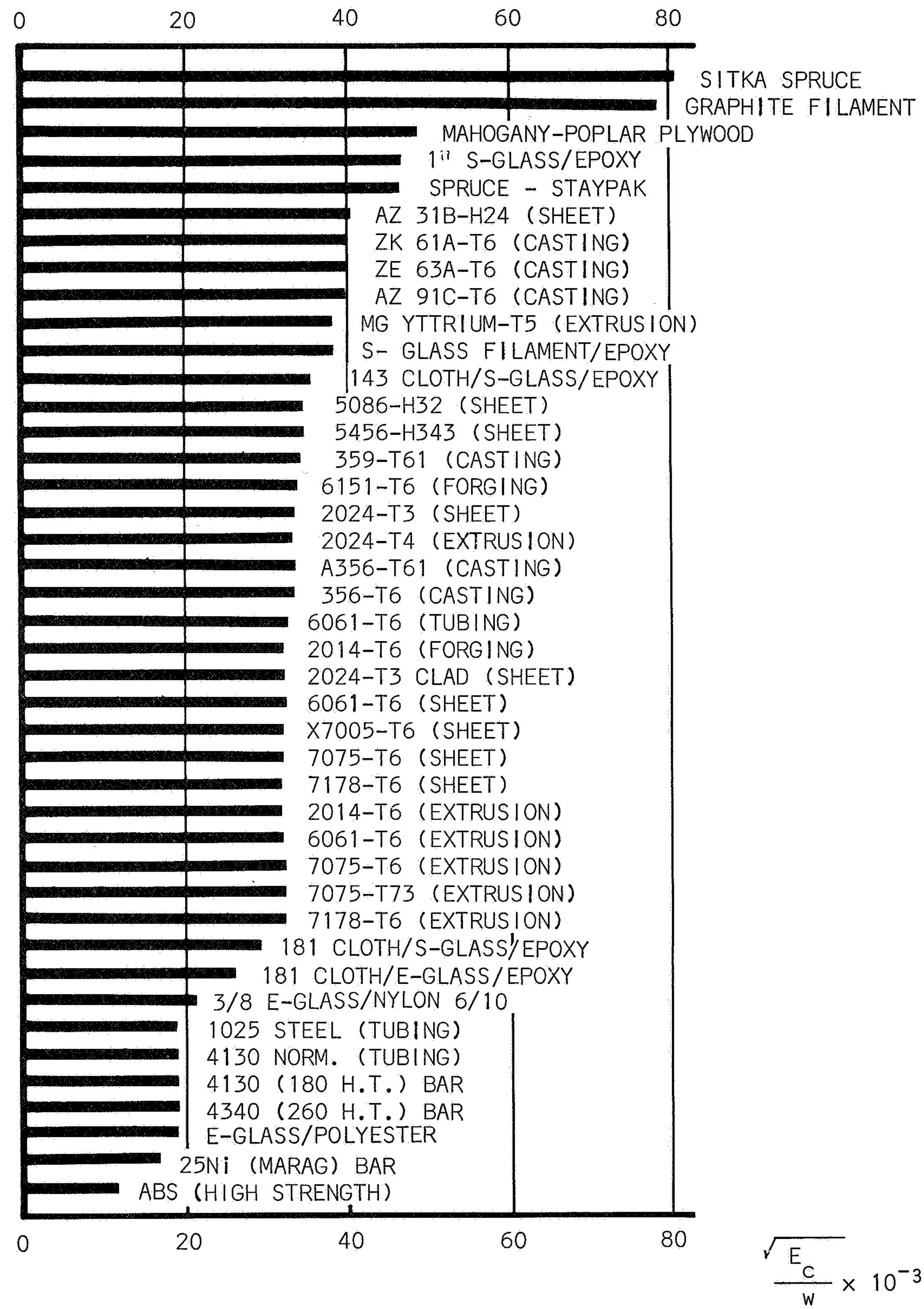


Figure 14

COMPARATIVE TENSION EFFICIENCIES

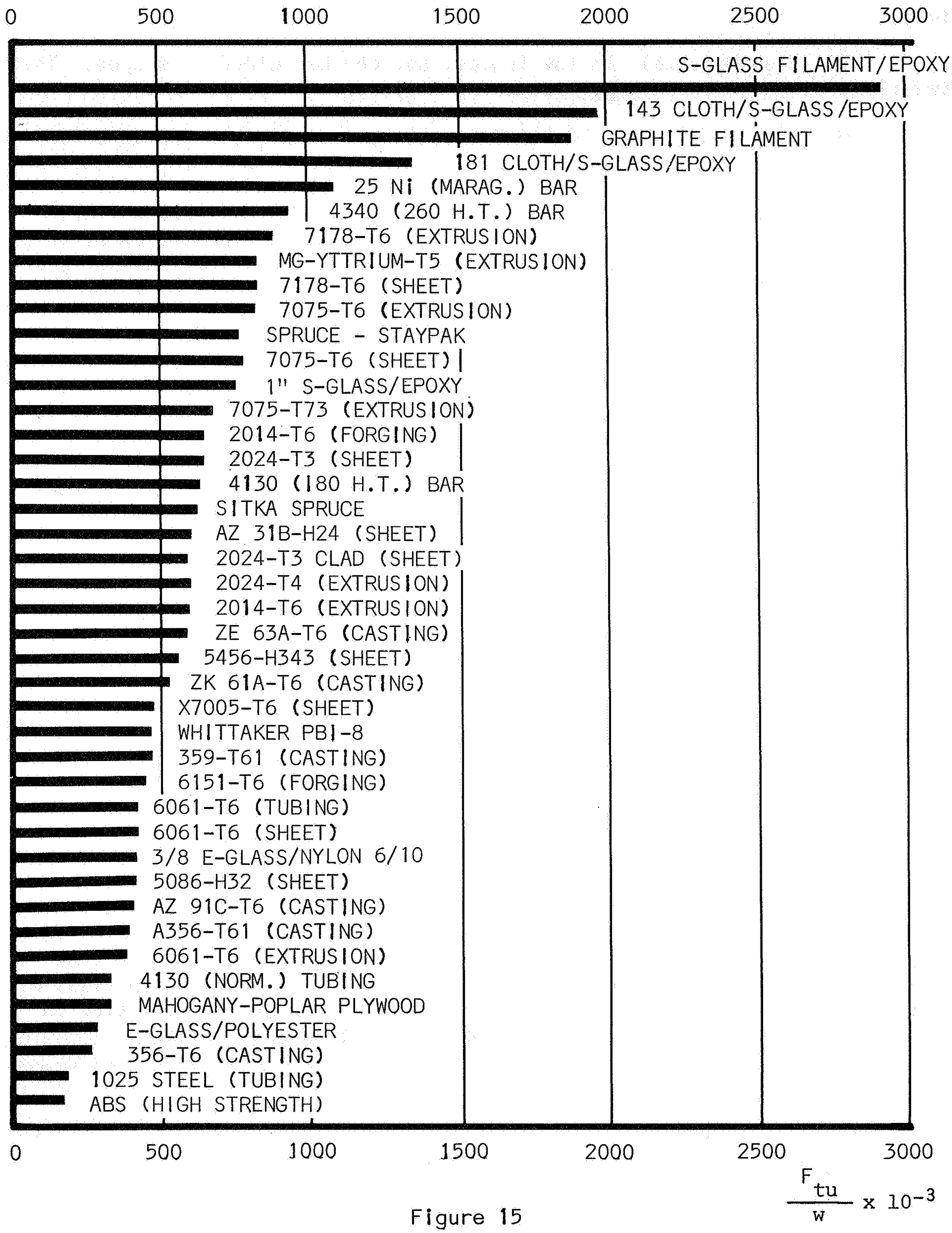


Figure 15

Non-Metallic Materials (Ref. Table V)

NON-REINFORCED THERMOPLASTICS are used for fairings and for low-stressed skin.

ABS (High Modulus) is low in cost and can be molded to shapes. This material, although not highly flammable, will support combustion.

CHOPPED FIBER-REINFORCED PLASTICS are best adapted for areas of low - loading intensity such as secondary fittings, fairings, and low-stressed skin.

3/8 E-Glass/Nylon 6/10, is a medium-cost injection moldable thermoplastic reinforced with 1/4-inch to 3/8-inch long glass fibers (30% by weight). It is finding use in the design of next-generation commercial transports in such areas as access covers for wing fuel tanks. Nylon 6/10 is a self-extinguishing material from the standpont of flammability.

E-Glass/Polyester is a low-cost discontinuous glass fiber, reinforced polyester-type sheet molding compound. Fairings, low-stressed skins, and fittings are possible areas of application for this material. It is also a flame-retardant (non-burning) material.

1-inch S-Glass/Epoxy, a one-inch chopped fiber system with an epoxy matrix, is a high-strength, high-cost material used in helicopter wheels.

CLOTH REINFORCED THERMOSETS may be used for all types of structures by providing the optimum fiber orientation for each type of loading. They are best used in multi-layer combinations in laminates or in sandwich construction.

Type 143 Cloth/E-Glass in an epoxy matrix is used in laminate and sandwich form in light aircraft and helicopters. Its use is restricted, as a rule, to secondary structure. However, the advancing state of the art of fiberglass composites and resin systems indicates that this material is a candidate for primary structure.

Type 143 Cloth/S-Glass and epoxy matrix system is a higher-strength and higher-cost composite than the E-Glass system. It is a candidate material when structural efficiencies outweigh material cost, or can be shown cost effective.

UNIDIRECTIONAL FILAMENT-REINFORCED COMPOSITES are in their infancy at present. Most of the composites are extremely expensive and are being used only in isolated cases. However, their superior structural efficiencies indicate that, projected ahead fifteen years from now, these composites, with reduced costs, will be potential candidates. They should be laminated in various fiber orientations, depending on the loading conditions.

TABLE V

PROMISING CANDIDATE MATERIALS - NON-METALLIC

MATERIAL	APPLI-CATION	F_{tu}	F_{ty}	F_{cu}	F_{su}	E_c	e	w	WEATHER-ABILITY	MATERIAL COST (2)	THERMAL CO-EFF. $\alpha/10^5$	$\frac{F_{tu}}{w}$	$\frac{F_{tu}}{w\ \$/LB}$	$\frac{\sqrt{E_c}}{w}$	$\frac{\sqrt{E_c}}{w\ \$/LB}$	$\frac{\sqrt[3]{E_c}}{w}$	$\frac{\sqrt[3]{E_c}}{w\ \$/LB}$	REF
												COMPARATIVE STRUCTURAL EFFICIENCIES						
	(3)	KSI	KSI	KSI	KSI	$\frac{PSI}{10^6}$	%	LB/IN^3		$ / LB	in/in/°F							
NON-REINFORCED																		
ABS (High Strength)	NT-FT	7.3	-	10.4	-	.180	20	.039	EXCEL	0.46	6.00	187	407	11	24	14	31	5,21
NON-CONTINUOUS FIBER REINFORCED																		
3/8 E-Glass/Nylon 6/10	FT	20	-	18	11	1.0	5-6	.048	EXCEL	1.34 (0.65)	2.50	418	(645)	21	(32)	21	(32)	13
1" S-Glass/Epoxy	FT	45	-	62	8	7.8	-	.060	EXCEL	4.00 (2.00)	-	750	(375)	46	(23)	33	(16)	14
E-Glass/Polyester	NT	20	-	26	-	1.99	-	.070	EXCEL	0.63	1.20	286	454	20	32	15	29	12
CLOTH REINFORCED																		
DAP Prepreg	NT-FT	49 (1)	-	-	-	2.6 (1)	-	.070 (1)	EXCEL	3.15 (1.58)	-	700	(446)	23	(14)	20	(12)	17,18
181 Cloth/E-Glass	NT-FT	45	-	45	-	3.3	-	.070	EXCEL	(1.00)	-	643	(643)	26	(26)	21	(21)	15
181 Cloth/S-Glass	NT-FT	94	-	65	-	4.2	-	.070	EXCEL	(2.00)	-	1340	(670)	29	(14)	23	(12)	16
FILAMENT REINFORCED (EPOXY MATRIX)																		
Unidirectional																		
Graphite	FT	95.9	-	56.5	5.2	15.4	-	.051	EXCEL	(1.00)	-	1870	(1870)	77	(77)	49	(49)	19
S-Glass	FT	210	-	120	13.6	7.6	-	.073	EXCEL	(2.00)	-	2880	(1440)	38	(19)	27	(13)	19
±45° Layers (t=.016 in)																		
Graphite	FT	5.8	-	31.6	24.8	2.1	-	.051	EXCEL	(1.00)	-	114	(114)	28	(28)	25	(25)	19
S-Glass	FT	17.7	-	37.3	50.0	2.5	-	.073	EXCEL	(2.00)	-	349	(174)	22	(11)	19	(10)	19
±45°,0° Layers (t=.024 in)																		
Graphite	FT	35.8	-	39.9	20.4	6.6	-	.051	EXCEL	(1.00)	-	702	(702)	50	(50)	37	(37)	19
S-Glass	FT	81.8	-	64.9	39.5	4.2	-	.073	EXCEL	(2.00)	-	1120	(560)	28	(14)	22	(11)	19
±45°,0°,0° Layers (t=.032 in)																		
Graphite	FT	50.8	-	44.0	17.8	8.8	-	.051	EXCEL	(1.00)	-	1000	(1000)	58	(58)	41	(20)	19
S-Glass	FT	113.8	-	78.7	34.0	5.1	-	.073	EXCEL	(2.00)	-	1560	(780)	31	(15)	24	(12)	19
±45°,0°,0°,0° Layers (t=.040 in)																		
Graphite	FT	59.8	-	46.5	15.9	10.2	-	.051	EXCEL	(1.00)	-	1170	(1170)	63	(63)	43	(43)	19
S-Glass	FT	133.1	-	86.9	30.5	5.6	-	.073	EXCEL	(2.00)	-	1825	(912)	32	(76)	24	(12)	19
				F_{cy}														
WOOD																		
Sitka Spruce	NT	9.4	5.3	3.5	1.0	1.4	-	.015	POOR	0.67	-	626	935	79	118	-	-	24
Mahogany/Poplar Plywd	NT	6.7	-	2.6	1.9	.9	-	.020	POOR	2.05	-	335	167	48	23	48	23	24
Spruce - Staypak	NT	35.8	25.9	4.3	1.3	4.7	.75	.047	FAIR	(4)	-	760	(4)	46	(4)	-	-	24

NOTES: (1) ESTIMATED (2) ( ) = 1982 ESTIMATE (3) NT - NEAR TERM, FT - FAR TERM (4) EXPERIMENTAL, NO PRICE AVAILABLE

Graphite filament/epoxy matrix composite exhibits exceptional structural efficiencies due to low density and high modulus.

S-Glass/epoxy matrix composites show superior tension efficiencies and modulus as compared with Graphite; however, they do not compare with the column and shear buckling efficiency of the Graphite system.

WOOD has been used as primary and secondary structure in light aircraft for many years. Although aluminum alloys have predominated the light aircraft field for the past decade, there are still a few airplanes being constructed of wood. Generally speaking, a wooden structure (such as a wing) is aerodynamically smoother and lighter than its metal counterpart. However, it is also more expensive to build. Another disadvantage to wood construction is its higher maintenance cost due to weathering and moisture absorption.

Sitka-Spruce is probably the most common wood used in light aircraft. It has a column efficiency more than twice that of the aluminum alloys.

Mahogany (poplar core) plywood is one of the more common woods used for skins. Its shear buckling efficiency is twice that of the aluminum alloys.

Spruce-Staypak is a compressed wood with greatly increased mechanical properties and higher density.

The promising candidates are now compared on the basis of types of members and concepts. Composites, which are anisotropic, require some mention being made as to allowables versus fiber orientation. When these materials in single-laminate configuration are loaded at an angle to the direction of the fibers, their strength is reduced considerably. The reduction in allowable is a function of the angle. Figure 17 illustrates the effect due to the low shear transfer capability of the resin matrix. For this reason, composite systems are normally found in various combinations of fiber-oriented layers. As an example, a wing skin panel carrying torsion might require three layers with the following orientation (see Figure 16):

SKIN PANEL FIBER ORIENTATION

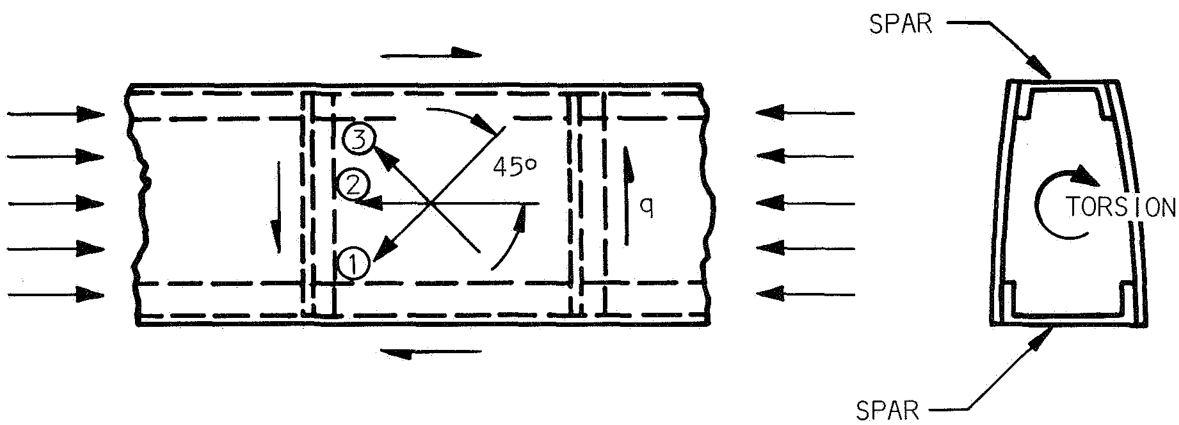
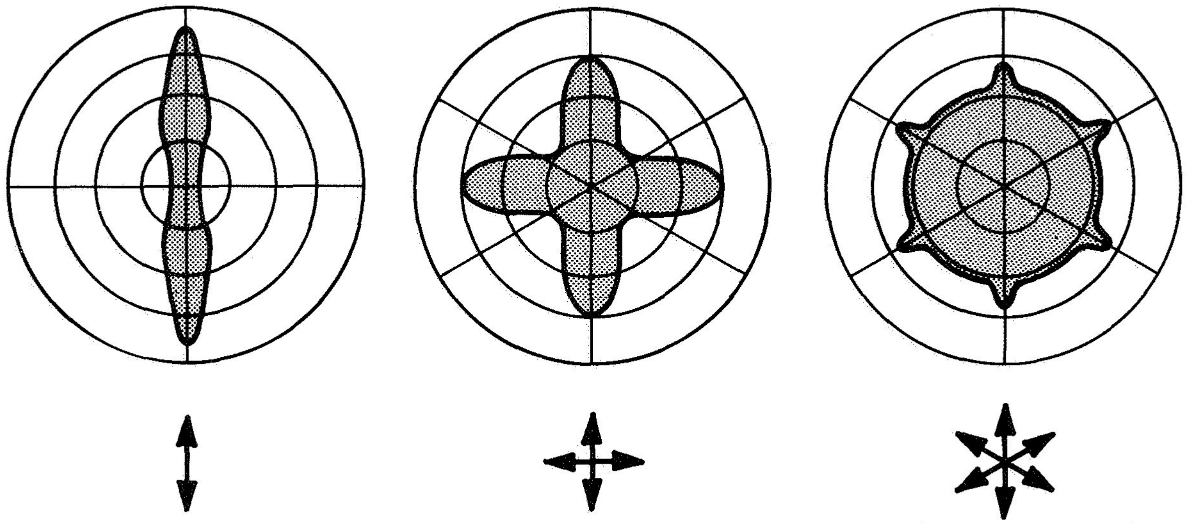


Figure 16

Layers (1) and (3) stabilize the panel against shear buckling; while layer (2) resists the direct shear and axial loading in the panel skin. Figure 17 also shows variation in strength with several combinations of fiber orientations. Figure 18 indicates variation in compression modulus with change of filament direction. Basic good design practices, when using laminated structure, are presented in Figure 19. Fiber-to-resin matrix proportion is another important relationship, strengthwise. A resin-rich composite is weakened by the influence of the lower strength matrix, while a resin-starved composite is unsatisfactory because of insufficient bonding between each fiber. In filament-wound structures, 70-to-85 percent by volume is considered normal for fiber content. Included in the comparisons, where appropriate, are several composite laminate combinations. A summary of the basic properties of candidates is presented in Table V. For more detailed or added information see Ref. 15

STRENGTH VS ANGLE OF STRESS IN TENSION FOR UNIDIRECTIONAL AND MULTI-DIRECTIONAL LAYUPS OF EQUIVALENT MATERIAL AND THICKNESS (REFERENCE 15 and 26)



FIBERS ARE ORIENTED AT 0° TO EACH OTHER, GIVING MAXIMUM STRENGTH PARALLEL TO WARP

NINETY-DEGREE ORIENTATION GIVES LESS MAXIMUM STRENGTH, BUT EQUAL STRENGTH FOR PARALLEL AND PERPENDICULAR LOADING

THIS FABRIC ORIENTATION YIELDS STILL LESS MAXIMUM STRENGTH, BUT EQUAL STRENGTH IN EACH DIRECTION OF FIBER WARP

Figure 17

COMPRESSION MODULUS VS PERCENT FILAMENT IN 0° DIRECTION (REFFERENCE 19)

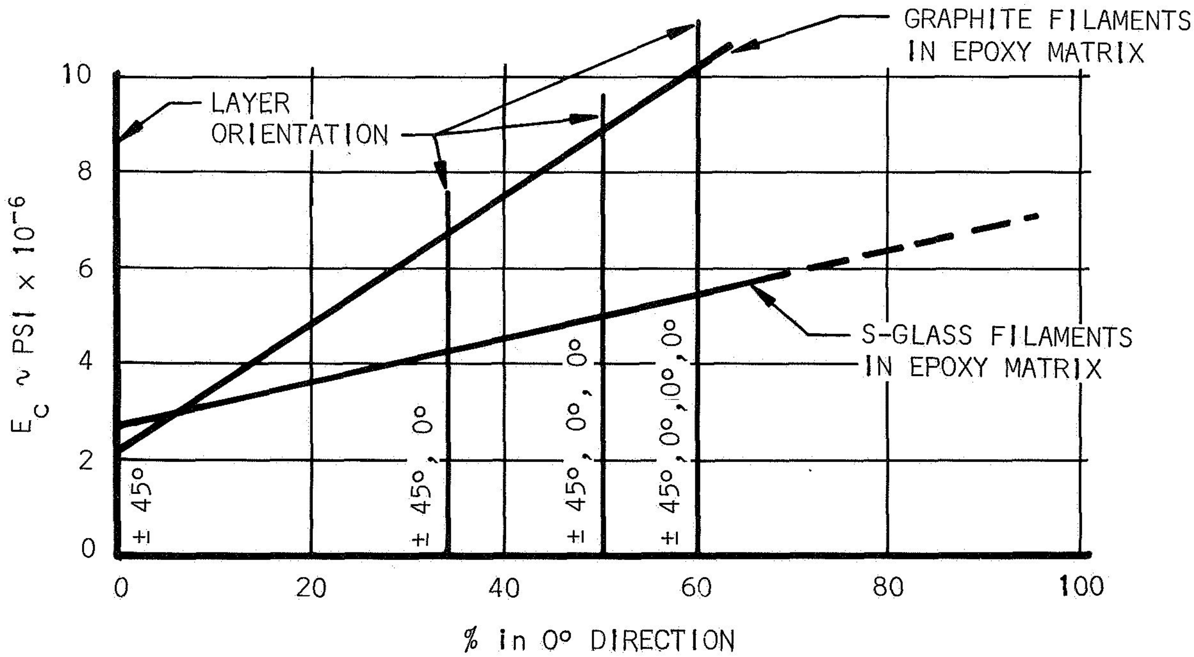
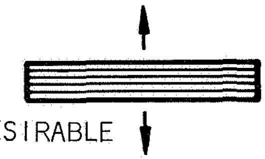


Figure 18

RELATION BETWEEN DIRECTION OF LAMINATIONS
AND DIRECTION OF LOAD APPLICATION

TENSION

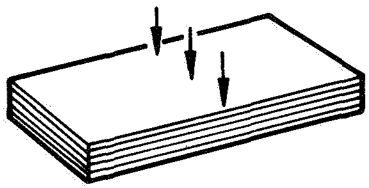


RECOMMENDED

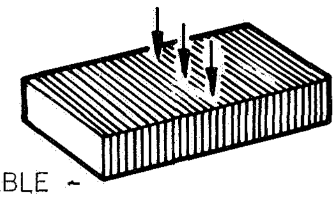
UNDESIRABLE

TENSILE STRESSES SHOULD BE SUSTAINED BY LAMINATIONS, NOT ACROSS BONDING PLANE

COMPRESSION



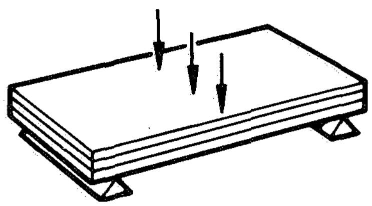
RECOMMENDED -
FLATWISE AT RIGHT ANGLE TO LAMINATIONS



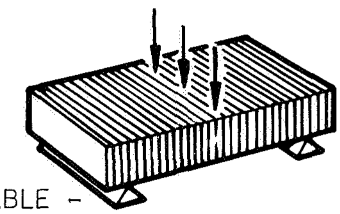
UNDESIRABLE -
EDGEWISE PARALLEL TO LAMINATIONS

COMPRESSION STRENGTH OF LAMINATES IS GREATER FLATWISE THAN EDGEWISE

FLEXURE



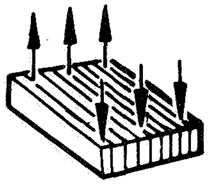
RECOMMENDED -
FLATWISE AT RIGHT ANGLE TO SPAN

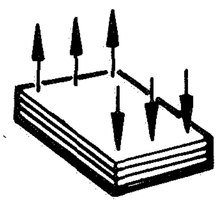


UNDESIRABLE -
LAMINATIONS AT RIGHT ANGLE TO SPAN

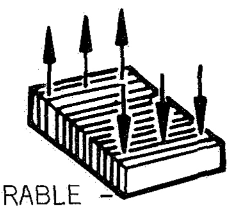
BENDING STRESSES SHOULD BE SUSTAINED BY LAMINATIONS, NOT ACROSS BONDING PLANE

SHEAR





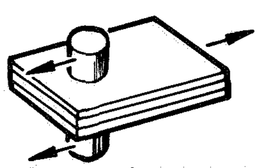
RECOMMENDED -
FLATWISE AT RIGHT ANGLES TO LAMINATIONS



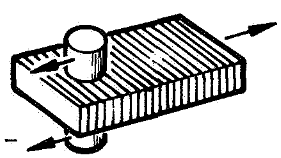
UNDESIRABLE -
EDGEWISE PARALLEL TO LAMINATIONS

SHEARING STRESSES SHOULD OCCUR IN A PLANE NORMAL TO LAMINATIONS
TO PREVENT CLEAVAGE ACROSS BONDING PLANES

BEARING



RECOMMENDED -
LOAD DISTRIBUTED TO LAMINATIONS



UNDESIRABLE -
LOAD CARRIED THRU BOND

BEARING STRESSES SHOULD BE APPLIED THRU LAMINATIONS
RATHER THAN ACROSS BONDING PLANES

Figure 19

AXIALLY LOADED MEMBER

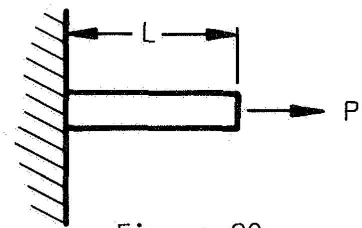


Figure 20

Tension Members

Figure 21 shows weight per inch versus axial load (4,000 pounds maximum) for the various materials. The ordinate provides for the use of an efficiency factor which might be encountered under conditions of riveting or welding.

Derivations: $f = \frac{P}{A}$, $W = A\,L\,w$, $A = \frac{W}{L\,w}$ and $f = K_{eff}\,F$

To develop curves of $\frac{WK}{L}$ efficiency versus Tension Load P , let:

$$K_{eff}\,F = \frac{P}{W/L\,w}$$

$$K_{eff}\,\frac{W}{L} = \frac{P}{F/w} \quad \text{(Figure 21)}$$

SYMBOLS:
- f = Stress
- A = Cross section area
- W = Weight
- w = Density
- K_{eff} = Efficiency factor
- F = Smaller of F_{tu} or: 1.5 F_{ty}

WEIGHT/IN. VS TENSION LOAD

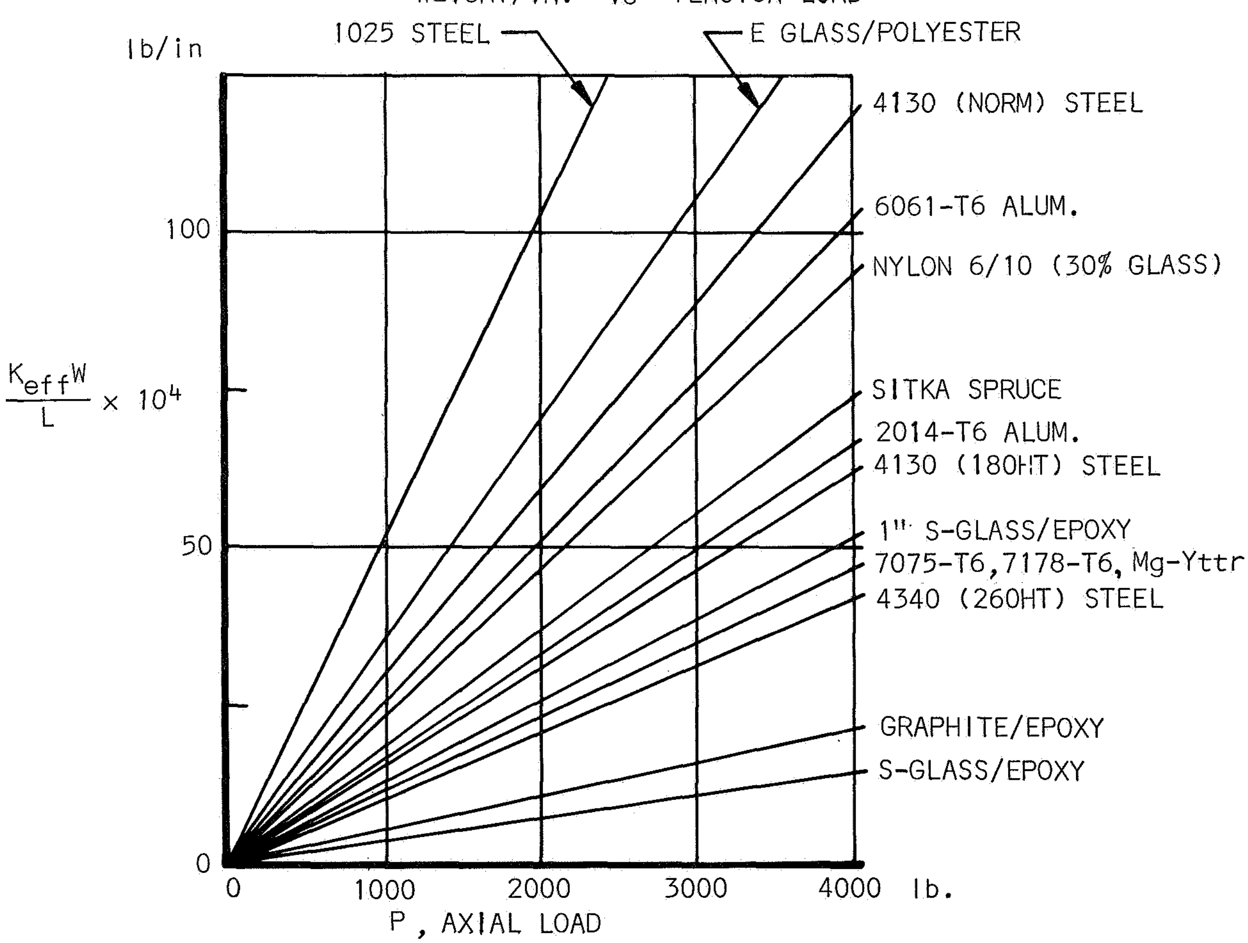


Figure 21

Simple Columns (assume round tubes)

Structural indexes were used to assist in the evaluation of promising candidate materials when applied as simple columns. As defined in reference 27, a structural index is a measure of loading intensity and has the advantage of eliminating the effect of size in dealing with allowable stresses. For a simple column, the structural index becomes P/L^2. Derivations:

Primary buckling $$F_c = \pi \left(\sqrt{E_t}\right)\left(\sqrt{\frac{D}{8rt}}\right)\left(\sqrt{\frac{P}{L^2}}\right)$$

and crippling $$F_{cr} = K_2 \frac{\sqrt{EE_t}}{D/t}$$

Equating the two equations gives optimum value of D/t

$$\left(D/t\right)_{opt} = 2\left(\frac{K_2^2\, E}{\pi P/L^2}\right)^{1/3} = .742\left(\frac{E}{P/L^2}\right)^{1/3} \qquad [K_2 = .40 \text{ (Reference 27)}]$$

Figure 22 plots D/t ratios versus structural index for the materials under consideration.

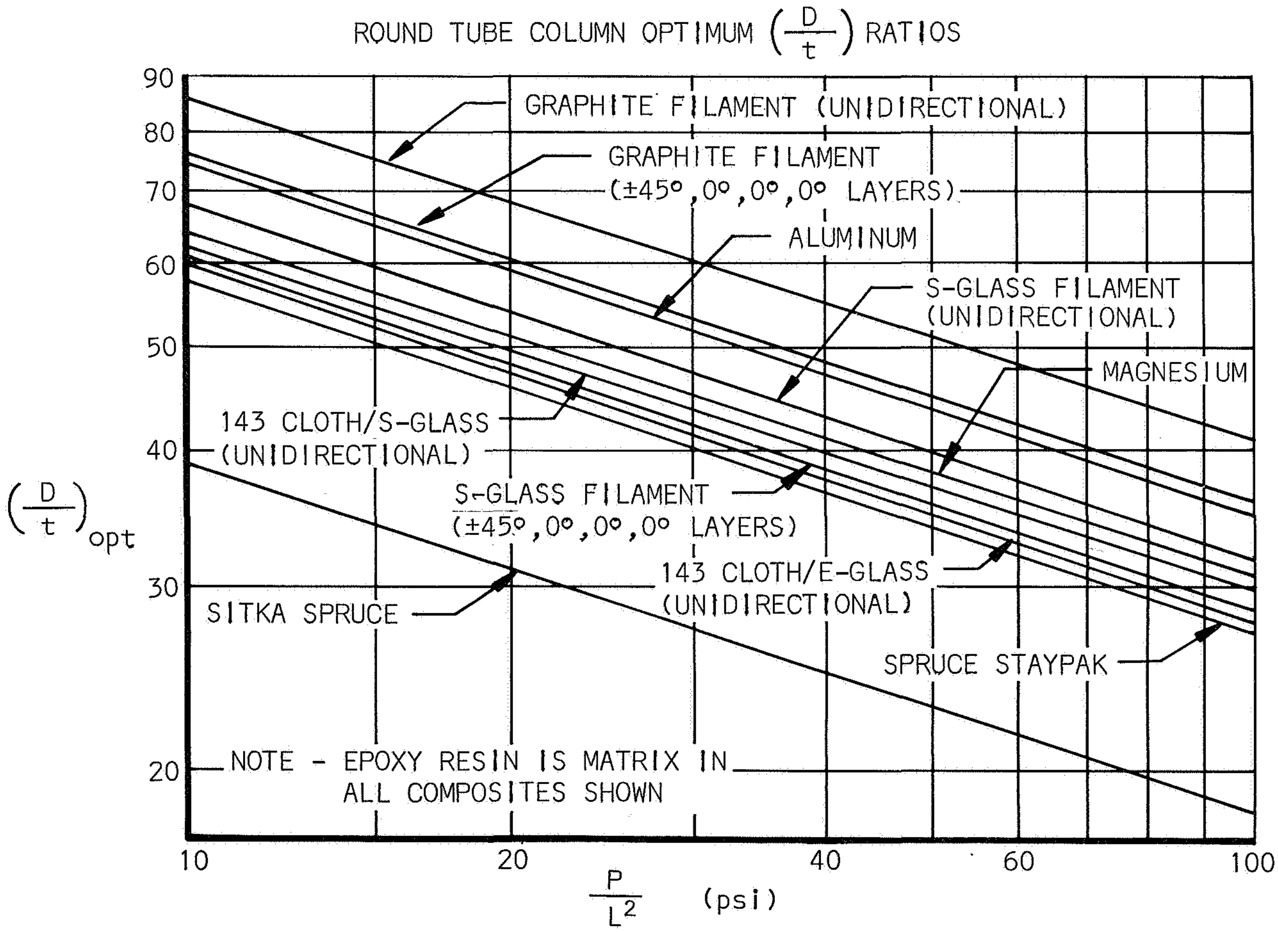


Figure 22

To obtain allowable compression stresses for optimum round tube columns, substitute the value for optimum D/t in the primary buckling equation:

$$P/L^2 = \frac{8f^3}{\pi K_2 E^{1/2} E_t^{3/2}} = \frac{6.37 f^3}{E^{1/2} E_t^{3/2}}$$

For study purposes, limit f to $.80F_{cy}$.

The allowable F_c may then be calculated and plotted for various materials, as shown in Figure 23.

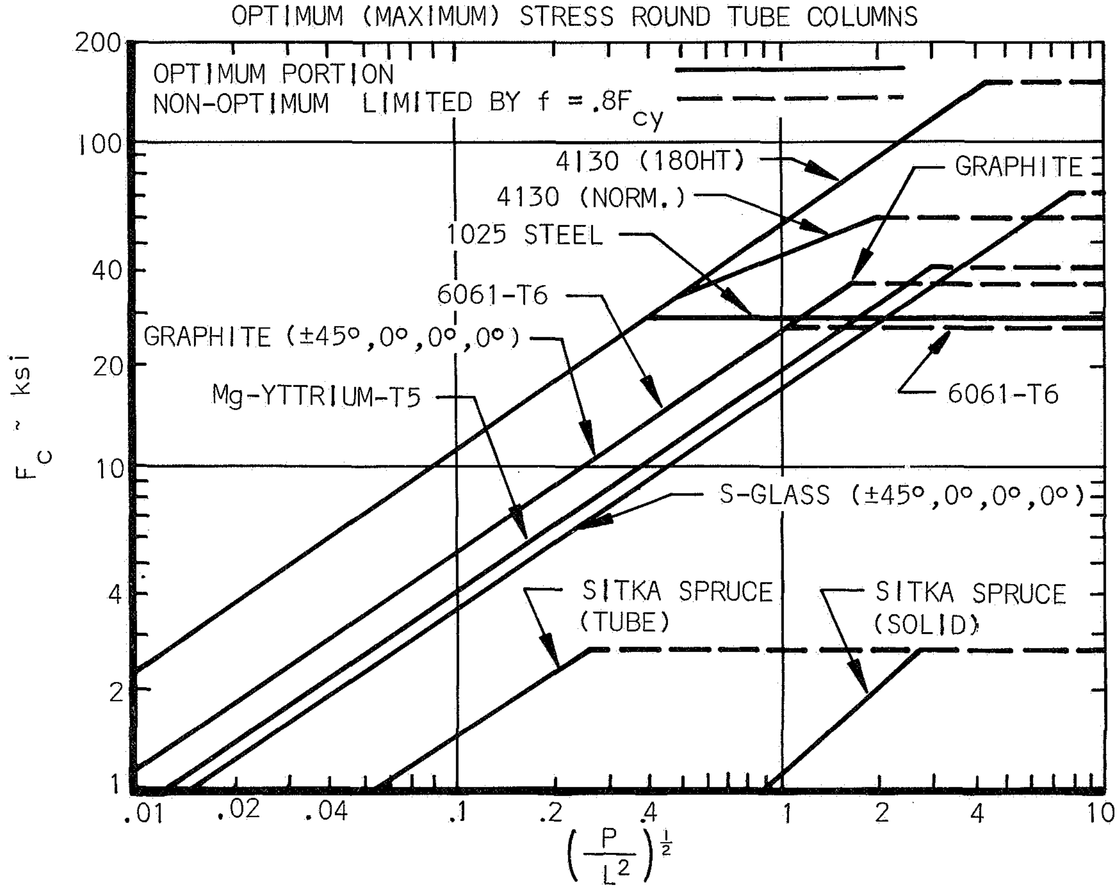


Figure 23

It is now possible to develop a formula for minimum weight, as follows:

(1) Divide structural index by allowable F_c and multiply by density of material: $\frac{P\,(w)\,/L^2}{F_c}$

(2) By substituting $\frac{P}{F_c} = A$ & $w = \frac{W}{AL}$, the following identity is obtained: $WC/L^3 = \frac{P/L^2}{F_{c/w}}$, where C is restraint coefficient.

Values for WC/L^3 versus P/L^2 may now be determined and plotted for a number of materials (see Figure 24).

MINIMUM WEIGHT ROUND TUBE COLUMNS

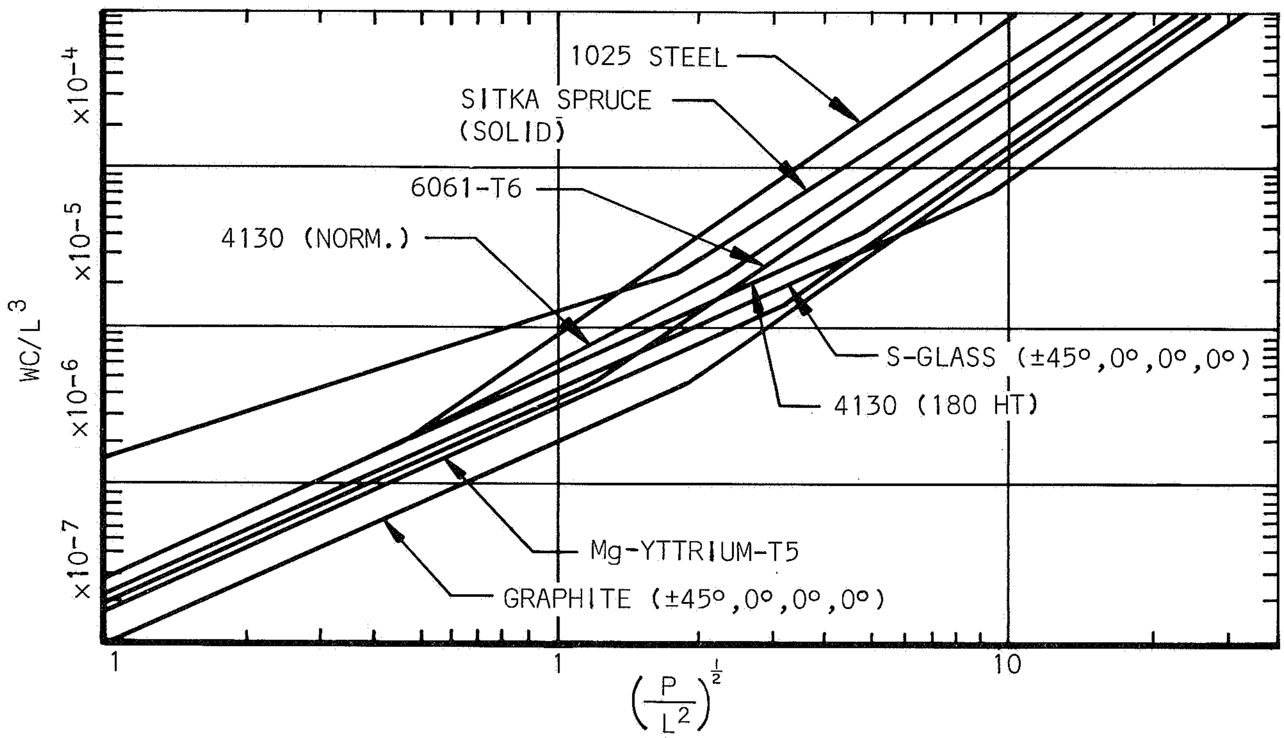


Figure 24

Compression Structure

Probably the most detailed and extensive evaluation of structure occurs during the design of compression critical sections of the airframe. The section under compression is generally treated either as a wide column or a compression panel. The wide-column approach is used when the length of the panel is short compared to its width, as in a multi-rib wing box. A compression panel concept is assumed when the length of the panel is long compared to its width, as in a multi-spar wing box.

The wide-column analysis assumes primary buckling between the ribs, which provide simple supports for loaded edges of the column. The following equation, taken from reference 28, is a result of equating general and local instability formulas:

$$\frac{N_x}{L\bar{\eta}E} = \varepsilon\left(\bar{t}/L\right)^2$$

Where:
- N_x = compressive load in pound/inch
- L = length of column in inches
- $\bar{\eta}$ = plasticity reduction factor
- E = modulus of elasticity, psi
- $\bar{t}$ = cross-sectional area per unit width
- ε = efficiency factor, a function of buckling coefficient & shape factor

The analysis of compression panels is based upon all edges of the panel being simply supported, while plate theory expressions for local and general stability are equated to obtain the following equation:

$$\frac{N_x}{b\bar{\eta}E} = \varepsilon\left(\bar{t}/b\right)^n$$

Where:
- b = width of plate
- n = an exponent which is a function of configuration

In the evaluation of wide-column and compression panel concepts, truss core sandwich, honeycomb sandwich, flat plate, and zee-stiffened plate construction will be considered for each case.

Minimum area equations for optimized wide columns and compression panels of zee-stiffened plate, flat plate, and truss core sandwich construction are presented in Table VI. Efficiency factors, ε, were obtained from reference 28, while the plasticity reduction factor, $\bar{\eta}$, was taken as unity for all cases.

TABLE VI
MINIMUM AREA EQUATIONS FOR OPTIMIZED WIDE COLUMNS AND COMPRESSION PANELS (Reference 28)

TYPE OF CONSTRUCTION	WIDE COLUMN	COMPRESSION PANEL
Zee-Stiffened Plate	$\frac{N_x}{LE} = 0.911\ (\bar{t}/L)^2$	$\frac{N_x}{bE} = 1.030\ (\bar{t}/b)^{2.36}$
Truss Core Sandwich	$\frac{N_x}{LE} = 0.605\ (\bar{t}/L)^2$	$\frac{N_x}{bE} = 1.108\ (\bar{t}/b)^2$
Flat (unstiffened) Plate	$\frac{N_x}{LE} = 0.823\ (\bar{t}/L)^3$	$\frac{N_x}{bE} = 3.62\ (\bar{t}/b)^3$

Minimum area curves for truss core sandwich, honeycomb sandwich, flat plate, and zee-stiffened plate of wide column and compression panel construction are shown in Figures 25 and 26.

The zee-stiffened plate, flat plate, and truss core curves were developed from the data in Table VI. Minimum area curves for honeycomb sandwich were obtained from reference 28. Curves were generated by calculating typical weights and strengths, and algebraically converting the results to the general form of the other configurations. As stated in reference 28, the high efficiency of honeycomb sandwich construction is attributed to the fact that the full compressive strength of face sheets can be utilized by reducing the cell size of the honeycomb core.

A panel optimization computer program was used in reference 19 for evaluating numerous filament-wound materials in truss core and honeycomb sandwich construction. These configurations, in their optimum proportions of unidirectional to cross-ply fibers are pictured in Figure 27. By utilizing data from reference 19, optimum weight and corresponding core thickness versus structural index may be determined for graphite and S-Glass wide columns and compression panels.

MINIMUM AREA CURVES - WIDE COLUMN CONCEPT

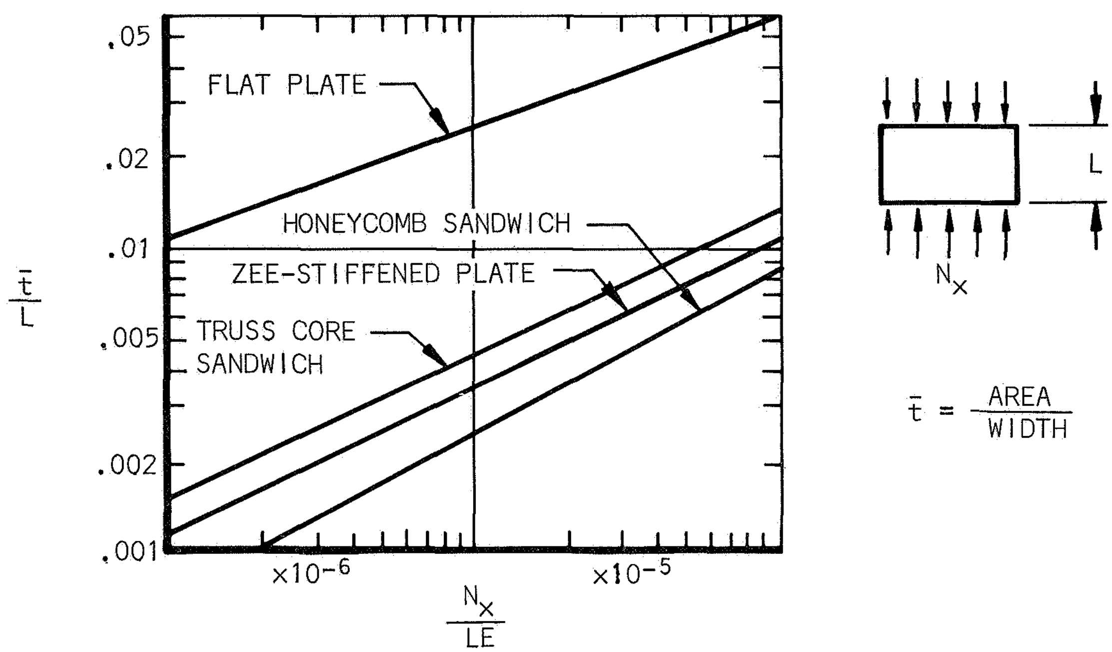


Figure 25

$\bar{t}$ = Equivalent cross sectional area/unit width of panel of all material effective in carrying axial load.

MINIMUM AREA CURVES - COMPRESSION PANEL CONCEPT

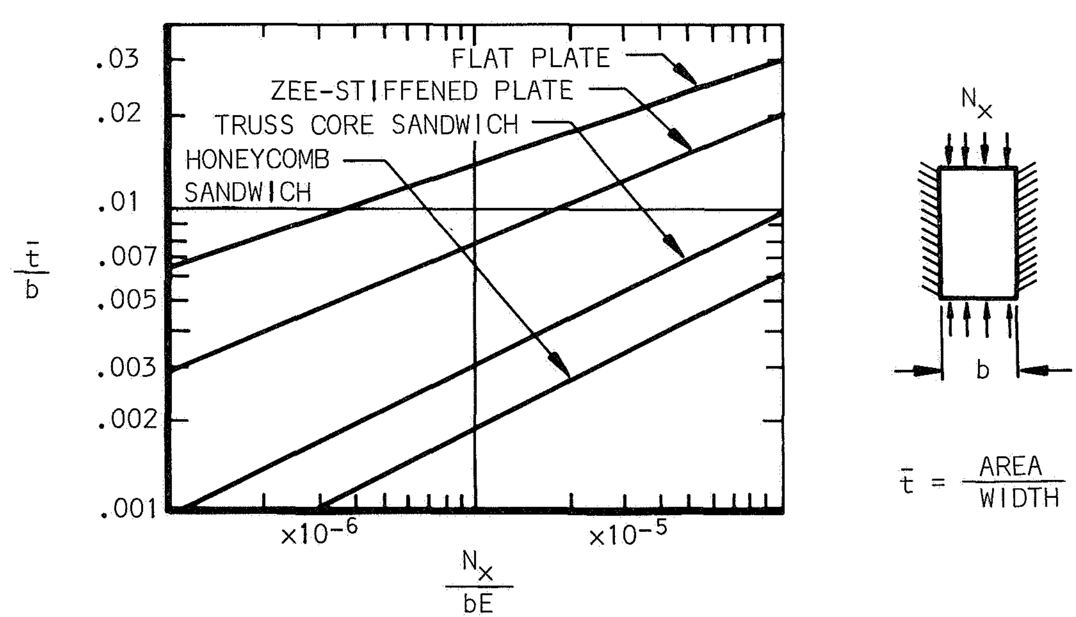
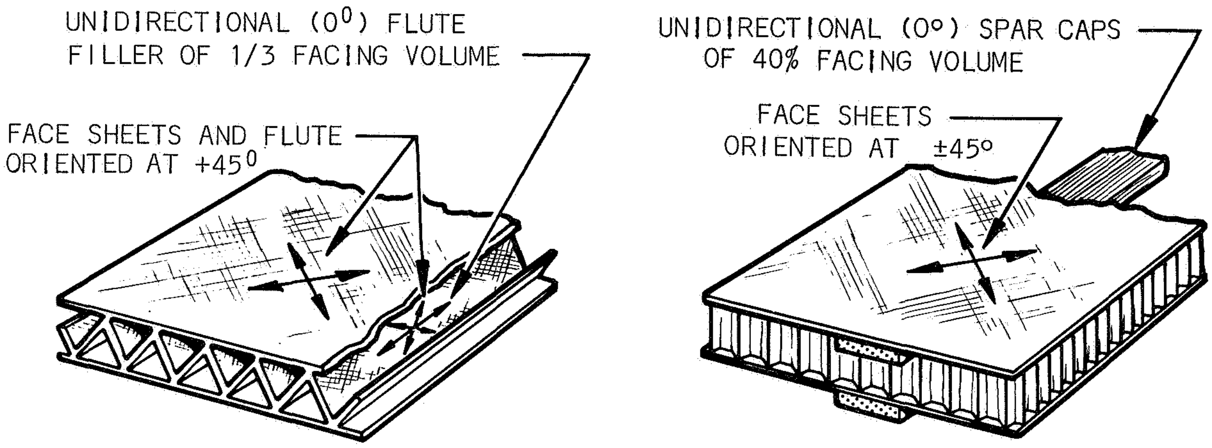


Figure 26



CORRUGATION SANDWICH PANEL HONEYCOMB SANDWICH PANEL

Figure 27

Resulting values are plotted in Figures 28 through 30. Optimized configuration weights reflect ±45° fiber orientation in the skins for the most efficient alignment to react torsional shear. Minimum skin gages are set at .020 inches. Four failure modes considered were: general buckling, face wrinkling, intercell buckling, and shear crimping.

Minimum weight diagrams can also be developed from minimum area curves in Figures 25 and 26, as follows:

(1) Multiply ordinate $\bar{t}/L$ by material density, w :

$$w\bar{t}/L = W/bL^2 \quad \text{because} \quad W = bL\bar{t}w \ , \ w = W/bL\bar{t}$$

(2) Multiply abscissa N_x/LE by material modulus, E :

$EN_x/LE = N_x/L$; the weight is thus presented as a function of the structural index: N_x/L (or q/L).

Minimum weights for various materials and concepts are shown in Figures 32 and 33.

In the discussion of sheet stringer-type wide columns, mention should be made of extruded Y stringers developed by NACA (NACA TN 1389) for increasing allowable stresses in compression structures. Figure 34 compares allowable stress versus structural index of sheet stringer wide columns constructed of 2024 and 7075 Y-stringers against a 2024 conventional stringer envelope.

These same constructions are compared on a weight basis in Figure 35 which was derived from optimum stress curves by dividing N_x/L by F_c and then multiplying by w to obtain:

$$\left(N_x/L\right)\left(1/F_c\right)\left(w\right) = \bar{t}w/L = W/bL^2$$

THEORETICAL VS OPTIMUM WIDE COLUMN WEIGHTS
GRAPHITE AND S-GLASS FILAMENT SANDWICH CONSTRUCTION

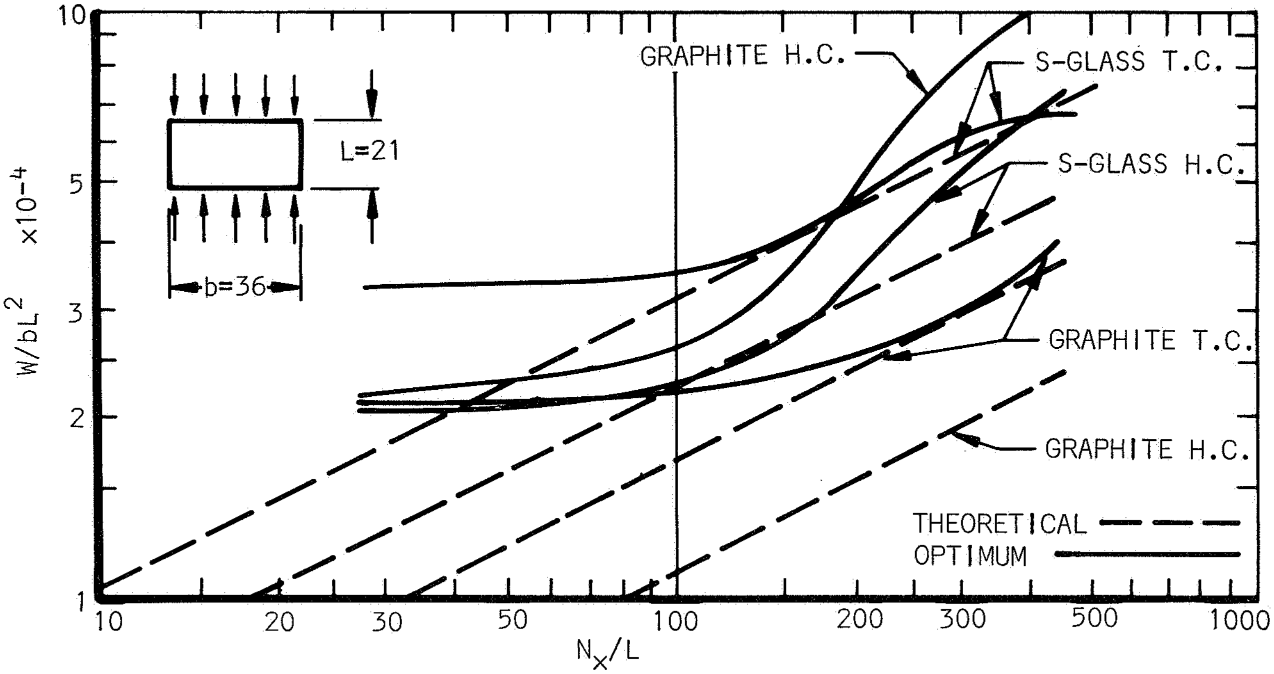


Figure 28

THEORETICAL VS OPTIMUM COMPRESSION PANEL WEIGHTS
GRAPHITE AND S-GLASS FILAMENT SANDWICH CONSTRUCTION

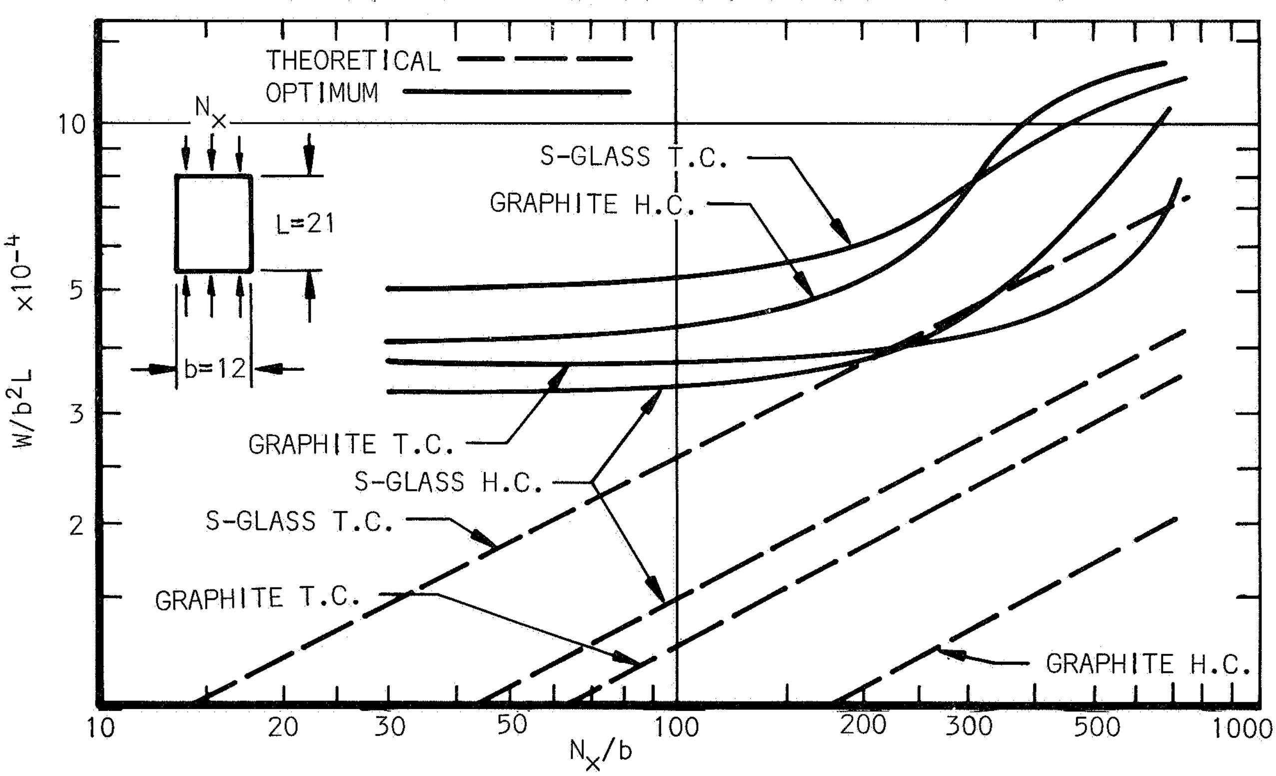


Figure 29

THEORETICAL VS OPTIMUM CORE THICKNESSES
GRAPHITE AND S-GLASS FILAMENT SANDWICH CONSTRUCTION

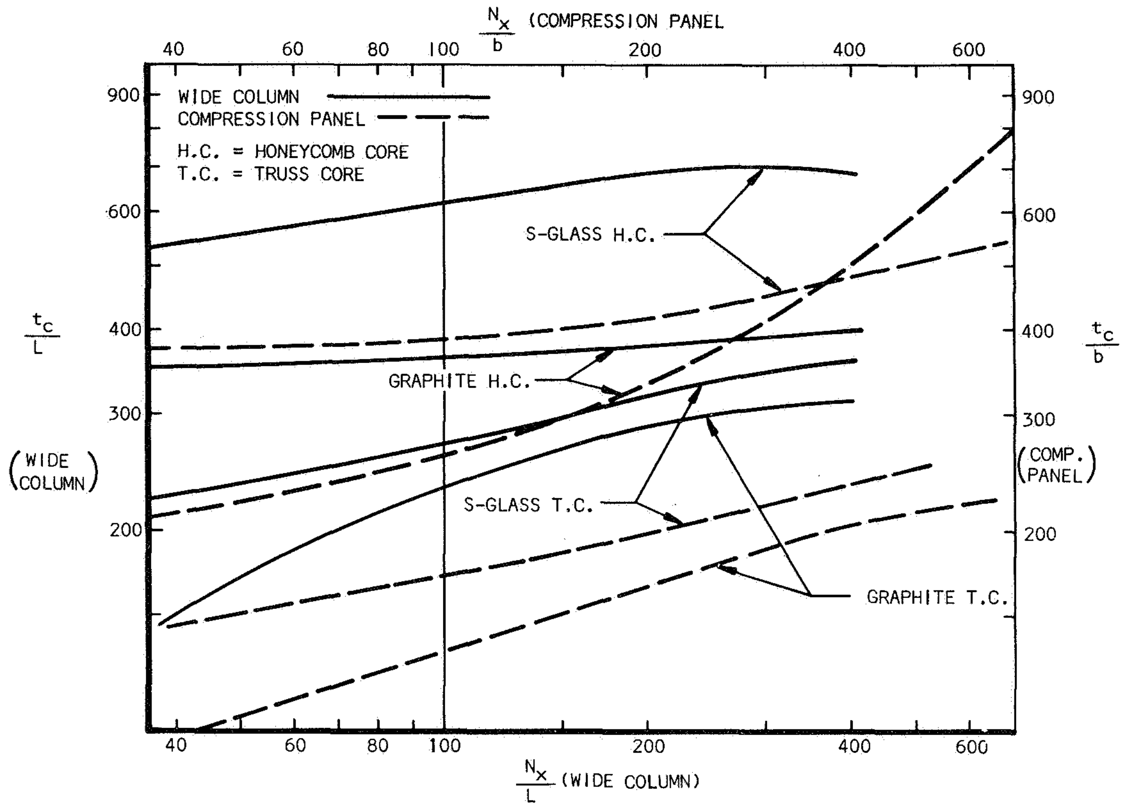


Figure 30

OPTIMUM WEIGHT - WIDE-COLUMN CONCEPT

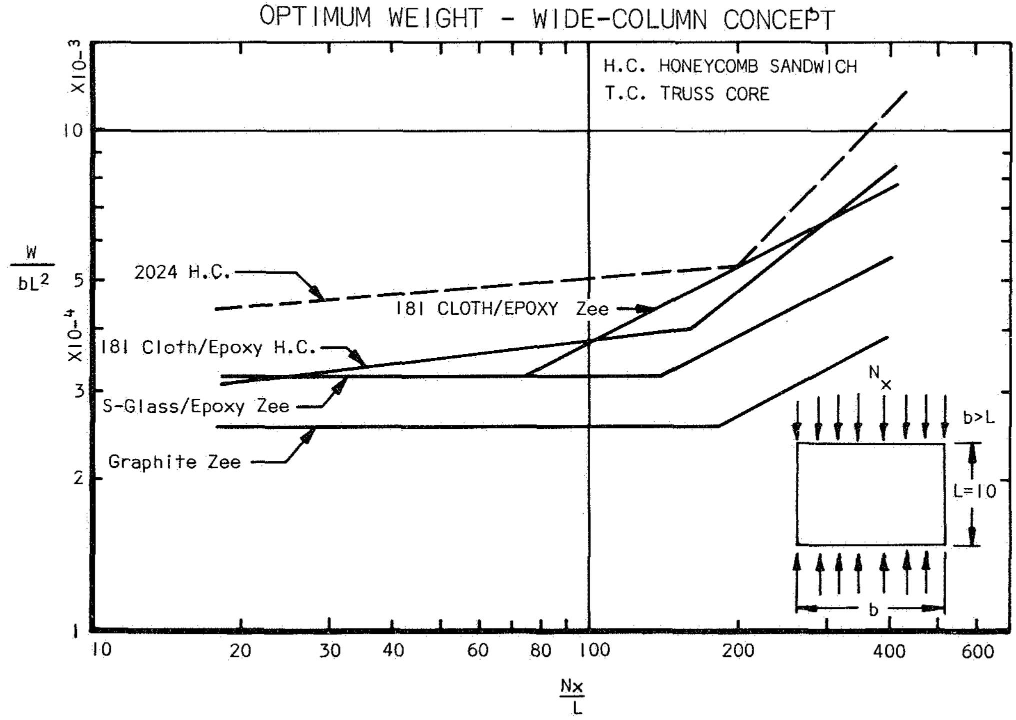


Figure 31

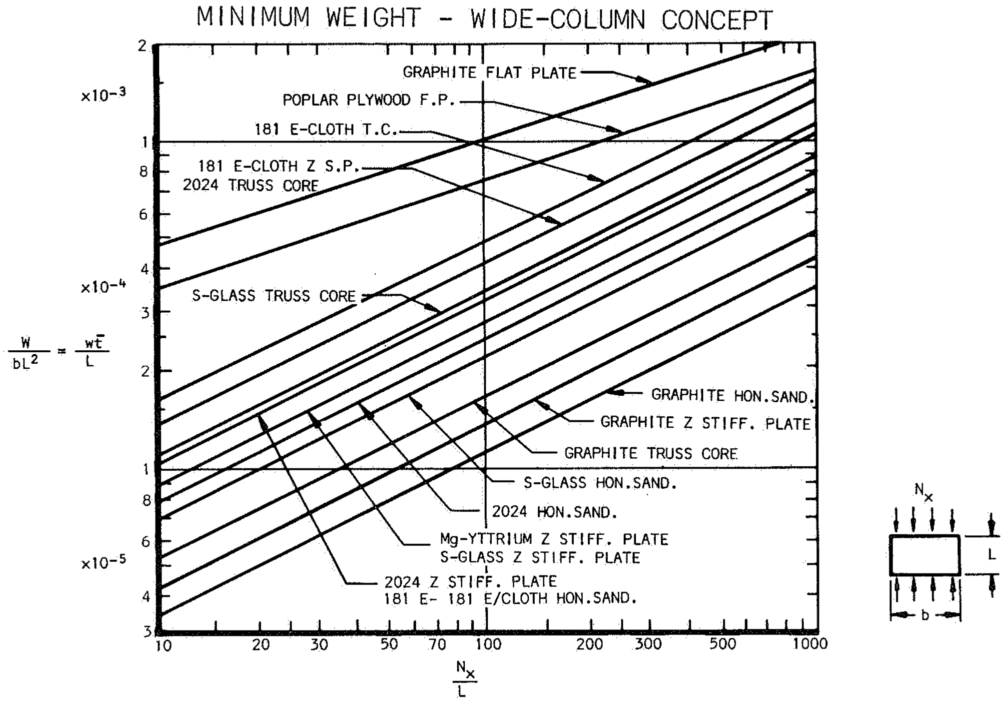


FIGURE 32

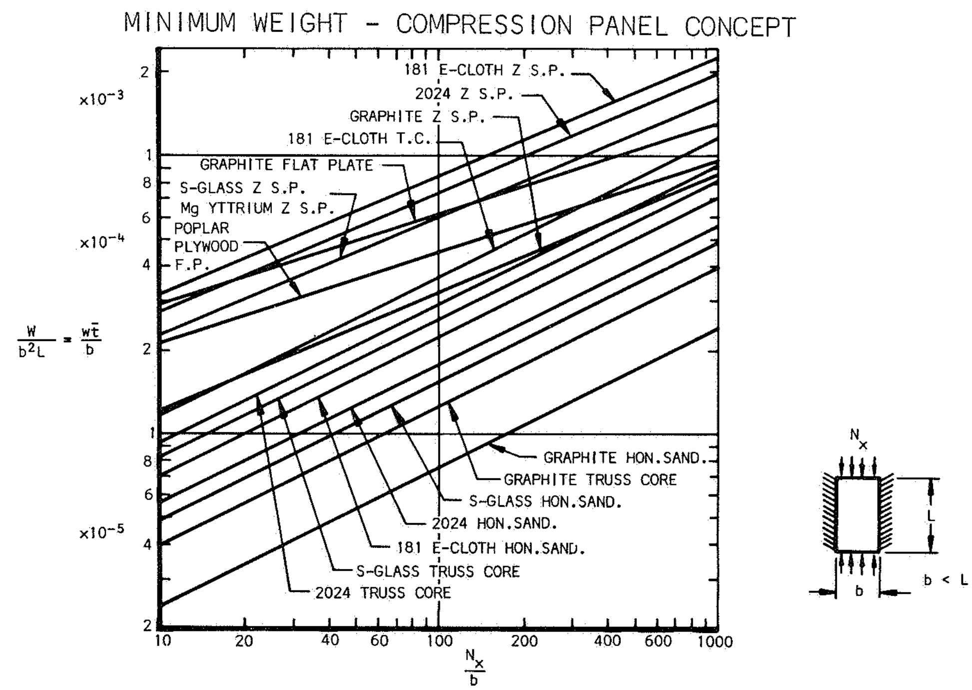


FIGURE 33

S.P. = STIFFENED PLATE
F.P. = FLAT PLATE
H.C. = HONEYCOMB SANDWICH = HONEYCOMB CORE
T.C. = TRUSS CORE

OPTIMUM (MAX.) STRESS - WIDE COLUMNS
ALUMINUM SHEET - STRINGER TYPE

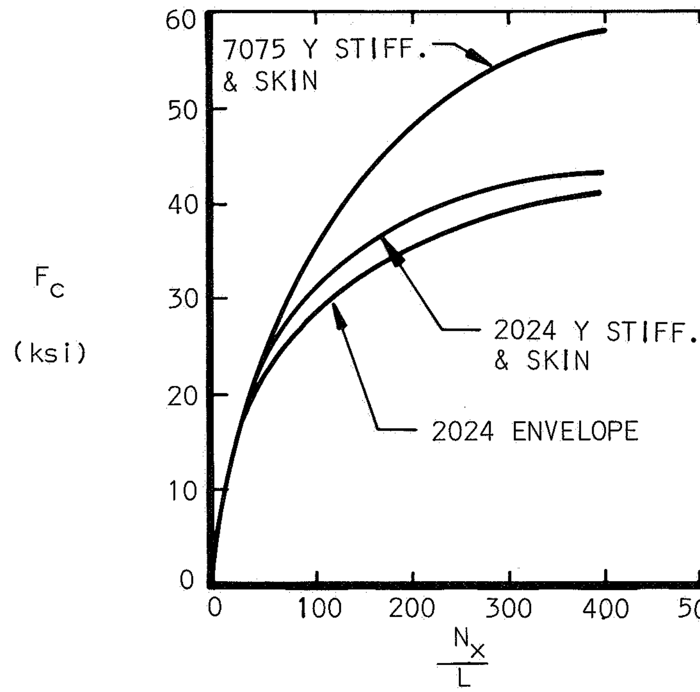


Figure 34

MINIMUM (OPT.) WEIGHT - WIDE COLUMNS
ALUMINUM SHEET - STRINGER TYPE

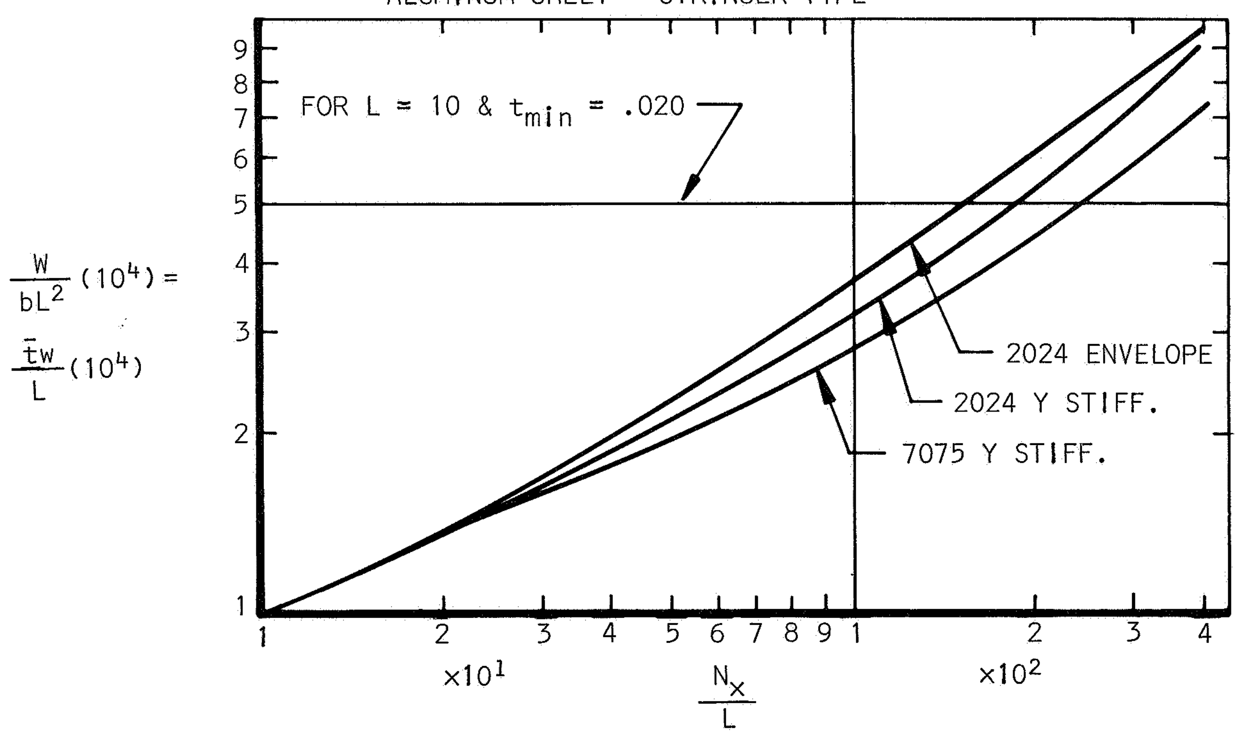


Figure 35

Shear Panels

Wing, fuselage, and empennage skins on small aircraft (including helicopters) are of light-gage construction. Loading intensities due to torsional shear are low level; therefore, the panels are normally designed for shear buckling at the 1-to-1.2 g level. This requirement is established for appearance purposes since the panel itself has ample strength to carry the ultimate torsional shear flow as a tension field member.

Materials for shear panel application are compared on a thickness basis in Figure 36. The curves were obtained through a substitution and division process of the shear buckling equation for flat plates.

Shear buckling: $\tau_{cr} = \dfrac{K_s E_c t^2}{b^2}$

$\tau_{cr} = N_{xy}/ t$,

$N_{xy} = q =$ torsional shear flow;

Where:
- τ_{cr} = shear stress at which panel will buckle
- K_s = shear buckling coefficient dependent upon edge conditions around panel (Ref.Fig.37)
- b = short side dimension of panel
- t = panel thickness
- E_c = compression modulus of elasticity

Therefore:

$$N_{xy}/t = \frac{K_s E_c t^2}{b^2} , \quad N_{xy} = \frac{K_s E_c t^3}{b^2}$$

Obtain structural index (abscissa) : $N_{xy}/b = \dfrac{K_s E_c t^3}{b^3} = K_s E_c (t/b)^3$

Calculate ordinate: $t/b \sqrt[3]{K_s} = (N_{xy}/bE)^{1/3}$

Minimum weights versus structural indexes for flat plate shear panel materials are presented in Figure 38. Curves were derived by multiplying shear buckling equations, as modified for minimum thickness form, by material density, w :

$$wt/b \sqrt[3]{K_s} = w \left(N_{xy}/bE\right)^{1/3}$$

But: $W = wabt$, $w = W/abt$

Where: W = panel weight
a = long side of panel

Therefore: $W/b^2a = \sqrt[3]{K_s} = w \left(N_{xy}/bE\right)^{1/3}$

Shear buckling coefficients, K_s , for various edge conditions are shown in Figure 36.

Compression Flanges

In reviewing candidate materials for use as compression flanges on spars and similar bending members, the following structural index will be applied to represent crippling efficiency:

$$S = \frac{\sqrt{F_{cy} E_c}}{w}$$

This relationship is in general agreement with Needham's equation for crippling in reference 29 and assumes b/t , flange width to thickness ratio, to remain constant.

Crippling structural efficiencies for candidate materials are illustrated in Figure 39.

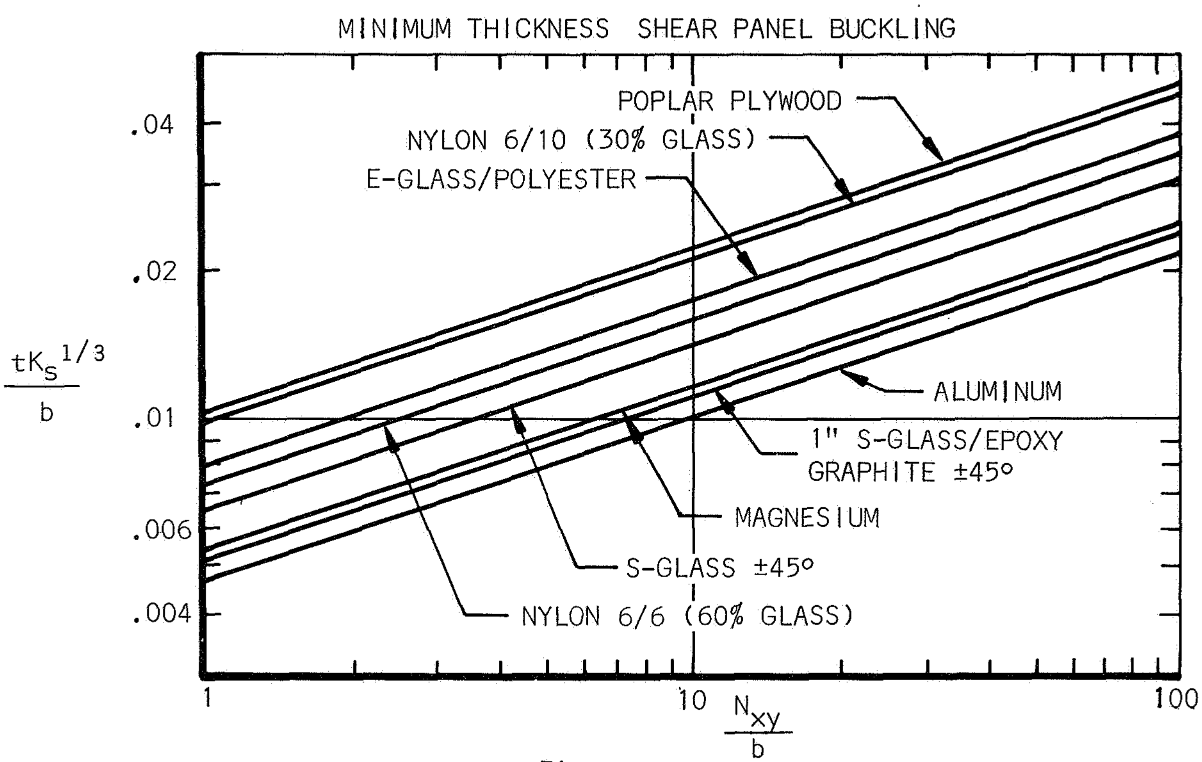


Figure 36

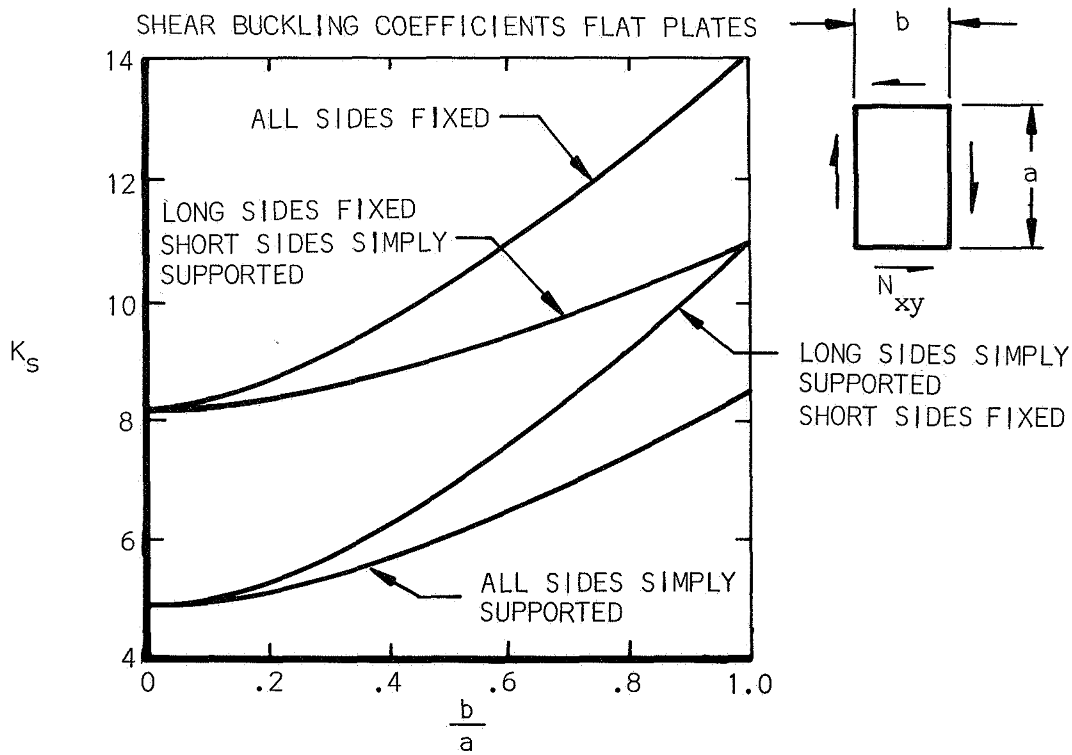


Figure 37

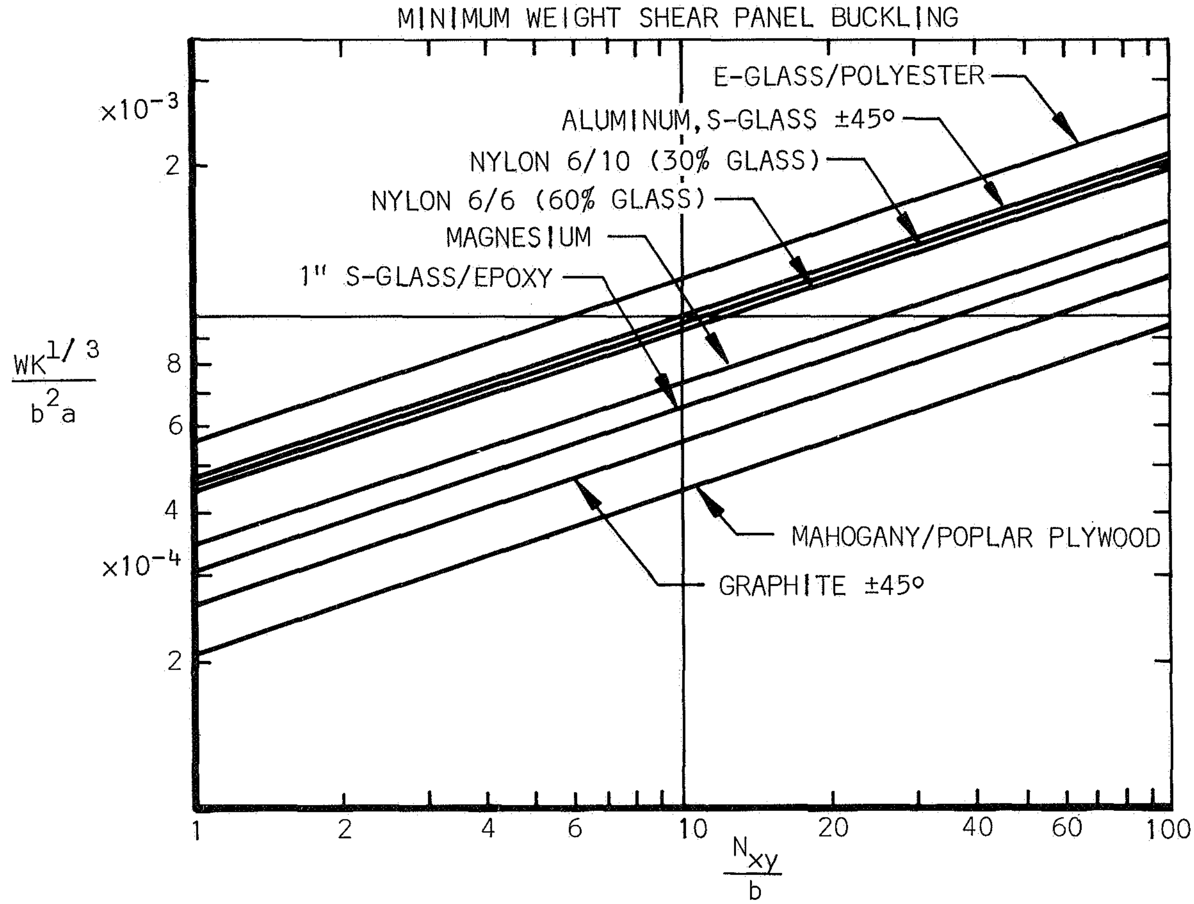


Figure 38

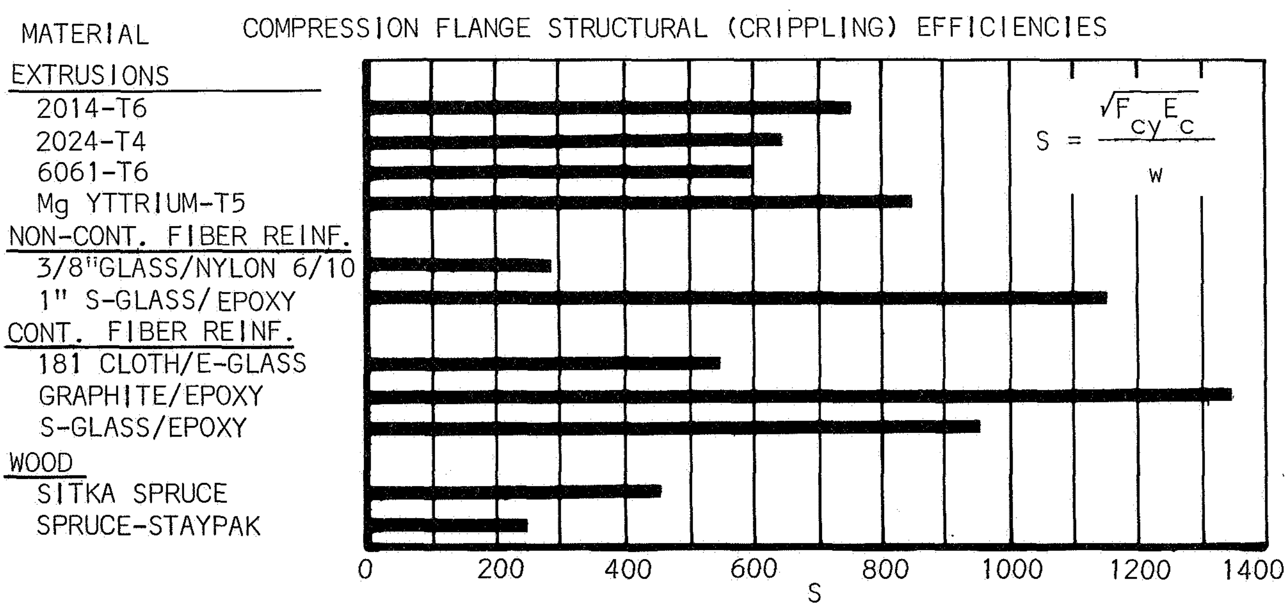


Figure 39

Installation Costs

In determining the feasibility of various structural material concepts, the total cost of the installation must be compared against the dollar's worth value of a pound of material saved. The installation cost includes material cost plus fabrication cost. In order to justify a material/concept change, one of the following conditions must be satisfied:

(1) significant weight savings with no increase in total installation cost

(2) significant decrease in installation cost with no appreciable increase in weight

(3) significant weight savings with significant cost savings

The dollar's worth value of a pound of weight saved for the typical four-place light airplane, which will be discussed in the following main section, has been calculated versus service life. See Figure 40.

In the following evaluation of required break-even costs versus material/concept, a $2.00 per pound value for a pound of weight saved will be used for the light aircraft, based on a 333 hr/yr utilization rate with an original single-owner expectancy of 20 years. See Appendix B for a detailed discussion.

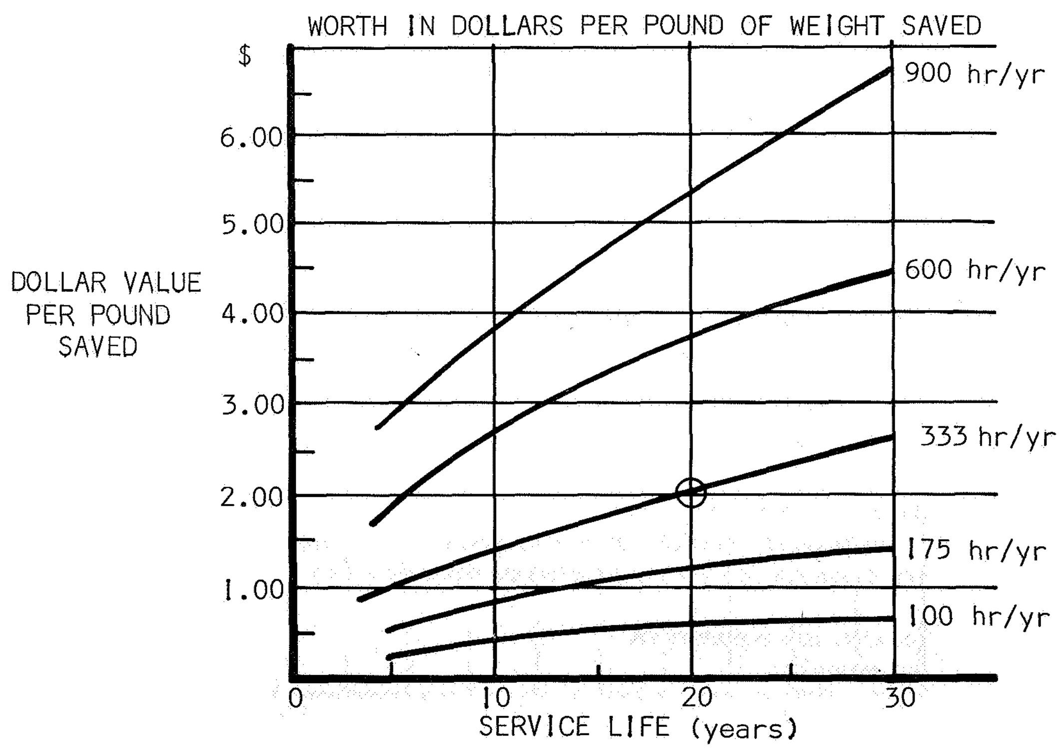


Figure 40

A typical light aircraft will be used as a baseline against which weights and costs will be compared. This airplane utilizes aluminum sheet metal stringer-stiffened construction, with a two-spar wing. Its installation cost per pound, C_{ib}, is $7.00 for an empty weight of 1500 lbs. (ref.pg.10).

To determine the required break-even fabrication cost per pound for candidate materials/concepts, the following derivation is performed, noting that the letters n and b in the subscripts indicate new candidate and base line materials, respectively:

Installation Cost = Material Cost + Fabrication Cost; $P_i = P_m + P_f$

* Where: P_m = Mat'l Cost/lb x Weight = C_mW, And: P_f = Fab'n Cost/lb x Weight = C_fW

* Therefore: $P_i = W\left(C_m + C_f\right)$. Substituting candidate mat'l for base line mat'l:

$\frac{\text{Price Increase}}{\text{Weight Decrease}} \leq$ Dollar's worth of a pound of material saved, or $\frac{\Delta P}{\Delta W} \leq C_w$

Where: $\Delta P = P_{in} - P_{ib} = W_n\left(C_{mn} + C_{fn}\right) - W_b\left(C_{mb} + C_{fb}\right)$, and $\Delta W = W_b - W_n$

$$\text{Therefore, } C_w = \frac{W_n\left(C_{mn} + C_{fn}\right) - W_b\left(C_{mb} + C_{fb}\right)}{W_b - W_n}, \quad \text{but: } W_n = W_b\left(S_b/S_n\right)$$

Where: S_b = structural efficiency of baseline material

S_n = structural efficiency of new candidate material

$$\text{So: } C_w = \frac{W_b\left(S_b/S_n\right)\left(C_{mn} + C_{fn}\right) - W_b\left(C_{mb} + C_{fb}\right)}{W_b - W_b\left(S_b/S_n\right)} = \frac{\left(S_b/S_n\right)\left(C_{mn} + C_{fn}\right) - \left(C_{mb} + C_{fb}\right)}{1 - S_b/S_n}$$

Re-arrange, in terms of new candidate fabrication cost required to break even on material change:

$\left(1 - S_b/S_n\right)\left(C_w\right) + \left(C_{mb} + C_{fb}\right) = \left(S_b/S_n\right)\left(C_{mn} + C_{fn}\right)$. Finally, the required fabrication cost is:

$$C_{fn} = \frac{\left(1 - S_b/S_n\right)\left(C_w\right) + \left(C_{mb} + C_{fb}\right)}{S_b/S_n} - C_{mn}$$

From which the required installation cost is: $C_{in} = C_{fn} + C_{mn}$

The maximum breakeven fabrication and installation costs for material/concepts used as tension members, shear panels, simple columns and wide columns and compression flanges are calculated in Tables VII and VIII. In the case of wide columns, non-optimum factors due to practical stringer spacing and joint reinforcement are accounted for in calculating breakeven costs (ref. Table VIII).

* Mat'l = Materials; Fab' = Fabrication

TABLE VII

BREAK-EVEN VS ACTUAL FABRICATION & INSTALLATION COSTS

MATERIAL	C_{mn}	S_n	$\frac{Sb}{Sn}$	BREAK-EVEN C_{fn} FABR.	BREAK-EVEN C_{in} INSTL.	ACTUAL C_{fn} FABR.	ACTUAL C_{in} INSTL.	FEASIBILITY
	(1)	(2)		(3)(4)	(3)(4)	(5)		BRKEVN ≥ ACT.
SHEAR PANELS								
Baseline Material = 2024-T3 Clad, $S_b = \sqrt[3]{Ec/w} = 22$								
$C_{ib} = C_{mb} + C_{fb} = 0.66 + 5.90 = 6.56$								
AZ31B-H24	1.10	29	.74	8.48	9.58	5.90	7.00	Yes
Graphite (±45°)	(1.00)	39	.56	12.30	13.30	8.85	9.85	Yes
Mahogany/Poplar Plywood	2.05	48	.45	14.95	17.00	11.80	13.85	Yes
1" S-Glass/Epoxy	(2.00)	33	.65	9.20	11.20	5.90	7.90	Yes
3/8" E-Glass/Nylon	(0.65)	23	.96	6.27	6.92	5.90	6.55	Yes
S-Glass (±45°)	(2.00)	22	.98	4.74	6.74	8.85	10.85	No
TENSION MEMBERS								
Baseline Material = 2014-T6 Extr., $S_b = F_{tu}/w = 590$								
$C_{ib} = C_{mb} + C_{fb} = 0.97 + 5.90 = 6.87$								
MG Yttrium-T5	(6.00)	820	.72	3.90	9.90	5.90	11.90	No
Graphite (0°)	(1.00)	1870	.32	23.80	24.80	8.85	9.85	Yes
S-Glass (0°)	(2.00)	2880	.20	38.80	40.80	8.85	10.85	Yes
1" S-Glass/Epoxy	(2.00)	750	.79	6.84	8.84	5.90	7.90	Yes
Sitka Spruce	0.67	626	.94	6.45	7.12	11.80	12.47	No
Spruce-Staypak	(1.34)	760	.78	7.66	9.00	11.80	13.14	No
ZK60A-T5	3.06	682	.86	4.89	7.95	5.90	8.96	No
COMPRESSION FLANGES								
Baseline Material = 6061-T6, $S_b = \sqrt{F_{cy}\, Ec/w} = 599$								
$C_{ib} = C_{mb} + C_{fb} = 0.44 + 5.90 = 6.34$								
2014-T6 Extr.	0.97	760	.79	7.60	8.57	5.90	6.87	Yes
1" S-Glass/Epoxy	(2.00)	1160	.52	12.00	14.00	5.90	7.90	Yes
MG Yttrium T-5	(6.00)	852	.70	3.90	9.90	5.90	11.90	No
Graphite (6)	(1.00)	1350	.44	15.95	16.95	8.85	9.85	Yes
S-Glass (6)	(2.00)	955	.63	9.25	11.25	8.85	10.85	Yes

(1) ( ) indicates 1982 estimate
(2) Ref. pp. 21 and 27
(3) C_w = \$2.00/Lb. ref p. 46
(4) Formulas on p. 47
(5) Ref. p. 50 & estimated
(6) ±45°, 0°, 0°, 0° layers

TABLE VIII

BREAK-EVEN VS ACTUAL FABRICATION AND INSTALLATION COSTS WITH NET SAVINGS FOR FEASIBLE MATERIALS

MATERIAL / CONCEPT	C_{mn}	$\frac{W_n}{bL^2(10^{-4})}$	NON-OPTIMUM Z SPACING K_1	NON-OPTIMUM JOINTS K_2	K_n	$\frac{K_nW_n}{K_bW_b}$	BREAKEVEN C_{fn} FABR.	BREAKEVEN C_{in} INSTL.	ACTUAL C_{fn} FABR.	ACTUAL C_{in} INSTL.	FEASIBILITY	$\Delta\$_{PP}$	$\Delta\$_{OC}$	$\$_{Savings}$
	(2)	Figs. 28 & 31			K_1K_2		(3)(4)	(3)(4)	(5)		BRKEVN ≥ ACT.	(6)	(7)	(8)
Baseline Material = 2024-T4 Zee, $K_b = K_1\ (K_2) = 1.20\ (1.1) = 1.32$												+INCREASE	+ SAVINGS	
$C_{ib} = C_{mb} + C_{fb} = 1.10 + 5.90 = 7.00$														
@Nx/L = 30, $W_b/b_L2 = (10^{-4}) = 5.0$														
2024 Honeycomb	0.93	4.6	1.0	1.4	1.4	.977	6.29	7.22	19.20	20.13	No			
181 Cloth Zee (1)	1.00	3.2	1.2	1.1	1.32	.64	10.95	12.05	8.85	9.85	Yes	-1.02	0.72	1.74
181 Cloth Honeycomb	1.45	3.3	1.0	1.4	1.4	.70	9.45	10.85	19.20	20.65	No			
Graphite Zee z(1)	1.00	2.55	1.2	1.1	1.32	.51	14.65	15.65	8.85	9.85	Yes	-2.87	0.98	3.85
Graphite Honeycomb	1.45	2.12	1.0	1.4	1.4	.446	16.75	18.20	19.20	20.65	No			
Graphite Truss Core	1.20	2.2	1.0	1.4	1.4	.467	16.10	17.30	15.00	16.20	Yes	0.90	1.06	0.16
S-Glass Zee	2.00	3.2	1.2	1.1	1.32	.64	10.05	12.05	8.85	10.85	Yes	-0.07	0.72	0.79
S-Glass Honeycomb	2.23	2.14	1.0	1.4	1.4	.457	15.47	17.70	19.20	21.43	No			
S-Glass Truss Core	2.40	3.3	1.0	1.4	1.4	.70	8.47	10.87	15.00	17.40	No			
@Nx/L = 400, $W_b/b_L2\ (10^{-4}) = 9.8$														
2024 Honeycomb	0.93	11.0	1.0	1.4	1.4	1.19					No			
181 Cloth Zee	1.00	7.6	1.2	1.1	1.32	.78	8.55	9.55	8.85	9.85	No			
181 Cloth Honeycomb	1.45	8.2	1.0	1.4	1.4	.89	6.67	8.12	19.20	20.65	No			
Graphite Zee (1)	1.00	3.8	1.2	1.1	1.32	.38	20.70	21.70	8.85	9.85	Yes	-4.76	1.24	6.00
Graphite Honeycomb	1.45	10.0	1.0	1.4	1.4	1.08					No			
Graphite Truss Core	1.20	3.6	1.0	1.4	1.4	.39	19.80	21.00	15.00	16.20	Yes	-0.99	1.22	2.21
S-Glass Zee	2.00	5.5	1.2	1.1	1.32	.56	12.10	14.10	8.85	10.85	Yes	-1.34	0.88	2.22
S-Glass Honeycomb	2.23	6.4	1.0	1.4	1.4	.69	8.87	11.10	19.20	21.43	No			
S-Glass Truss Core	2.40	6.4	1.0	1.4	1.4	.69	8.70	11.10	15.00	17.40	No			

NOTES

(1) E-Glass

(2) Based on Values from Tables II and III with some modification where core materials are concerned.

(3) C_w = $2.00/Lb. Ref. Pg. 50

(4) Formulas on Pg. 47

(5) Ref. Pg 50 & Estimated.

(6) $\Delta\$_{PP}$ = Change in Purchase Price/Lb. of Baseline wt. of component

(7) $\Delta\$_{OC}$ = Change in Operating Cost/Lb. of Baseline wt. of component Ref. Pg. 50

(8) $Savings = Net Dollars Saved/Lb. of Baseline wt. of component

Material/Concept Feasibility

The feasibility of the various material/concepts is evaluated by comparisons of the maximum allowable break-even fabrication costs with the actual fabrication costs.

The actual fabrication costs are as follows:

Material/Concept	C_{fn} (\$/Lb.)
Truss Core	15.00
Honeycomb sandwich	19.20
Aluminum zee stringer	5.90
Reinforced plastic zee stringers	8.85
Wood construction	11.80

Tables VII and VIII also compare the break-even fabrication costs with the actual fabrication costs for the various types of members.

In the final analysis those material/concepts deemed feasible, are reviewed from the standpoint of change in purchase price of airplane, change in operating costs over 20 years (6667 hr.) period, and the net overall savings realized.

The wide column concept for two different structural index levels is shown as an example in Table VIII. The change in purchase price of the airplane is determined as follows:

$$\Delta\$_{PP} = \left(W_n\, C_{in} - W_b\, C_{ib}\right)\left(K_p\right)\left(K_d\right), \text{ where:}$$

$$\left.\begin{matrix} W_n,\ C_{in}, \\ W_b,\ C_{ib}, \end{matrix}\right\}\text{ defined on p. 47}$$

$K_p = 1.10$ (based on 10% profit)

$K_d = 1.33$ (based on 33% markup for) (distributor and dealer)

Therefore:

$$\Delta\$_{PP} = W_b\left[\left(S_b/S_n\right)\left(C_{in}\right) - C_{ib}\right]K_p K_d$$

$K_p K_d = 1.10 \times 1.33 = 1.46$

$W_n = W_b\,(S_b/S_n)$ (ref. p. 47)

The change in operating costs over 20 years (6667 hrs.) is based on the worth of a pound of material saved being equal to C_w=\$2.00 (ref. p. 46).

$$\text{Therefore: } \Delta\$_{OC} = \left(W_b - W_n\right)\left(C_w\right) = W_b\left(1 - S_b/S_n\right)\left(C_w\right)$$

The net overall savings realized is equal to:

$$\$_{savings} = \$_s = \Delta\$_{OC} - \Delta\$_{PP}$$

APPLICATION OF MATERIALS AND CONCEPTS

In this section, several appropriate and previously listed potential materials will be applied to a conceptual, but typical, light airplane. The airplane illustrated in Figure 41, is a single-engine, four-place configuration meeting the contract guidelines of this study and is referred to herein as the "Far Term Airplane". The guidelines for this airplane are listed in Table IX.

TABLE IX

FAR TERM AIRPLANE GUIDELINES

Accommodations		Performance	
Passengers and crew	4	Endurance	4 hrs. + 30 minutes
Baggage	200 lbs.	$V_{maximum}$	152 knots @ S.L.
Cabin volume	112 ft^3	V_{cruise}	130 knots @ 5000 ft.
		V_{stall}	48 knots @ S.L.
Propulsion		Takeoff distance/50 ft.	1000 ft.
Maximum power	250 hp	Minimum rate of climb	1000 ft. per minute
Maximum weight	380 lbs.	Service ceiling	14,000 ft.

These same material selections and applications would be applicable for other airplanes of similar structural loading magnitudes and manufacturing quantities; but the light airplane designer is not restricted to these same selections. The following discussions will make apparent the inter-relationship of such considerations as performance and configuration specifications, weight, cost, production rate, and manufacturing method.

Configuration Determination

Table X lists the dimensional specifications of the airplane which satisfies the contract guidelines in Table IX.

Certain major parameters describing the configuration were determined by an optimization technique developed for the study. These were the wing loading, power loading and gross weight, and hence wing area and installed power.

The wing has a tapered planform with no sweep at the quarter-chord. The aspect ratio of seven, typical of most current four-place light airplane wings, has evolved as the optimum trade-off between weight, structural integrity, and performance. The 63 series airfoil wing provides an appreciable amount of

THREE-VIEW OF FAR TERM LIGHT AIRPLANE

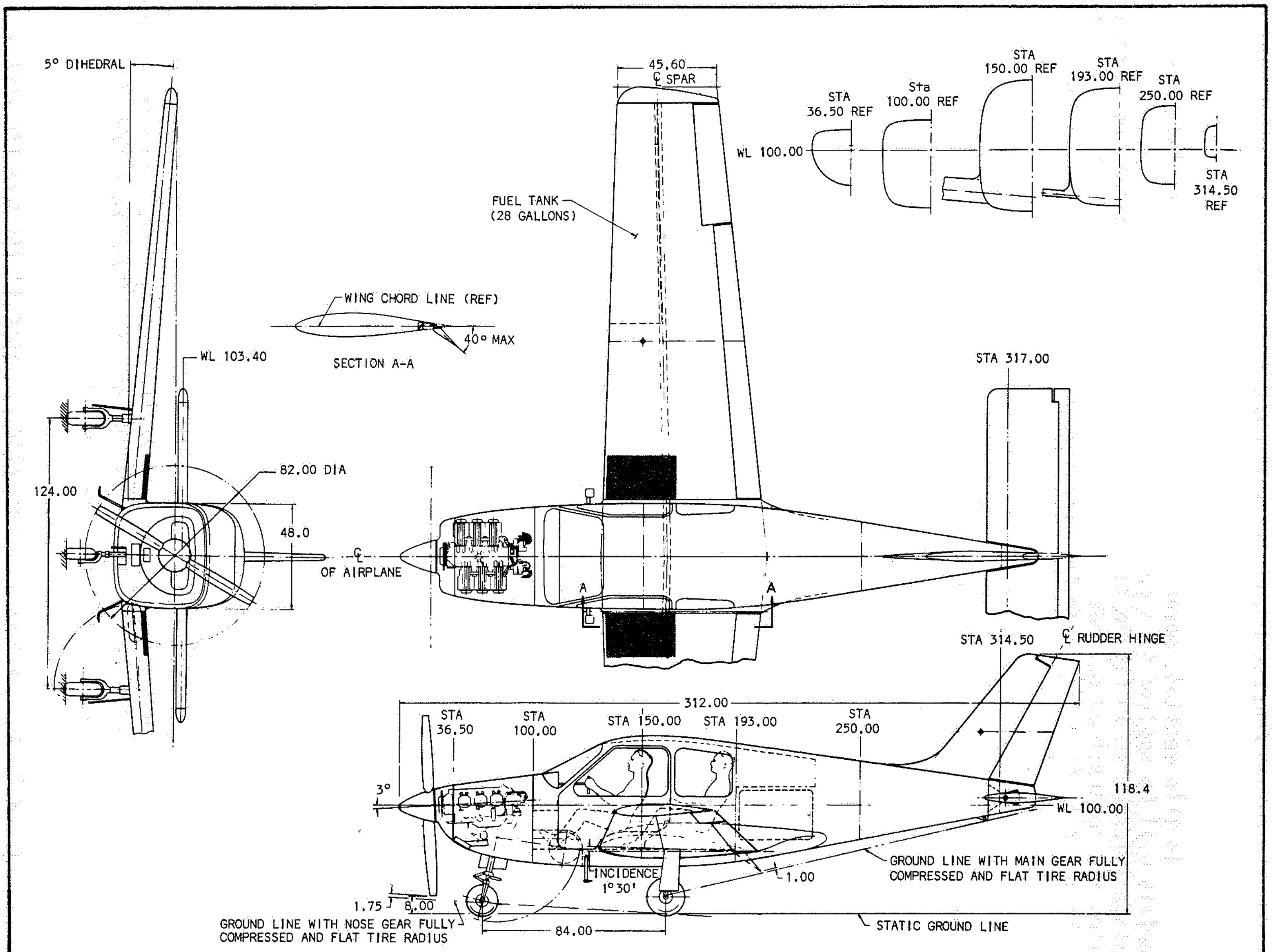


Figure 41

TABLE X

FAR TERM AIRPLANE SPECIFICATIONS

Gross weight	(W)	2977 lbs.	Vertical tail		
Power	(P)	250 BHP	Area	(S)	18.25 $ft.^2$
Wing			Height	(b)	5.06 ft.
Area	(S)	180 $ft.^2$	Aspect ratio	(AR)	1.4
Span	(b)	35.5 ft.	Taper ratio	(λ)	.5
Aspect ratio	(AR)	7.0	Root chord	(c_r)	57.5 in.
Taper ratio	(λ)	.6	Tip chord	(c_t)	29.0 in.
Root chord	(c_r)	6.338 ft.	Mean aerodyn.chord	(MAC)	54.7 in.
Tip chord	(c_t)	3.803 ft.	Sweep	(Λ)	35^o
Mean aerodyn.chord	(MAC)	5.173 ft.	Airfoil		NACA 0009
Sweep @ c/4	(Λ)	0^o			
Dihedral	(Γ)	5^o	Horizontal tail		
Airfoil		NACA 63_2-A215	Area	(S)	40 $ft.^2$
			Span	(b)	12.65 ft.
			Aspect ratio	(AR)	4.0
			Taper ratio	(λ)	1.0
			Chord (constant	(c)	3.16 ft.
Center of gravity travel		10%–30% MAC	Sweep	(Λ)	0^o
			Airfoil		NACA 0012

laminar flow if care is taken in manufacturing a smooth upper surface back to the main spar. Increasing the leading edge radius by about 20% prevents leading edge stall at high lift coefficients. A 70% of span, 25% of chord*, double slotted flap with fixed vane will provide a maximum lift coefficient of 2.3 for the wing. Maximum extension angle of the flaps is 40^o. The ailerons are 25% of chord and 30% of span. They are similar to a plain sealed flap and are continuously piano-hinged on the upper skin. Aileron movement is 25^o up and $12\frac{1}{2}^o$ down. A tapered wing (λ = .6) was selected because of its low weight, structural efficiency and slightly lower induced drag. Tapering also allows greater thickness near the root for gear retraction.

The wing has a single spar located at 40% of the chord, which is approximately the thickest portion of the airfoil section. The low wing was selected for crash worthiness, structural considerations and ideal main gear retraction arrangement.

The fuel is located entirely in integral wing leading edge tanks (28 gallons in each wing). The tanks will be at the outboard section of the wing as far as possible from the occupants, to reduce post-crash fire hazards. No fuel will be carried aft of the spar, to facilitate aileron and flap controls installation. A volume computation shows that the fuel tanks will extend approximately 100 inches inboard from the wing tips.

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*25% of chord for entire flap set, 20% of chord for main flap.

The horizontal and vertical tail areas were designed to give acceptable tail volumes for this type of airplane. The horizontal tail is an all moving stabilator used for simplicity and control effectiveness. It has an adjustable anti-servo tab to provide control feel and trim.

The cabin volume is 112 cu. ft. (excluding baggage space). Minimum width is 3.67 ft. The contract guidelines were 112 cu. ft., and 3.50 ft. minimum width. The baggage space exceeds 16 cu. ft. and is arranged to accommodate four 9" x 21" x 31" suitcases. The cabin will have an access door on each side. The baggage compartment will have an access door on the right hand side only. Part of this door will form the wing root fillet.

The retractable landing gear was decided upon because a trade off study during the performance of this contract indicated it would result in a lower direct operating cost providing the utilization exceeds 136 hours per year. It allows more efficient performance with less power at all speeds. The nose gear will retract in a conventional manner, between the front occupants. The main gear retracts inboard and after the main spar.

Material/Concept Selection

The material/concepts selected for the various airplane components are based primarily on the results of phase I of the Study, and are summarized in Tables VII and VIII. Several additional factors influenced the final structural arrangements. On the wing components, for example, single spar construction over stringer-spar construction was chosen for two reasons: (1) The airfoil components loading intensities were of such low magnitudes that little if any advantage could be gained with the stringer-spar concept: (2) Concern over the possibility that the stringer configuration would tend to create ridges in the smooth airfoil sections and thus degrade the aerodynamic characteristics.

In selecting materials for the components, primary concern was given to the wing. The importance of structural Integrity was paramount, therefore, continuous filament type composites were used for the main spar and wing skins. The continuous fiber, cross lamination configurations give optimum fracture toughness and fatigue strength because their inherent discontinuities tend to inhibit crack propagation between the filaments. A review of Tables VII and VIII indicate graphite/epoxy and S-glass/epoxy to be the most promising candidates for this structure.

The empennage, while treated as primary structure, was nevertheless considered to have slightly lower requirements from the standpoint of fatigue and fracture toughness. For these reasons non-continuous glass, with thermosetting resins were used for structure. Three non-continuous filament composites were considered in Phase I: (1): 3/8" E-glass/nylon 6/10; (2): 1/2" E-glass/polyester and (3): 1" S-glass/epoxy. The 1" S-glass/epoxy is the most efficient strengthwise, and will be used in the design of the horizontal tail. It is a compression moldable material. The 1/2" E-glass/polyester material although not always more efficent than the 3/8" E-glass/nylon 6/10 exhibited higher stiffness characteristics and resistance to environmental conditions. It is also a compression moldable material and will be used for the design of the vertical tail.

The fuselage utilized both types of composites. The longerons and other moment reacting members were made with the continuous filament S-glass/epoxy material while the low load intensity fuselage shear panels incorporated non-continuous 1" S-glass/epoxy moldable material.

Material/concepts involving aluminum alloys were not incorporated in the fabrication of the main components. A review of the phase I indicated the most promising composites exhibited superior structural efficiencies. In addition, the moldable reinforced plastics showed greater potential over the aluminum, from the standpoint of mass production processes which would offer greater fabrication cost savings.

Component Design

This sub-section will discuss the design of the vertical tail, horizontal tail, wing, and the fuselage.

Vertical tail.-Based on the three-view in Figure 41, the vertical tail has a total area (exposed) of 15.84 sq. ft. The fin area is 9.18 sq. ft., and the rudder area is 6.66 sq. ft. The design concept selected was based on a compression molded reinforced thermosetting plastic (i.e., ½ "E"-glass/polyester available in the industry in .025 thick prepreg sheets). See Figures 42 and 43. An alternate material, injection molded glass/nylon, will be discussed in a later section on cost and manufacturing considerations.

The four-piece stabilizer (Figure 42) consists of a R.H. skin, a L.H. skin, a spar, and a root closing rib. Early studies of the tail were based on the assumption that a grid pattern of internal stiffeners would be required to keep the panel sizes small in order to increase shear buckling allowables, but structural analysis indicated that "chordwise only" internal stiffeners would be adequate. As shown in Figures 42 and 43, the skin and stiffeners are integral, and the hinge fittings are integral with the spars.

Part release is a basic consideration on the design of molded parts. Fortunately, a relatively small draft angle is required for plastics (1°); and possibly some short sections could be released from the mold without draft.

Due to the relatively low bearing allowables for reinforced plastics (20,000 psi), most of the bolted connections will be critical in bearing. Large diameter bolts will be required. Some weight could be saved if hollow bolts were used. Most bolt holes will be cored, so no drilling will be required after molding. A minimum of two-diameter edge distance is used for all bolts. Molded-in-place inserts will be used at all hinge lugs. Vertical loads will be

VERTICAL STABILIZER, FAR TERM LIGHT AIRPLANE

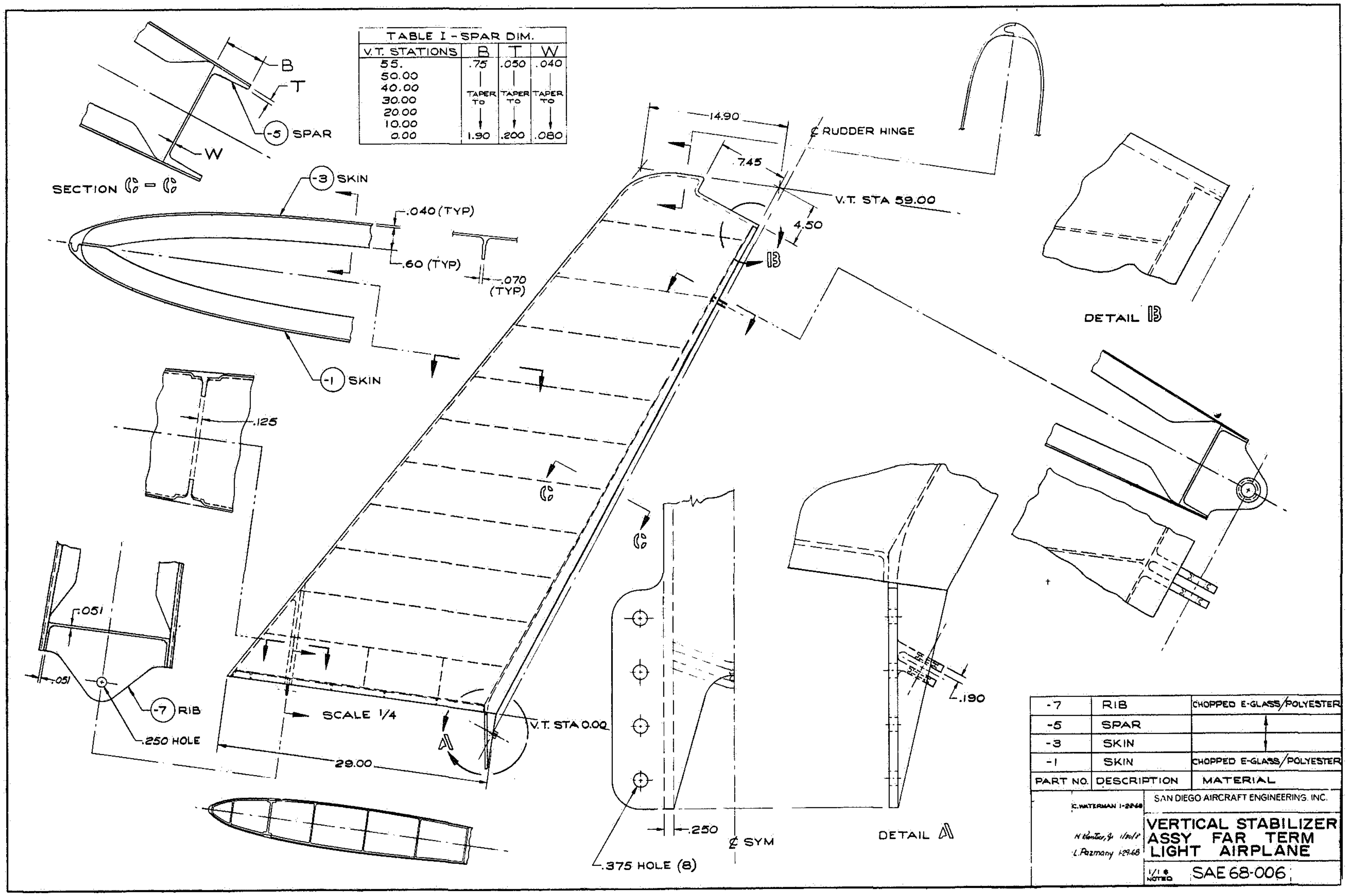


Figure 42

RUDDER, FAR TERM LIGHT AIRPLANE

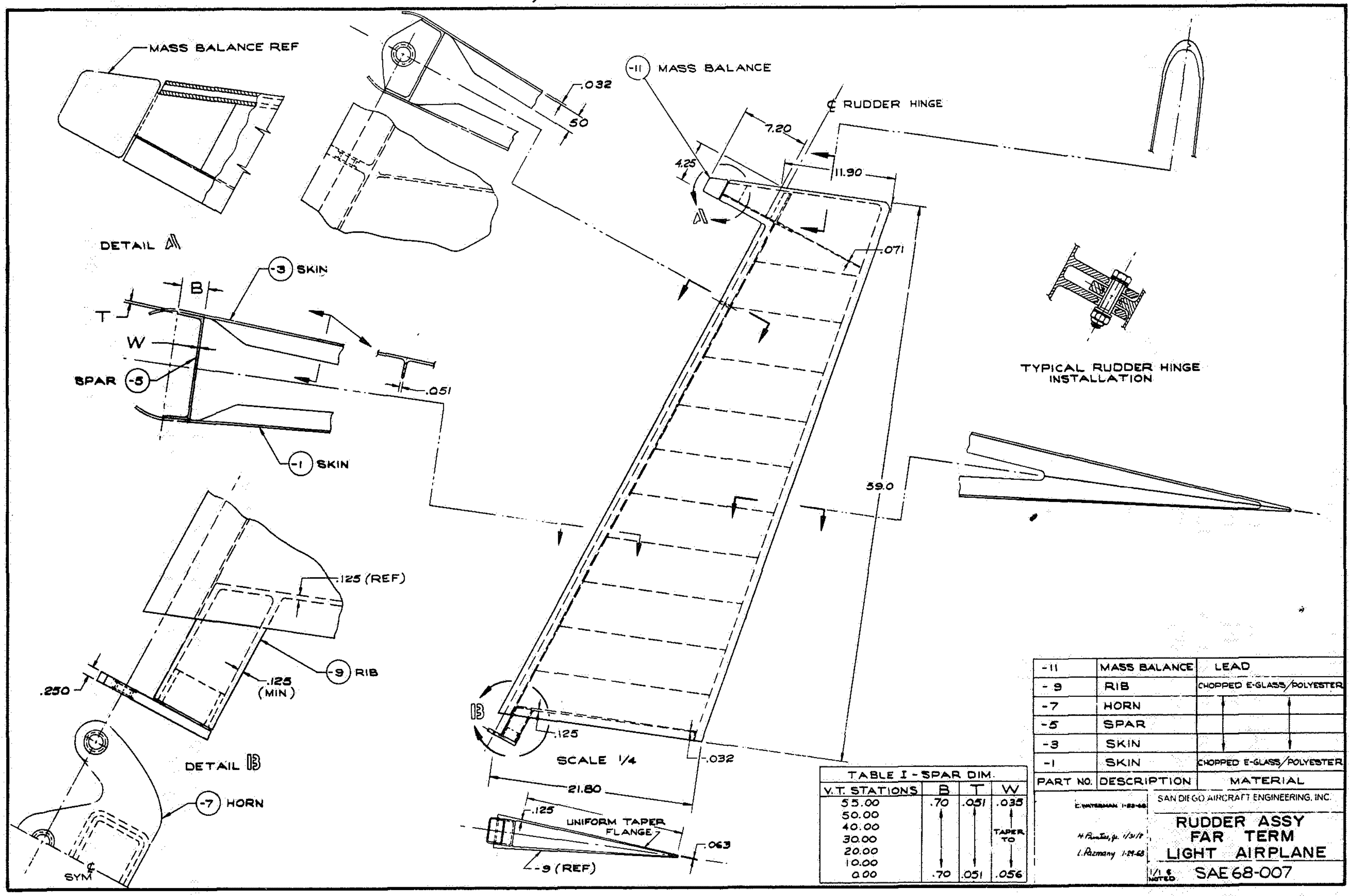


Figure 43

reacted at the bottom hinge only, which also reacts the rudder control horn loads.

The six-piece rudder is of similar construction (Figure 43) to the stabilizer or fin. The skins are reinforced with internal chordwise stiffeners, as on the fin. An attempt was made to reduce the number of parts by integrating the spar with the root rib, but part extraction, as on the fin, becomes a problem unless a complex mold is used. The lead mass balance will be bonded to the skins and upper arm. Due to the tapered shape, the mass balance is also mechanically locked in place.

The vertical fin and the rudder are entirely bonded. Adequate bonding surfaces are provided for on their respective peripheries. The leading edge of the fin has a tongue-and-groove design which insures alignment, and does not expose thin overlaps which might peel off. Also, the build up of material at the leading edge provides additional protection for erosion or hail damage. If necessary, a pressure-sensitive tape could be laid up over the leading edge.

Spar flanges, rib flanges, and skin stiffeners heights were designed considering the flow capability of the material into deep crevices. Industry sources have indicated that both glass/polyester prepreg and Nylon 6-10 will adequately fill these thin, deep grooves in the mold. An attempt was made to design the root rib and the spar in one piece, but it was found that this method resulted in locking of the part in the mold. Otherwise, the mold would have to be more complicated to permit ejection.

Horizontal tail.-The horizontal tail is a single-slab flying tail (stabilizer), hinged at the 25% point of its constant chord.

Referring to Figure 44, the structural concept for the stabilizer is based on an all-bonded construction of glass-reinforced plastic components. These components are compression molded from a prepreg 1 in. "S"-glass/epoxy composite. The stabilizer is made up of the following molded plastic components:

(1) Two each of two nearly opposite skins
(2) One main spar
(3) One trailing edge spar
(4) Two identical leading edge ribs
(5) One torque box
(6) Two identical anti-servo tab skins
(7) Two identical anti-servo tab closing channels
(8) One anti-servo tab control bracket
(9) Two identical anti-servo tab mass balance fairings

The remaining components are two identical mass-balance weights for the anti-servo tab, and the main stabilizer mass balance.

The skins are designed such that the upper right and left skins are interchangeable with the lower left and right skins, respectively. Each skin

HORIZONTAL STABILIZER, FAR TERM LIGHT AIRPLANE

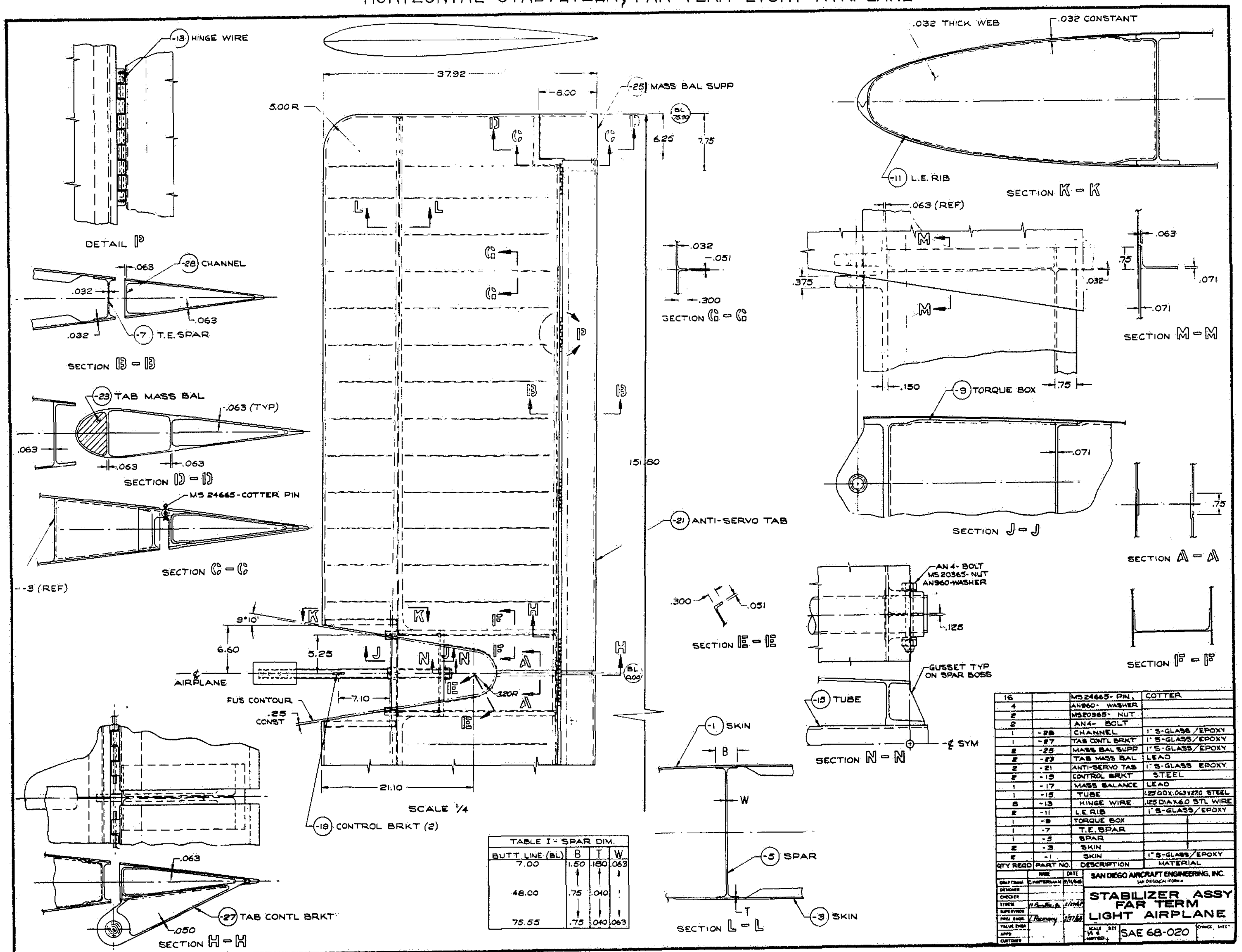


Figure 44

has integrally molded chordwise skin stiffeners and a tongue or groove in its leading and outboard edges. This wedge shaped tongue-and-groove design was recommended by a molder in preference to the full radii type specified on the vertical stabilizer. The wedge-shaped tongue-and-groove insures alignment and does not leave thin overlaps which might peel off. Also the extra material at the leading edge provides additional resistance to erosion and hail damage. If necessary, a pressure sensitive tape could be applied to the leading edge.

The main spar, molded all in one piece, has an I-beam cross section, the web thickness and height of which are constant. The cap width and thickness are tapered outboard. The upper and lower caps of the I-beam meet one another via an elliptical contour at each end of the spar. The center section of the spar caps have thin extensions which act as closures to the torque box. Also integrally molded on the spar are two sets of clevis hinge fittings and a boss with a cored hole for the main mass balance arm. Due to the low bearing allowables for reinforced plastics (20,000 psi), the clevis hinge fittings have molded-in inserts, sized for large diameter (possibly hollow) bolts (i.e., 3/8). A standard minimum edge distance of two diameters is specified for clevis hinge fitting holes. Each clevis hinge fitting is designed to mate with a set of three lugs on the fuselage.

The torque box consists of a pair of identical ribs, integrally connected with a channel. This so-called torque box becomes a true torque box when it is mated and bonded to the spar, between the spar cap extensions. These extensions are bonded to the ribs and to the interconnecting channel on the so-called torque box. Considerable effort was expended to eliminate load path discontinuities and to maintain efficient bonding joints. The interconnecting channel on the so-called torque box has a boss with a cored hole, which aligns with a similar hole in the spar web. These respective holes support the main stabilator mass balance arm. Manufacturing considerations and cost analyses of this part will likely dictate breaking this part into two separate (but identical) ribs and a shallow box with a hole in it. As it is now, it will require two massive cores normal to the direction of mold pressure application.

The trailing edge spar is molded full span in one piece, with eight sets of five-lug piano hinges molded integrally into its otherwise constant I-beam cross section. This I-beam cross section is closed on both ends to provide a continuous bonding interface with the skins. The aft ends of the torque box ribs nest into the front side of the trailing edge spar.

Mating with the eight sets of piano hinges on the aft spar is an anti-servo tab. The inboard end of the right hand and left hand portions of each tab is mated to one of the two male extensions on a single anti-servo tab control bracket. The lever arm on this control bracket has a No. 10(3/16 I.D.) insert integrally molded in. The right hand and left hand portions of the anti-servo tab each consist of a skin, a closing channel and a mass balance fairing, which are respectively interchangeable, one side for another. Each identical one-piece skin has a constant deep "V" cross section. Each identical closing channel has a constant cross section except for four sets of five-lug piano hinges, which mate with those on the trailing edge spar. An attempt was made to make the anti-servo tab a one-piece extrusion of glass/nylon (rather than a channel + skin). This was subsequently discarded due to the inadequate torsional stiffness of this material. An identical and interchangeable mass balance fairing closes off both outboard ends of the anti-servo tab. Identical lead weights are bonded into a cavity in each mass balance fairing.

A single leading edge rib is nested and bonded to the forward side of the main spar, immediately outboard of each clevis fitting on the spar. These two ribs are identical.

Table XI tabulates weights and unit weights for the primary empennage components. Table XII lists the comparable data for a conventional sheet metal construction empenage.

TABLE XI
FAR TERM LIGHT AIRPLANE EMPENNAGE WEIGHTS

		VERT. FIN	RUDDER	STAB.
	Area (ft^2)	9.18	6.66	40.00
Injection molding Nylon 6/10 (.051 lb./cu. in.)	Weight (lbs.) Unit weight $(\frac{lbs}{ft^2})$	14.44 1.58	9.35 1.4	NA
Compression molding Chopped E-glass/ polyester (.070 lb./cu. in.)	Weight Unit weight	13.13 1.43	8.5 1.28	NA
Compression molding 1" S-glass/epoxy (.062 lb./cu.in.)	Weight Unit weight	11.63 1.27	7.5 1.13	36.06 0.90

TABLE XII
CONVENTIONAL SHEET METAL EMPENAGE WEIGHTS

		VERT. FIN	RUDDER	STAB
Aluminum Sheet metal (Contemporary light airplane)	Area (ft^2) Weight (lbs) Unit weight $(\frac{lbs}{ft^2})$	7.45 11.00 1.47	4.11 4.5 1.1	24.4 26.0 1.07

Wing.- The wing has outboard leading edge wet fuel tanks and the main landing gear, mounted aft of the single spar, retracts inboard and slightly aft. See Figures 45 and 46. The wing has a single spar located at the 40% chord. It has an open-side-aft channel cross-section. The channel's height, cap thickness, and web thickness taper outboard, and the cap width remains constant. The spar is compression molded from high modulus graphite filament reinforced epoxy prepreg. The spar web is made up of prepreg epoxy/graphite tapes in a multi-layer, multi-direction pattern. The spar caps are also made up of the same (or similar) epoxy/graphite tapes, with 72% of the graphite running spanwise and the remainder at $\pm 45^0$. There will be a comparable constructed auxiliary spar between the main landing gear support rib and the root rib for reacting a part of the main landing gear loads. The main landing gear support fittings will be glass-reinforced plastic with metal bushings for bearing loads. One is mounted between the main spar and the above-mentioned auxiliary spar on each wing half. See Figure 47.

Each wing half is attached to the fuselage with two bolts through the spar web and into a fuselage frame, and one bolt each at the front and rear of the wing root closing rib. The main spar on each wing half,extends to the fuselage centerline, at which point they are joined by 18 to 20-in. splice plates nested to the outside and inside surfaces of each spar cap, and by a 4-in. wide splice plate on each side of the web.

The two aft closing members on each wing half are: a zee-section along the aileron interface and a channel along the flap interface (See Figure 46). Each wing half has five sets of ribs (leading edge + aft), plus two additional leading edge ribs. They are located at: (1) the root (see Figures 45 and 46,section D-D); (2) the landing gear interface (see Figures 45 and 47, section C-C); (3) the inboard end of the fuel tank at WS 105.6 (wet bulkhead); (4) midway in the fuel tank, or between the aileron and flap; and (5) the tip (see Figures 45 and 46, section A-A), which is a wet bulhead. The two additional leading edge ribs quarter the fuel tank. The first four ribs also provide integral hinge supports for the flap (see Figure 46, section D-D).

The skin consists of three details for each wing half (i.e., a leading edge skin from the top spar cap to the bottom spar cap, an upper aft skin, and a lower aft skin, each of which extend from root to tip). All of these skins are made from compression molded multi-directional graphite/epoxy prepreg tapes and have no integral stiffeners. The initial wing design specified separate "T"-section chordwise skin stiffeners, which will be bonded to the inner skin surfaces.

The aileron on each wing half consists of an upper and lower integrally stiffened skin. The skin stiffeners are spaced five inches apart for a total of 18 per aileron. The forward closing web is integral with the upper skin, as are the piano-hinge lugs. See Figure 46, section K-K. There is a closing rib at each end of the aileron. The aileron would be mass balanced at the outboard end with the weight traveling up and down within the wing tip fairing.

WING PLANFORM, FAR TERM LIGHT AIRPLANE

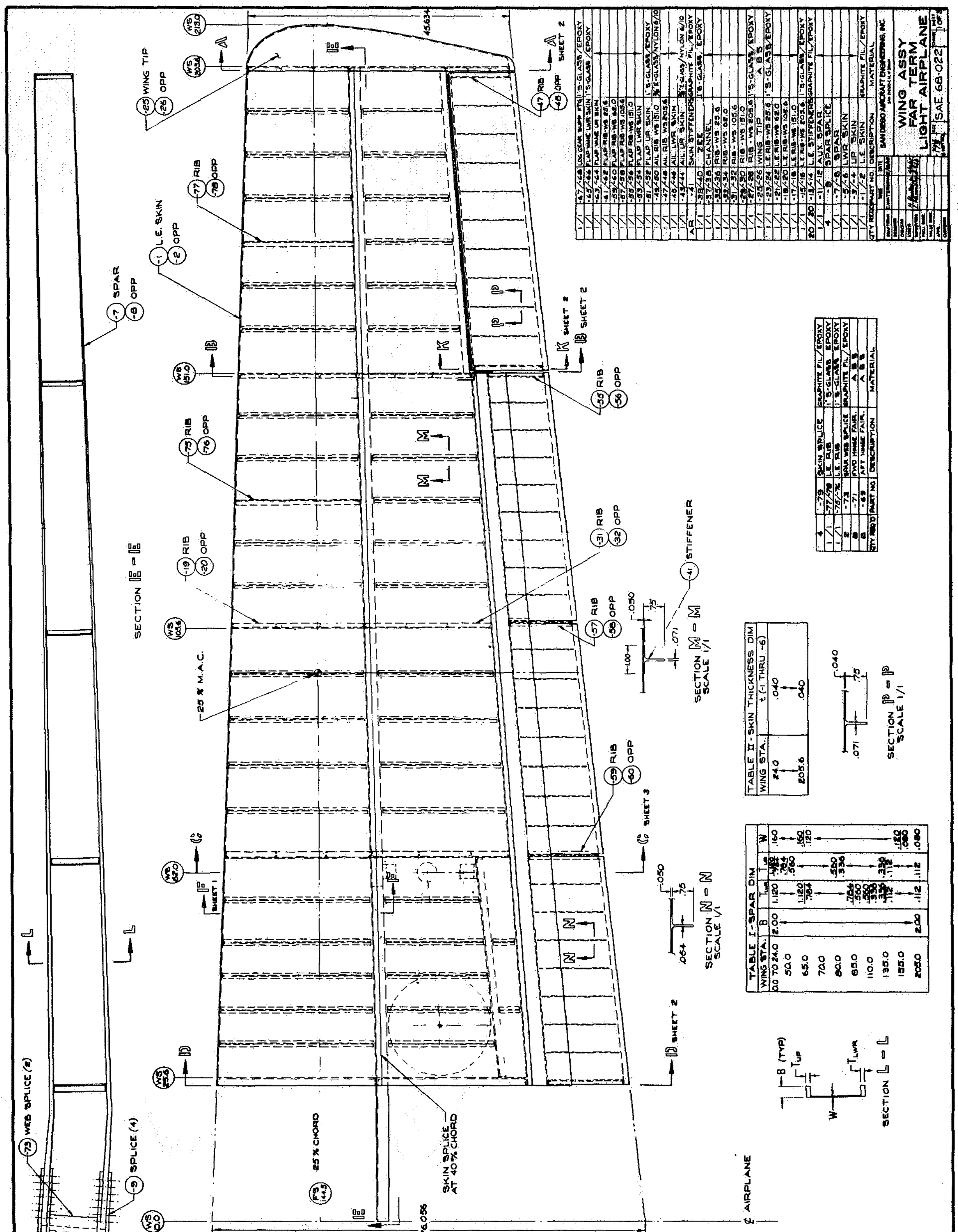


Figure 45

WING SECTIONS, FAR TERM LIGHT AIRPLANE

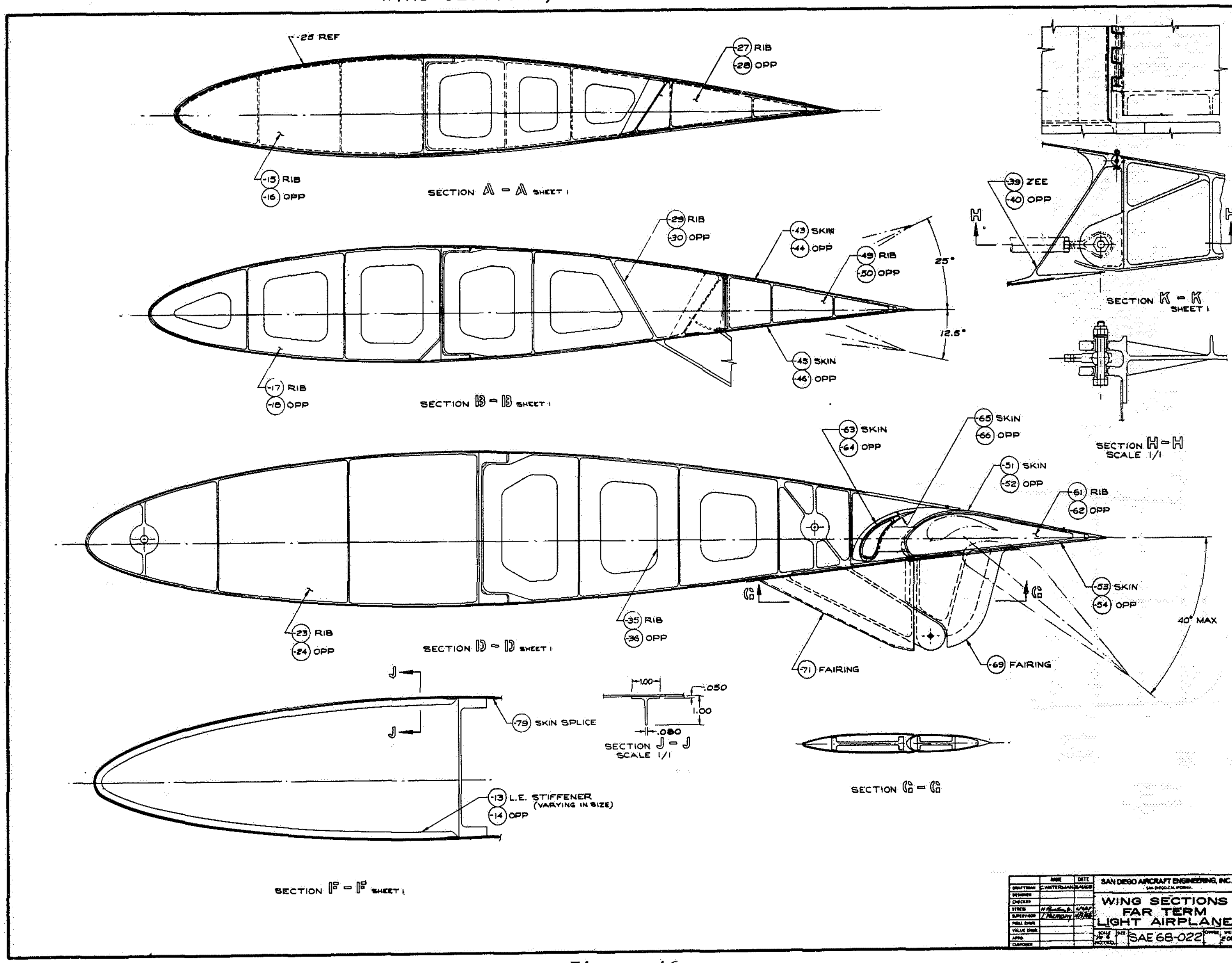


Figure 46

WING/LANDING GEAR INTERFACE, FAR TERM LIGHT AIRPLANE

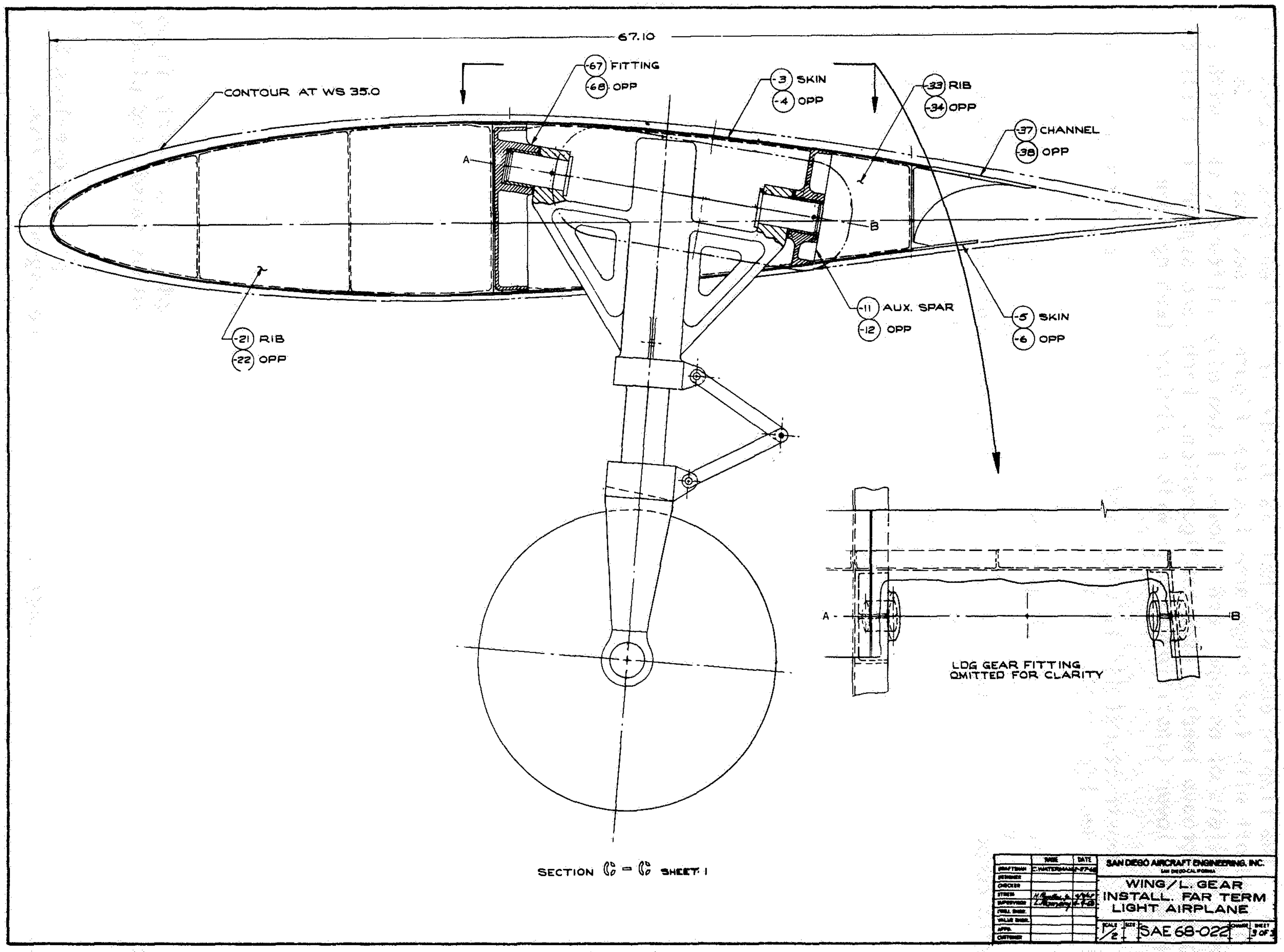


Figure 47

The flap on each wing half is divided into three segments connected and closed off with four hinge arm-ribs (see Figure 46, section D-D). Each flap segment consists of an upper and a lower, integrally stiffened, skin with a tongue-and-groove leading edge configuration. Each vane segment consists of an upper and a lower (unstiffened) skin with a similar leading edge joint.

Nonstructural tip fairings and hinge fairings are of hot-formed thermoplastic. See Figures 45 and 46.

Referring again to Figures 45, 46, and 47, the material selection and the type of molding considered for each of the 202 machine molded, reinforced plastic components are as follows: The spars, spar splices and skins (-7, -11, -9, -73, -1, -3, & -5) are made of compression molded high modulus graphite filament/epoxy; the ailerons (-43 thru -49) are made of injection molded E-glass/nylon; the tip fairings (-25) and the flap hinge fairings (-69 & -71) are made of hot-formed ABS; and the remainder of the components are made of compression molded S-glass/epoxy.

All of the above components are then appropriately prepared for bonding, fixtured and secondary bonded to form a right hand and a left hand wing half; which are subsequently attached to one another and to the fuselage with mechanical fasteners.

Two alternate wing construction concepts (designated II and III) were considered as possible weight and/or cost savers. Referring to Figure 48, Configuration II replaces the graphite channel section spar with an S-glass rectangular rigid urethane (foam core) section. Also, the graphite skins are replaced with S-glass skins. The resultant weight saving in the spar is exceeded by the weight penalty in the skins. See Table XIII. Configuration III is the same as II, except graphite is used in place of the S-glass. This concept (i.e. III) amounts to a 10% saving in total wing weight and, as will be discussed later, a 5% saving in wing cost. Both graphite wing construction concepts represent significant weight (and cost) savings over conventional sheet aluminum construction, (if the cost of graphite can be reduced to $1.00 or $2.00 per pound).

<u>Fuselage</u>.- The fuselage is conventional in size and shape. The overall dimensions include a maximum width of 48 inches, maximum height of 60 inches, and a length of 232.5 inches (firewall station 100.00 to aft tip of stringer fairing). There are two passenger doors, one baggage compartment door, two side windows, and a one-piece windshield. The fuselage design is only preliminary since neither loads nor stress analyses have been performed to size the various components.

Referring to Figure 49, the structural concept for the thirty-three piece fuselage is based on all bonded construction of glass reinforced plastic components. The firewall is stainless steel.

Skins and frames are compression molded from a prepreg one-inch S-glass/epoxy composite. The longerons and channels are bag-molded from

TABLE XIII
WING WEIGHTS (POUNDS)

ITEM	FAR TERM LIGHT AIRPLANE			CONTEMPORARY AIRPLANE
	CONFIG. I	CONFIG. II	CONFIG. III	ALUMINUM WING
	Graphite Constr.	S-Glass Constr. Foam Core Spar	Graphite Constr. Foam Core Spar	
Skins	77.3	110.6	77.3	108.0
Spars	92.6	82.6	65.1	85.0
Ribs	26.9	26.9	26.9	26.0
Stringers				7.0
Skin stiffener	16.5	16.5	16.5	
Skin splices	8.3	11.9	8.3	
Tip	1.5	1.5	1.5	1.5
Total	222.3	250.0	195.6	227.5

WING SPAR CONFIGURATIONS

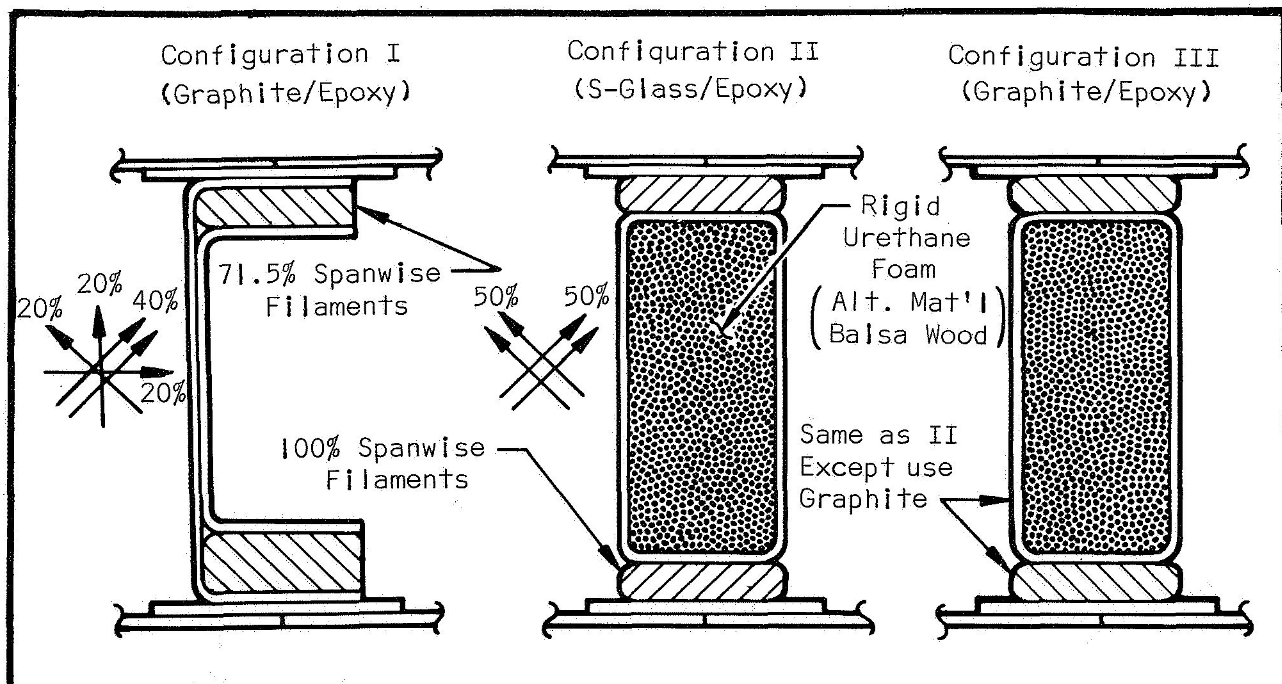
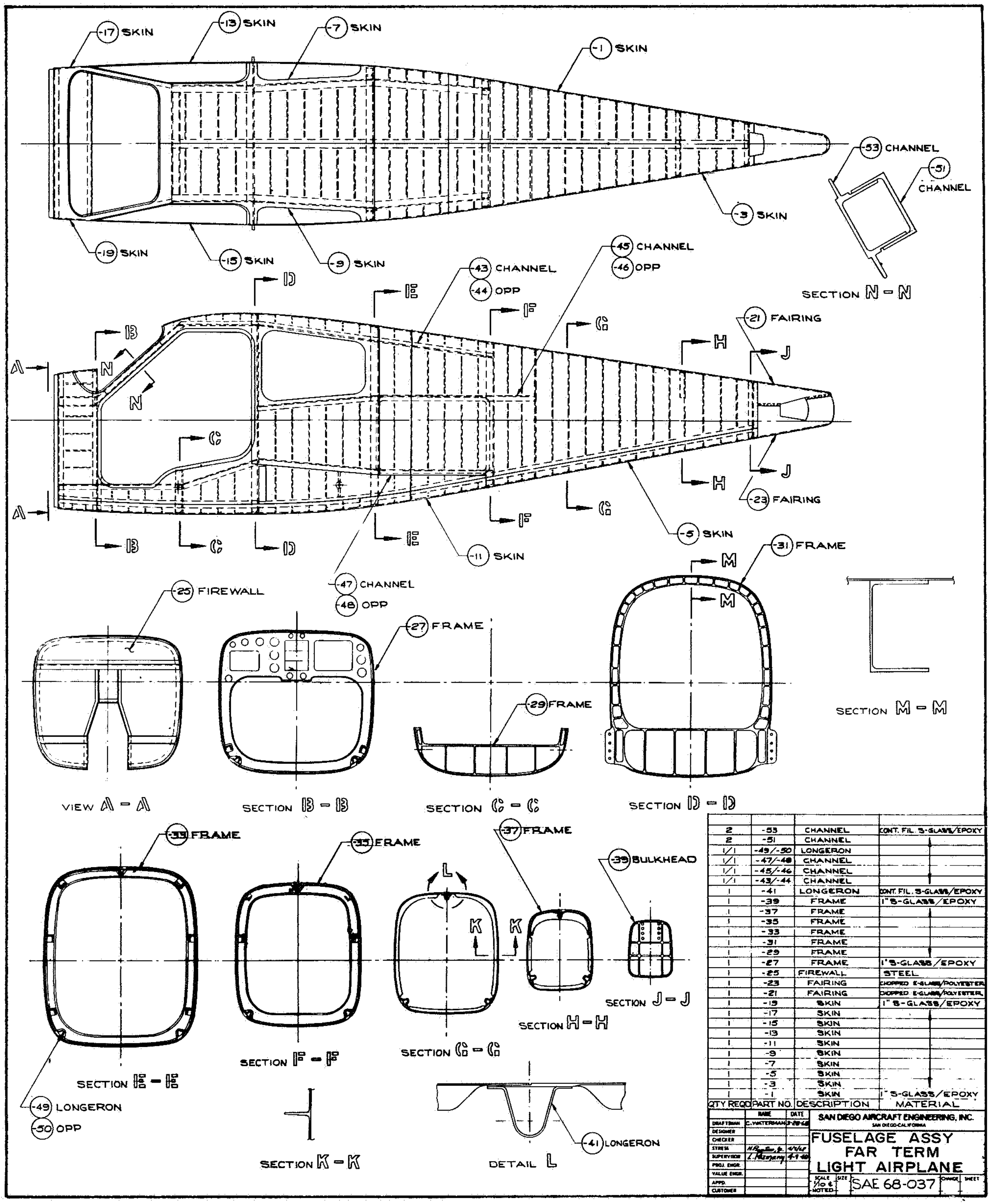


Figure 48



QTY REQD	PART NO.	DESCRIPTION	MATERIAL
2	-53	CHANNEL	CONT. FIL. S-GLASS/EPOXY
2	-51	CHANNEL	
1/1	-49/-50	LONGERON	
1/1	-47/-48	CHANNEL	
1/1	-45/-46	CHANNEL	
1/1	-43/-44	CHANNEL	
1	-41	LONGERON	CONT. FIL. S-GLASS/EPOXY
1	-39	FRAME	1" S-GLASS/EPOXY
1	-37	FRAME	
1	-35	FRAME	
1	-33	FRAME	
1	-31	FRAME	
1	-29	FRAME	
1	-27	FRAME	1" S-GLASS/EPOXY
1	-25	FIREWALL	STEEL
1	-23	FAIRING	CHOPPED E-GLASS/POLYESTER
1	-21	FAIRING	CHOPPED E-GLASS/POLYESTER
1	-19	SKIN	1" S-GLASS/EPOXY
1	-17	SKIN	
1	-15	SKIN	
1	-13	SKIN	
1	-11	SKIN	
1	-9	SKIN	
1	-7	SKIN	
1	-5	SKIN	
1	-3	SKIN	
1	-1	SKIN	1" S-GLASS/EPOXY

Figure 49

continuous-filament S-glass/epoxy prepreg tapes. The stringer fairings are molded from one-inch E-glass/polyester composite prepreg.

Component Cost and Manufacturing Considerations

The cost analyses discussed in this sub-section are limited to just two of the four primary structural components. These, the vertical stabilizer and the wing, are structurally the least and most demanding, respectively. In any event, these two analyses demonstrate the magnitude of the potential savings associated with machine molded/high production rate construction concepts. Manufacturing considerations for all four (vertical tail, horizontal tail, wing, and fuselage) primary structural components will be discussed briefly.

Vertical tail.-The vertical stabilizer, with its minimum structural requirements, is a feasible application for both compression molded thermosetting (reinforced) plastic and injection molded (reinforced) thermoplastic.

Compression molding of prepreg sheet molding thermosetting composites, such as E-glass/polyester, is considerably slower than injection molding. It does offer, though, a good possibility of achieving the required thin skins. This is possible due to the partial distribution of prepreg material, normally preheated, in the dies before the dies are closed. This means the material has a shorter distance to travel to the die extremities. Also, the material "setting" time is slower and the material has considerably more time to flow, since it "sets" or cures by chemical reaction rather than by "freezing" as with thermoplastics.

Compression molding, using prepreg sheet molding compound, does not lend itself to mass production as well as injection molding, due to its slower "set" time, hand loading requirements and supporting activity requirements such as precutting and preheating of the sheet molding compound. It is far superior though to the normal hand lay up procedures normally associated with reinforced thermosets.

Compression molding of the vertical tail, using E-glass/polyester would require only 1000 psi (approximately) and 300°F. The precut and preheated prepreg material is loaded (presently by hand) into the heated die halves, after which the die halves are slowly mated.

A hydraulic press of at least 450-ton capacity will be required to mold each vertical stabilizer skin. Closing and opening speed should be adjustable and variable within each cycle (i.e., the press should have a high speed initial closing rate to first die mate, followed by an adjustable final closing rate). This action should be semi-automatic. Such a press is estimated to cost \$11-\$12 per hour to operate.

The dies, most probably fabricated from aluminum, are estimated to cost from \$10,000 to \$24,000. These estimates are based on todays tool fabrication costs. It would be very difficult to predict whether such costs will be higher or lower in fifteen years. Compression molding die costs are higher than injection molding die costs since many more dies are required to produce parts at an equal rate. This will become evident in the following cost consideration discussions.

Coring is not as readily achieved with compression molding as with injection molding. This is due to the possibility of very high local pressure differentials that can exist between opposite sides of a core during distribution of the more viscous resin, as the dies are closing.

TABLE XIV

INDUSTRY ESTIMATES OF VERTICAL STABILIZER TOOLING COSTS (DOLLARS)

MOLDER OR MOLD MAKER	R. H. SKIN	L. H. SKIN	SPAR	RIB	BONDING FIXTURE
<u>Injection</u>					
A	←	60,000		→	—
B	50,000	50,000	—	—	—
C	50,000	50,000	—	—	—
D	←	16,000		→	(600)
E	←	100,000		→	—
E	←	(30-36,000)		→	—
For analyses purposes, use	24,000	24,000	—	—	1500
<u>Compression</u>					
F	10-12,000	10-12,000	—	—	—
G	24,000	24,000	5000	4000	3000-(1500)
E	24,000	24,000	5000	4000	3000
H	←	65-70,000		→	15,000
J	—	—	1800	3000	—
For analyses purposes, use	24,000	24,000	5000	4000	2000

NOTES: 1) ( ) = Est. for Aluminum Tooling

2) Molders A,B,C....are located in the Los Angeles - San Diego area.

The design of vertical stabilizers constructed of the above materials differs only in that the injection molded nylon stabilizer is about 10% heavier (due to its lesser strength/stiffness) and the nylon stabilizer can be molded in two pieces rather than four, due to its superior moldability. See Figure 50.

The injection molded stabilizer can be molded as a left-hand skin/rib and a right-hand skin/spar. The compression molded stabilizer possibly could be molded into the same two components, but would more likely be molded into a separate left hand skin, right hand skin, spar, and a rib, as shown earlier in Figure 42. Earlier attempts at two-piece construction, with the bond line all in a single plane, were abandoned due to inherent structrual/weight penalties, i.e., using the tongue-and-groove joint on a split spar and split rib. Figure 50 illustrates the tongue-and-groove joint on the leading edge only.

TWO-PIECE CONCEPT VERTICAL STABILIZER

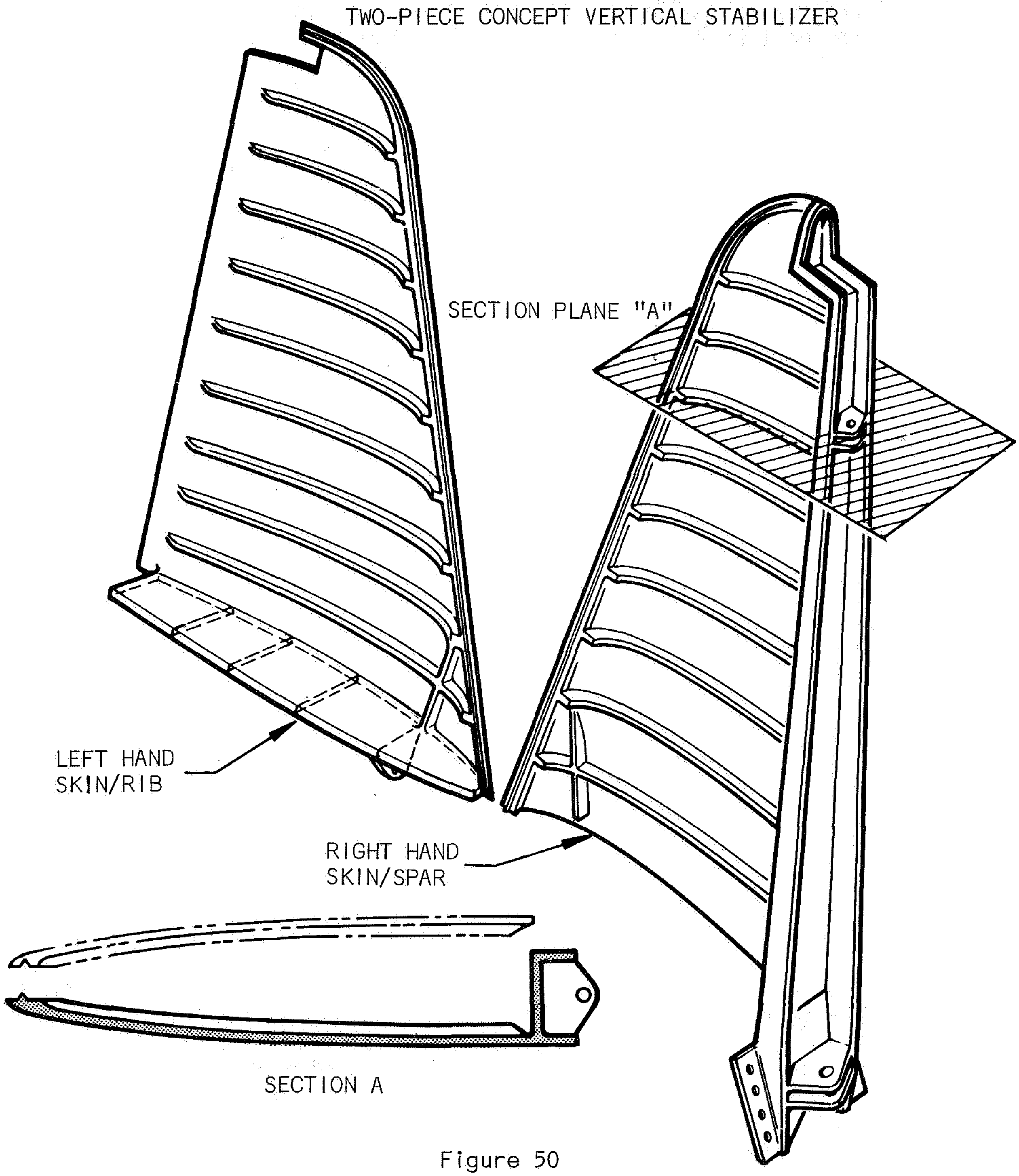


Figure 50

Section plane A, identical in nature on both components, taken from Figure 50, is illustrated in Figure 51, with the method that would be employed in molding both parts (i.e., the right hand skin/rib and the left hand skin/spar). This molding die arrangement is applicable particularly to injection molding but could possible also be applicable to compression molding. No movable cores are required, except to hold the molded in place metallic inserts in the clevis fittings on the spar. The skin/spar or skin/rib at first glance might appear to be trapped in the female mold, but it can readily be stripped from the die by using a lateral mode of extraction. In the worst case, die segment B in Figure 51 might have to be stripped from the part after the part is removed from the female die half. This two-piece concept is not at all unusual for injection molding. Die segment B would be retracted automatically as would all the other cores for the fastening and hinge fitting holes.

VERTICAL STABILIZER MOLDING DIE ARRANGEMENT
(For injection and possible compression molding)

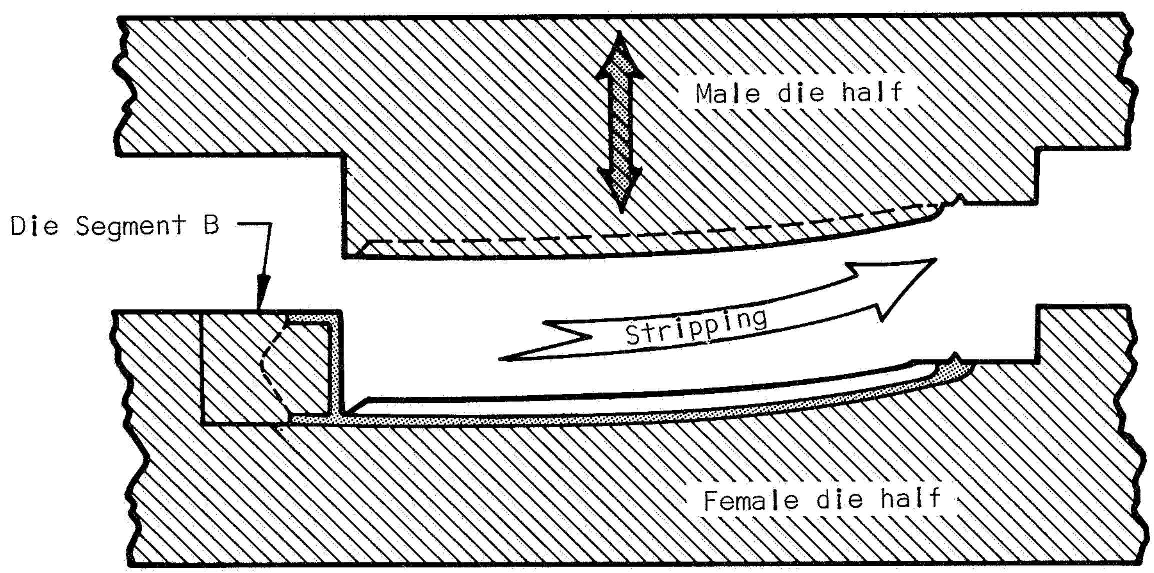


Figure 51

The rudder, of similar configuration, can also be constructed using this skin/spar + skin/rib concept.

This two-piece construction eliminates the otherwise required separate tooling costs and separate molding time costs for both the spar and the root rib. Additionally, the amount of trimming, inspection, bonding prep, actual bonding, and joint clean-up are reduced.

Additional discussions with injection molders reveal optimism concerning the feasibility of molding large thin skin components. It is quite reasonable to assume that the thin skins would be readily achievable in fifteen years, and are probably achievable today. The United States is lagging Europe and Japan, where the world's largest molding machines are built, in injection molding capability.

More and more injection molders in the United States are beginning to use aluminum dies. They are significantly cheaper and steel inserts can be used in high wear areas. Also, the higher thermal conductivity of aluminum provides for reduced cycle time, i.e., higher production rates.

An injection molded vertical stabilizer, according to molders, would require little or no clean-up after molding. The part could be submarine gated, so it would be removed from the dies, free of any gate projections. Any clean-up that would be required could be accomplished by the molding machine operator. The vertical stabilizer would be molded at a rate, conservatively estimated at 30 parts per hour, and more likely at 60 parts per hour.

Bonding of the stabilizer components, whether injection molded nylon or compression molded E-glass/polyester, would be accomplished in a fixture with provision for heating to accelerate curing of the adhesive. The surfaces to be bonded would require light sanding before application of the adhesive. Should the components be made of nylon, bonding will require considerably more attention. Nylon, and particularly glass reinforced nylon, is difficult to bond. It would require a special etch* of the surfaces to be bonded.

Table XV summarizes all the cost analyses performed on the vertical stabilizer. It is significant to note that one injection molding machine, operating two shifts per day, can produce both components of the nylon vertical tail in 100,000 units per year quantities, while it requires twenty compression molders, operating three shifts per day, to mold the four glass/polyester components in like quantities. Production rate for injection molding, estimated at 60 pieces per hour, is a liberal estimate of today's capability, but a conservative estimate of molding rates fifteen years from now. The four-per-hour production rate for compression molded glass/polyester is possibly attainable today and will surely be routine fifteen years from now. Estimates of fabrication sequences and times associated with both the injection molded and the compression molded vertical stabilizer are detailed in Table XVI.

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*e.g., calcium chloride-ethanol

TABLE XV

COST ANALYSIS TO PRODUCE 100,000 VERTICAL STABILIZERS PER YEAR

	INJECTION MOLDING	COMPRESSION MOLDING
PIECES PER ASSY	2	4
CYCLE TIME/PIECE	$\frac{1\ \text{min}}{80\%\ \text{eff}} = 1.25$ min	$\frac{15\ \text{min}}{80\%\ \text{eff}} = 18.75$ min
TOTAL TIME FOR 100,000 ASSEMBLIES	$\frac{1.25\ \text{min} \times 100{,}000\ \text{assy} \times 2\ \text{pcs/assy}}{60\ \text{min}} = 4160$ hrs	$\frac{18.75\ \text{min} \times 100{,}000\ \text{assy} \times 4\ \text{pcs/assy}}{60\ \text{min}} = 125{,}000$ hrs
FABRICATION COSTS	INJECTION MOLDING	COMPRESSION MOLDING
Raw Materials	(14.3 lbs/assy) X (.65 $/lb) X 100,000 assy = $929,500	(14.3 lbs/assy) X (.60 $/lb) X 100,000 = $858,000
Tooling Prepreg cutters Die sets Trim tools Bonding fixtures	 Not required 1 @ $48,000.00 = $48,000.00 Not required 20 @ $ 1,500.00 = $30,000.00 $78,000.00	 1 @ $ 1,500.00 = $ 1,500.00 5 @ 57,000.00 = 285,000.00 1 @ 3,000.00 = 3,000.00 20 @ 2,000.00 = 40,000.00 $329,500.00
Molding Machine charge Labor & overhead	 $\frac{\$7000}{\text{mo}} \times \frac{12\ \text{mo}}{^{*}2080\ \text{hrs} \times 3\ \text{shifts}} = \$13.50/\text{hr}$ 10.00/hr $23.50/hr 4160 hrs X $23.50/hr = $97,760	 Estimated = $ .56/hr 10.00/hr $10.56/hr 125,000 hrs X $10.56/hr = $1,320,000
Auxiliary Operations	$\frac{9.35\ \text{min}}{\text{assy}} \times \frac{100{,}000\ \text{assy}}{80\%\ \text{eff}} \times \frac{1\ \text{hr}}{60\ \text{min}} = 19{,}500$ hrs 19,500 hrs X $10/hr = $195,000	$\frac{16.23\ \text{min}}{\text{assy}} \times \frac{100{,}000\ \text{assy}}{80\%\ \text{eff}} \times \frac{1\ \text{hr}}{60\ \text{min}} = 33{,}812$ hrs 33,812 hrs X $10/hr = $338,120
SUMMARY Raw Materials Tooling Molding Auxiliary Operations	 $ 929,500 78,000 97,760 195,000 $1,300,260	 $ 858,000 329,500 1,320,000 338,120 $2,845,620
UNIT COST	$\frac{\$1{,}300{,}260}{100{,}000\ \text{assy}} = \$13.00/\text{assy}$	$\frac{\$2{,}845{,}620}{100{,}000} = \$28.45/\text{assy}$

Note: *Total available working hours per year for one shift:
8 hrs X 5 days X 52 weeks = 2080 hrs

TABLE XVI

FABRICATION SEQUENCES AND ESTIMATED TIMES
(for vertical stabilizer)

SEQUENCE		TIME (min.)
	Compression Molding	
1)	Die cut SMC* to spec. shapes	
	a) Skins (10 pcs @ 20/min)	.50
	b) Spar (10 pcs @ 20/min)	.50
	c) Rib (5 pcs @ 10/min)	.50
2)	Preheat SMC blanks	1.00
3)	Load & cure in press (part of molding charge)	-
4)	Degrease	.33
5)	Cool (no charge)	-
6)	Trim flash (4 parts @ .30 ea)	1.20
7)	Inspect "	1.20
8)	Bonding preparation (4 parts)	1.20
9)	Load all four parts in fixture	.30
10)	Apply adhesive	.60
11)	Close fixture & cure	6.45
12)	Remove from fixture	.15
13)	Inspect	1.20
14)	Dress joints	1.00
15)	Stock or convey to assembly area	.10
		16.23
*sheet molding compound		
	Injection Molding	
1)	Inspect (after molding)	.90
2)	Place skin/spar and skin/rib in bonding fixture	.10
3)	Prepare mating surfaces for bonding	.30
4)	Apply adhesive	.30
5)	Close fixture	.05
6)	Cure adhesive	6.55
7)	Open fixture & remove bonded fin	.15
8)	Dress bonded joints & inspect	1.00
		9.35

End result of the analyses indicates that the vertical stabilizer, manufactured at the rate of 100,000 units per year, can be produced at a manufacturer's cost of: (1) $13.00 when injection molded of glass/nylon 6-10, or (2) $28.45 when compression molded of glass/polyester. These costs are significantly competitive with conventional sheetmetal construction as indicated in Figure 52. Of prime significance is the indication that both injection molded and compression molded vertical stabilizers can be manufactured at a lower cost than conventional sheetmetal, even at current quantities. E.g., compare the following price-quantity relationships for the three types of construction.

	Current Quantities (i.e., 1000/Yr)	High Production Quantities (i.e., 100,000/Yr)	Production Rate Break-even Point With Sheetmetal (Units/Yr)
Sheetmetal	$110	$34	*
Compression molded	88	28	620
Injection molded	61	13	360

VERTICAL STABILIZER UNIT COST VS PRODUCTION RATE

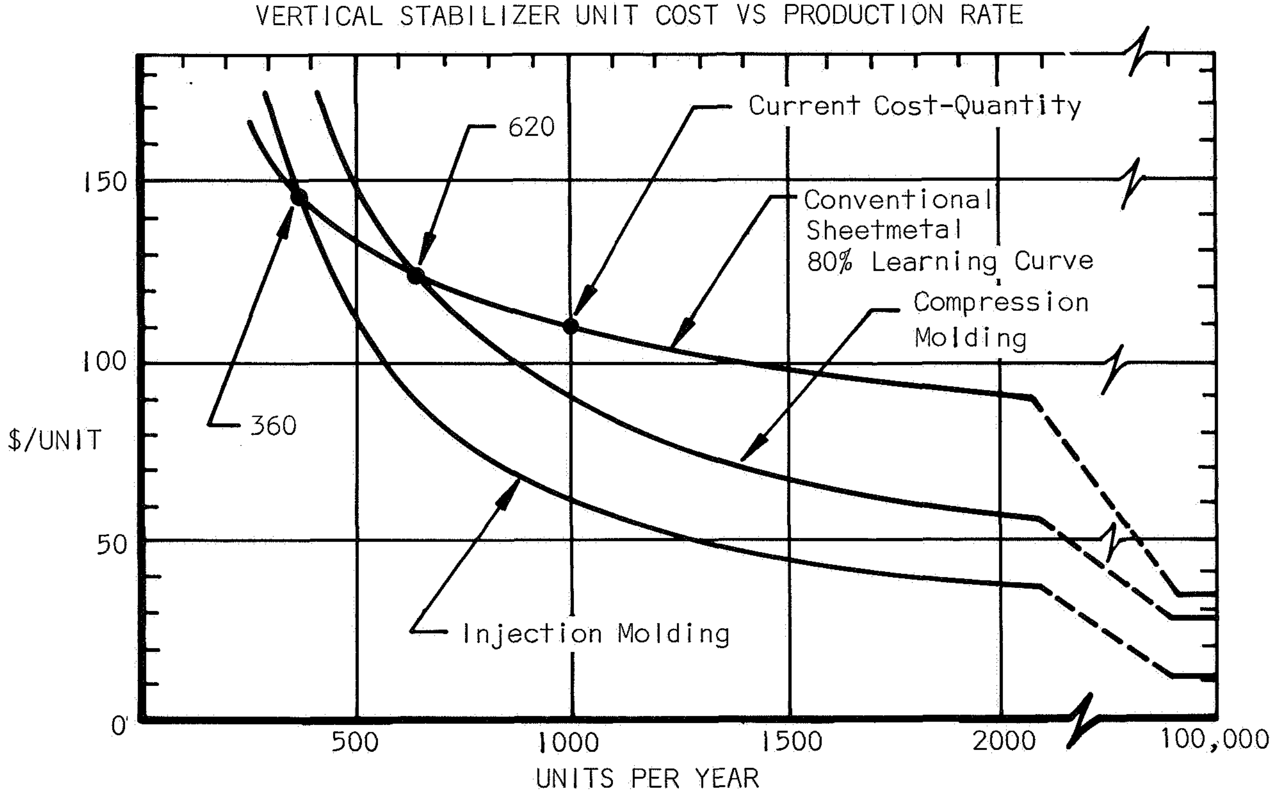

Figure 52

* The reader should be aware that the "learning curve" is a function of cumulative quantities, which were assumed to have occured within one year; for comparison with the yearly production rates of the molded units.

Referring to Figure 52, conventional sheetmetal construction unit cost is less than that for compression molded and injection molded construction only at production rates less than 620 and 360 units per year, respectively.

Horizontal tail.-This portion of the airplane, being more heavily loaded than the vertical stabilizer, requires the use of an epoxy/glass composite. Neither chopped E-glass/polyester nor injection molded nylon 6-10 is structurally adequate for most of the horizontal tail components. Therefore, most if not all of the ten different reinforced plastic horizontal tail components will be compression molded from an epoxy/glass composite.

Referring to Figure 53, components (-11, -25, & -27) might later be proven to be more economically produced from injection molded nylon 6-10. The skin quarter-panels, i.e., top or bottom on either side (-1 or -3), will require a molding press capacity of approximately 1500 tons (for compression molding) This is easily within today's readily available capacity.

The main spar (-5) will require 400 to 500 tons for molding, but a larger tonnage capacity press might have to be used to provide large enough platens. The spar, as molded, is only 4.5 in. wide, but is over 151 in. long. Alternate approaches might be to build extensions for the platens outside the main platen area, or to mold the spar in two presses set side by side.

The trailing edge spar, and the anti-servo tab channel and skin (-7, -28, & -21, respectively), also being of outsize lengths, will each require either excess press capacity (tonnage), or two or more presses set side by side.

The torque box (-9), if it is made in one piece as indicated, will require a large core on each side to form the pans on each side. As mentioned earlier, this part might be easier and cheaper to fabricate if it were separated into two identical ribs and a shallow box.

Wing.-The wing, being the most demanding of all the primary structural components, requires the use of at least S-glass/epoxy, and preferably high modulus graphite/epoxy. For the purpose of this study, the wing components were assumed, in general, to be fabricated in a manner similar to the vertical tail. I.e., die costs, in general, were estimated on a projected area basis, proportionate to the vertical tail die costs. This is valid for dies of comparable depth and complexity. Most of the wing components are no larger than the horizontal tail components. Exceptions to this are the spar and the skins. Molding presses of more than adequate capacity are available today, even for components as large as the spar and skins. Dies for the spar could possibly be made in segments due to their out-size length requirements. The spar could be molded in one big press or in a series of presses set side by side.

For outsize, but simple, components such as the wing skins (with no integral stiffeners) a new castable ceramic mold material offers significant

HORIZONTAL TAIL, EXPLODED VIEW

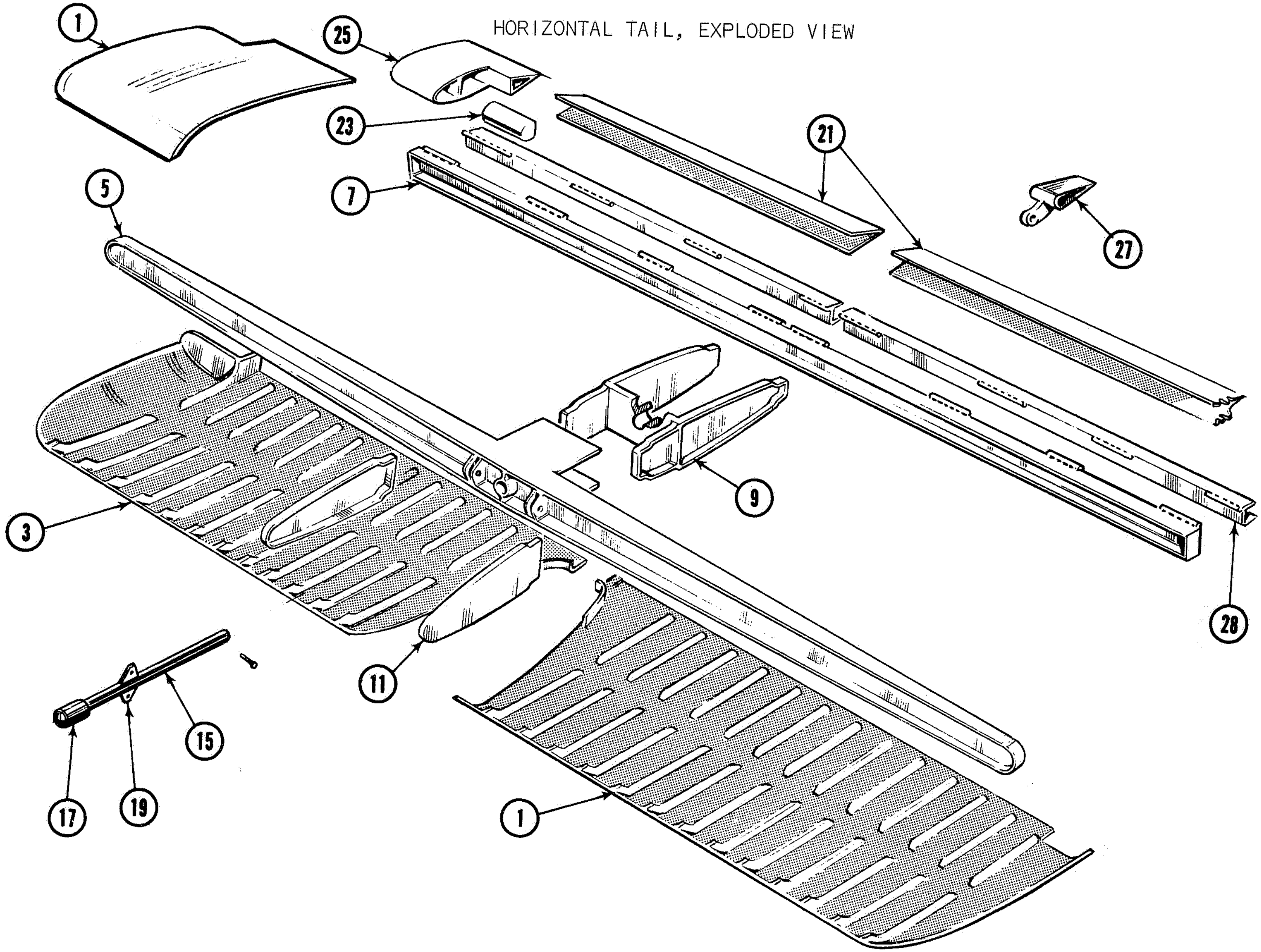

Figure 53

WING, EXPLODED VIEW

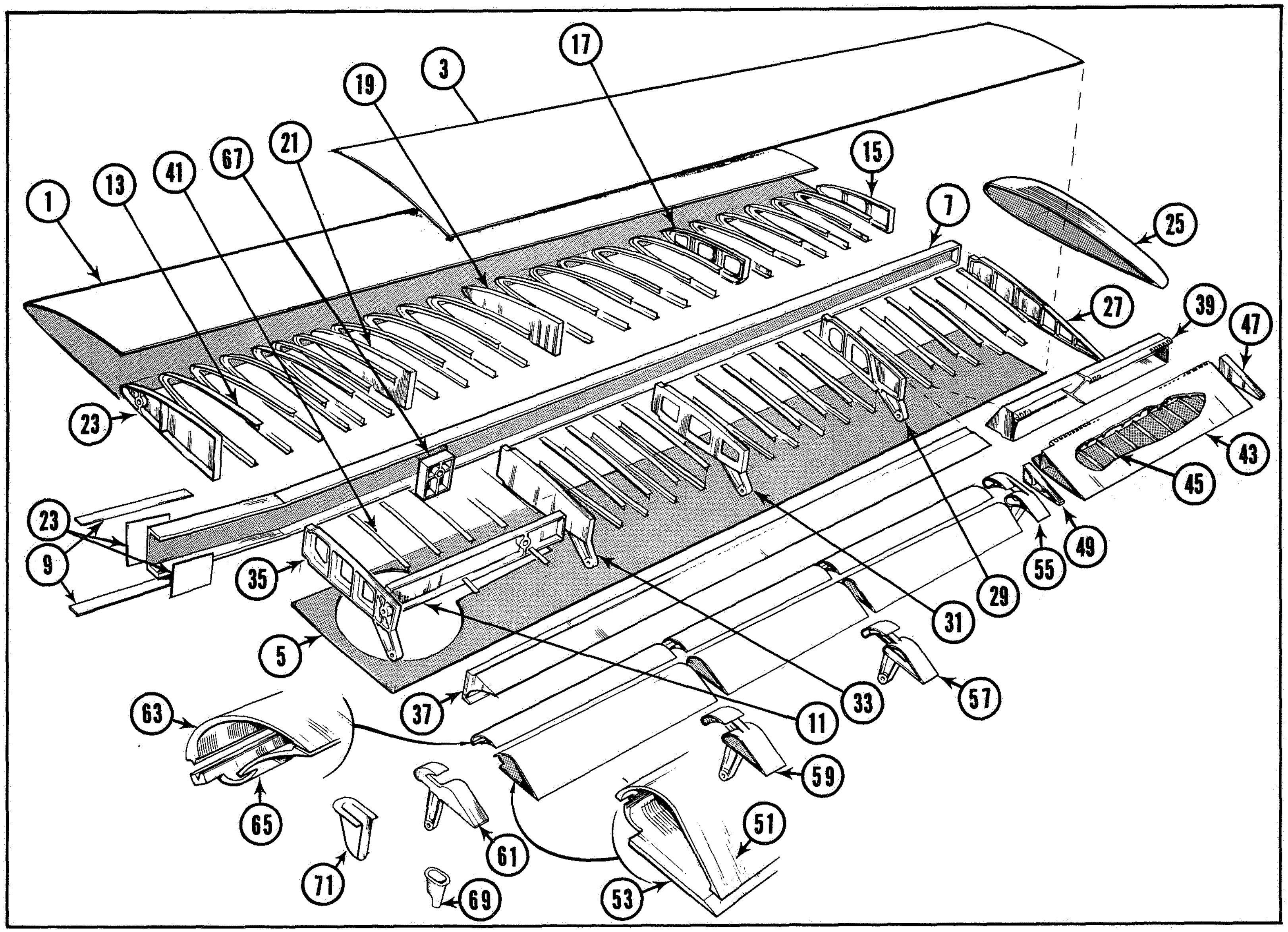

Figure 54

cost savings. It is not recommended for applications such as shapes with stand-up ribs or where cores are required. Such molds are normally fabricated with a two-inch thickness of the ceramic material backed up with foamed fused silica blocks. The bonded-on foam blocks are cut smooth and flat and mounting studs are then potted into the foam. No internal reinforcing is employed. Another advantage is the ability to cast-in-place all the necessary electric heaters or steam lines. The basic cost of this ceramic material is $1100/ton (i.e., $0.55/lb). It has a density of 120 lb/ft^3.

These molds can be fabricated in matched sets (male and female) and are completely adequate for the 1000 psi compression molding requirements. Die cost for the wing skins was based on the use of matched sets of the above ceramic molds. Using only a female mold and a pressure bag reduces total wing manufacturing cost by a maximum of 3%. Since the (bagged) inner skin surface is not as reproducible as with matched molds, the 3% is well spent, to minimize bonding preparation for the skin stiffeners.

As with the vertical and horizontal tail, the wing is assumed to be assembled by secondary bonding in appropriate jigs and fixtures.

The first cost analysis, based on the tapered wing illustrated in Figure 54, assumed that each component* would be machine molded individually in a press of appropriate capacity. This first analysis considered both the 30-minute cure time for current epoxies and an estimated cure time of 15 minutes for future epoxies. Referring to Figure 55, bars (1) thru (5) represent the above described wing. Bar (1), for single-cavity molding and 30-minute epoxy cure time, has a molding cost which is 54.2% of the total wing manufacturing cost. Therefore the savings in bar (2) are large when the production rate is doubled, by halving the current 30-minute cure time.

Subsequent analyses of the wing based on the use of multi-cavity dies, took advantage of the potential savings attainable with higher production rates. Since factory time is not practically available below $10 per hour, the minimum size molding press considered was 650-ton capacity, which cost about $12 per hour to operate. It turned out that platen area, not component projected area/pressure requirements, determined the number of cavities per die or the number of die modules. It was first assumed that only a constant chord constant thickness wing, with its many identical parts, could take advantage of multi-cavity molding. I.e., it would be impractical to attempt to mold dissimilar or unidentical components on the same stroke of the press. This would be true due to the slight difference in molding requirements between unidentical components. It turns out that the no-two-parts-alike tapered wing can also take advantage of multi-cavity tooling, when the associated quantities are on the order of 100,000 units per year, as in these analyses.

Referring again to Figure 55, bars (3) and (4) represent the same wing as bars (1) and (2), respectively, except for the use of multi-cavity tooling.

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*There are no two components alike in the tapered wing.

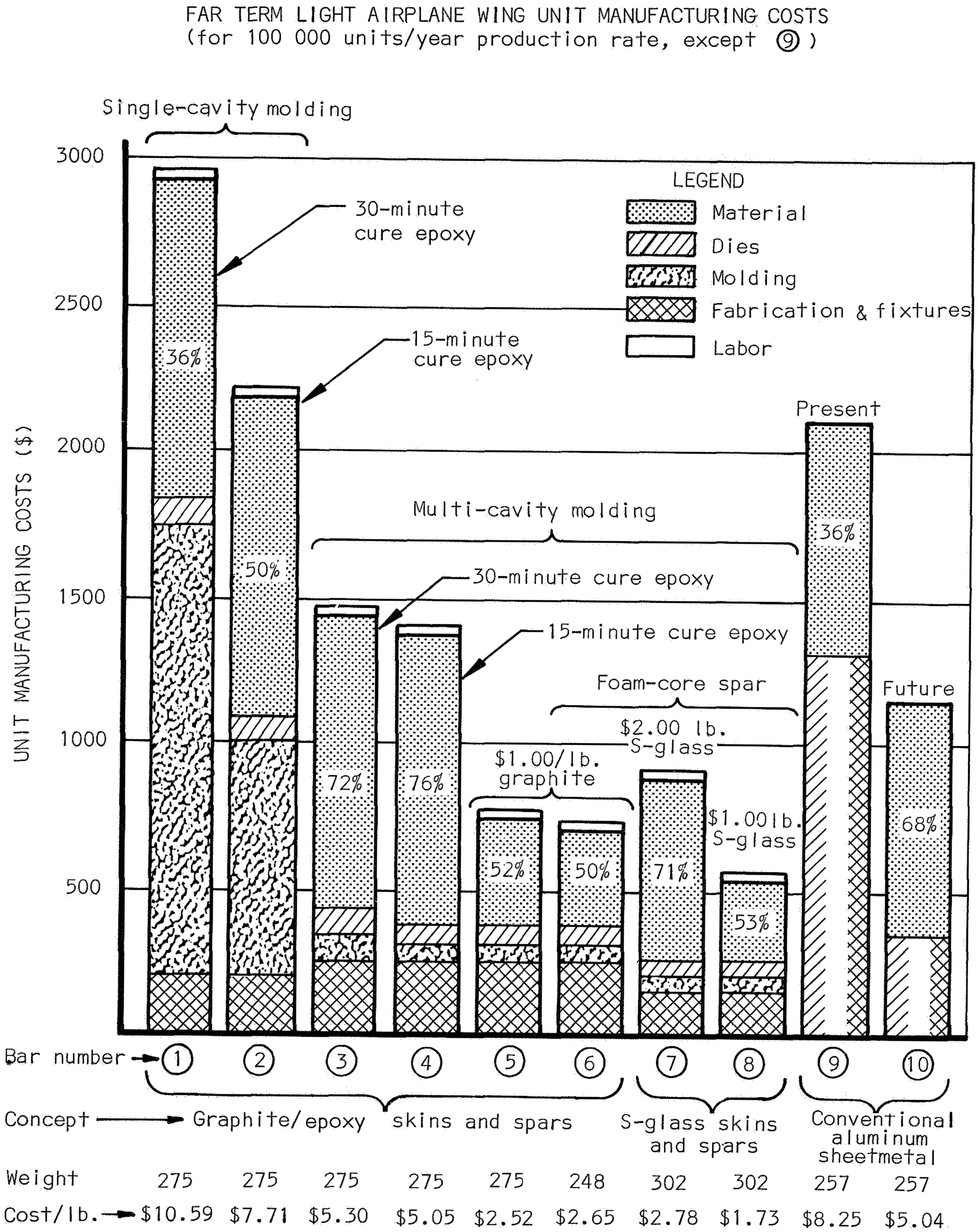

FAR TERM LIGHT AIRPLANE WING UNIT MANUFACTURING COSTS
(for 100 000 units/year production rate, except 9)
Single-cavity molding
30-minute cure epoxy
15-minute cure epoxy
LEGEND
Material
Dies
Molding
Fabrication & fixtures
Labor
Present
Multi-cavity molding
30-minute cure epoxy
15-minute cure epoxy
Foam-core spar
$2.00 lb. S-glass
$1.00/lb. graphite
$1.00 lb. S-glass
Future
36%
50%
72%
76%
52%
50%
71%
53%
36%
68%
UNIT MANUFACTURING COSTS ($)
3000
2500
2000
1500
1000
500
Bar number 1 2 3 4 5 6 7 8 9 10
Concept Graphite/epoxy skins and spars
S-glass skins and spars
Conventional aluminum sheetmetal
Weight 275 275 275 275 275 248 302 302 257 257
Cost/lb. $10.59 $7.71 $5.30 $5.05 $2.52 $2.65 $2.78 $1.73 $8.25 $5.04

Figure 55

Multi-cavity molding appears to offer a significant reduction in unit manufacturing cost; i.e., about 35%, for the 15-minute cure wings.

Examination of bar (4) makes apparent the high (76%) portion of the wing unit cost represented by the raw material. Most (82%) of the raw material in bar (4) is for graphite/epoxy at $5.00 per pound. Obviously, the unit manufacturing cost of the wing is a significant function of the cost of graphite.

Industry sources have estimated the cost of graphite in fifteen years, ranging from $1.00/lb. to $100.00/lb. Bar (5) optimistically charts wing unit manufacturing cost for the same wing as bar (4), using $1.00/lb. rather than $5.00/lb. graphite. Figure 56 plots the cost of the multi-cavity molded, 15-minute epoxy cure wing as a function of the cost of graphite up to $10.00 per pound.

Bar (6) in Figure 55 plots a wing comparable in cost to bar (5) which which has a foam-core spar, offering a significant (10%) weight reduction over the wings considered in bars (1) through (5).

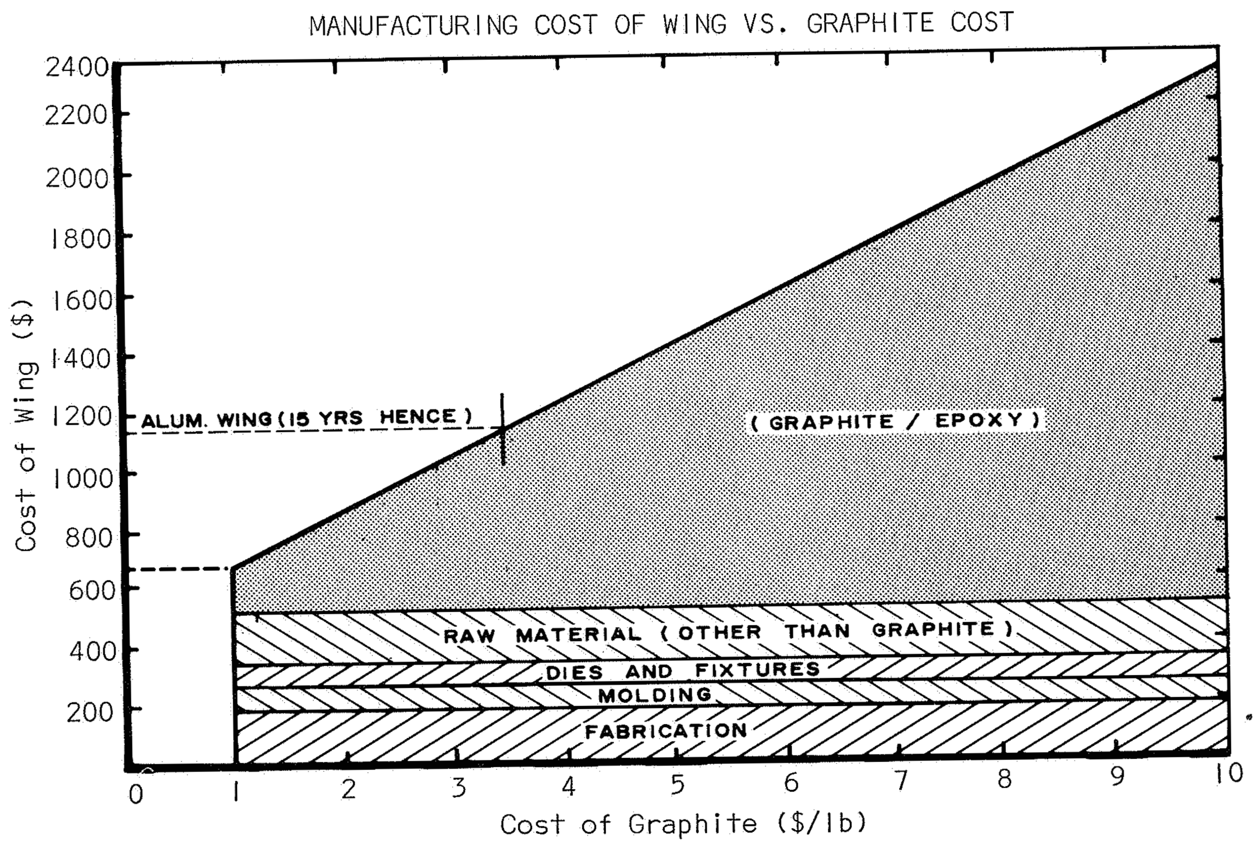


Figure 56

FUSELAGE, EXPLODED VIEW

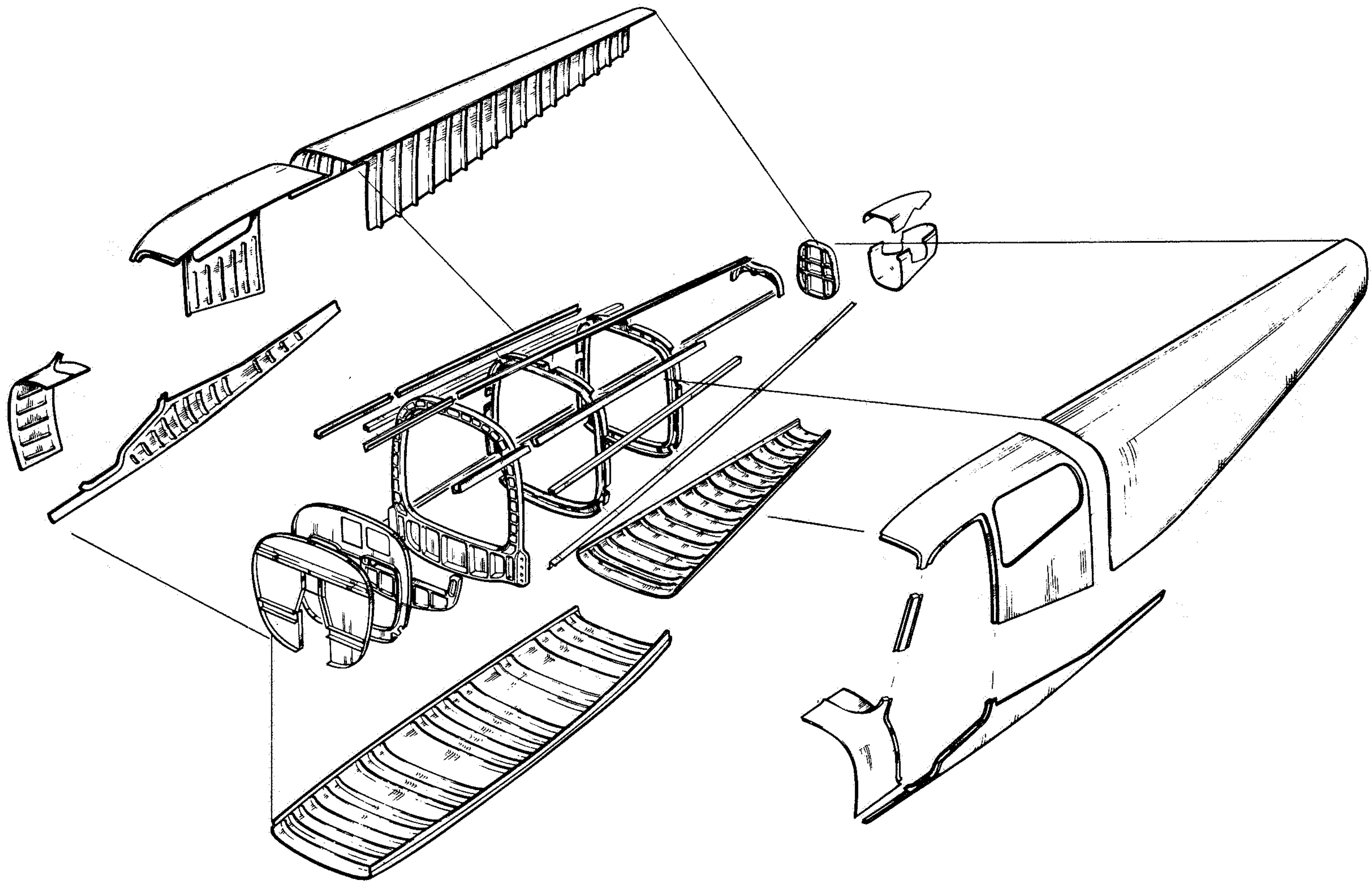


Figure 57

For comparison, a wing which replaces the graphite/epoxy components with S-glass/epoxy components is plotted as bar (7). Its cost also is a significant function of the cost of the S-glass/epoxy (i.e., $2.00 per pound). Some savings in fabrication are realized by molding the many skin stiffeners integral with the skins. This is accomplished by first partially curing the unidirectional filament skins and then integrally molding the chopped fiber stiffeners to the skins, finally curing them together.

It is only fair to assume that S-glass could eventually be procured at a cost equally as low as graphite. Therefore, in Figure 55, bar (8) indicates a low unit manufacturing cost of approximately $521.00 for an S-glass wing using $1.00/lb. S-glass.

Referring to Figure 55, the foam-core graphite wing [ref. bar (6)]then appears to have the lowest weight with a very low unit cost; but the S-glass wing [ref. bar (8)] has the lowest unit cost and a significantly lower specific unit cost of $1.73 per pound.

Bars (9) and (10) plot the wing unit manufacturing cost of a conventional sheetmetal (aluminum) wing. Bar (9) is based on current production quantities and bar (10) represents reduction in cost due to high production rates and the classic 80% learning curve.

Fuselage.- All the fuselage (see Figure 57) components except the stainless steel firewall and the channels and longerons are large, but conventional, compression moldings. All the previous discussions of compression molding consideration associated with the tail and wing, are equally applicable to the fuselage components. The channels and longerons, having constant cross-sections, can readily and economically be bag molded over male dies. Even with the specification of continuous and unidirectional filaments for the channels and longerons, there is a possibility that each might be molded in a continuous lay up and cure operation. Like the vertical and horizontal stabilizers and the wing, the fuselage would be an all-bonded assembly.

In conclusion, it can be said that the most significant reductions in light airplane unit manufacturing cost will be the result of high (mass) production methods and processes. E.g., machine molding and forming of primary components, all-bonded assembly, numerically controlled spot welding and riveting, prepriming (at the mill) of aluminum sheets (for bonding), automatic nondestructive inspection of bonded joints, etc. Less tangible, but significant savings are realized with the elimination of corrosion on plastic componets.

Although this study has concentrated heavily on the utilization of plastic materials, aluminum will remain a prime candidate for light airplane structure in the future. Aluminum is exceptionally machinable, formable, and joinable. Its use will continue with the greater use of mass production techniques mentioned above. Greater use of 6061 T6 and 5086-H32 aluminum alloys will likely occur with resultant savings in material cost. No one group of materials, metallic or nonmetallic, will be used universally. It will still remain for the designer to weigh the pros and cons of each material for each individual application. See Appendix A for an estimated consummer price breakdown of the Far Term airplane.

FATIGUE CONSIDERATIONS

Existing requirements for the strength of light airplane structures are based largely on the concept of "one-time" loading. For many years this appeared to be satisfactory but recently it has been recognized that the margin of safety provided against failure under "one-time" loading may no longer be adequate with respect to the repeated loads which occur during the lifetime of the aircraft. A survey of the 1963 General Aviation Accident Reports indicates evidence that some airframe failures could be attributed to fatigue.

Whether or not the failures involved were the result of inadequate pilot proficiency, lack of respect for adverse weather, or the result of inadequate inspection and maintenance is of secondary importance. The point is that the airplane involved encountered flying conditions which resulted in loads being applied to the airframe of sufficient magnitude and frequency to cause catastrophic failure of the primary airframe structure.

Establishing a Fatigue Load Spectrum

Up to the present time, light airplane manufacturers have designed their aircraft to FAA requirements per F.A.R. part 23. This document does not require proof by analysis or test of the "safe life" or "fail safe" characteristics of their aircraft. At the same time little data is available with regard to what load spectra should be used by operators of the various category airplanes.

An assessment of repeated loads on general aviation and transport aircraft is being conducted with the F.A.A. by NASA's Langley Research Center; the results to date are presented in references 32 and 33. They reveal a large amount of scatter in the repeated load history, due principally to the diverse nature of general aviation.

Composite VG records (positive and negative accelerations vs airspeed) from references 32 and 33 for different types of operations are presented in Figure 58. These data are superimposed upon their respective V-n diagrams to indicate where the most severe areas might be in respect to possible exceedances of the design flight envelope. Design flight envelope exceedances in the low speed portions are probably due to landing shocks and are not considered significant.

A review of the instructional flying records, Figure 58, reveals a case where a particular aircraft exceeded the design dive speed as well as the positive and negative limit load factors at the design dive speed.

The twin-engine executive operations, Figure 58, show one case of exceeding the negative limit load factor at a speed slightly less than design cruise. Investigation revealed the incidence to be gust induced.

The following significant conclusions can be made after reviewing the composite VG records.

COMPOSITE VG RECORDS — FIVE TYPES OF OPERATIONS

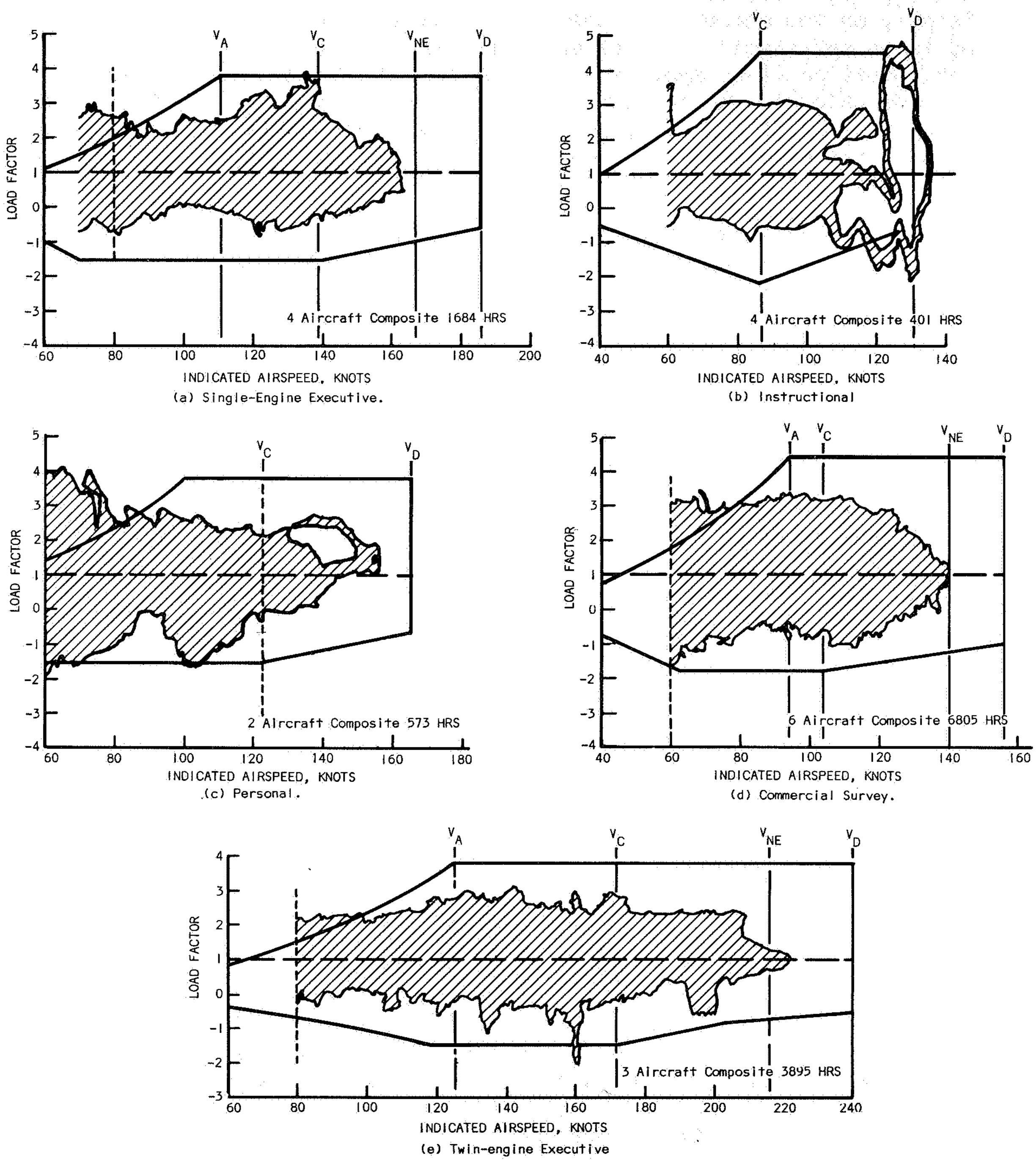


Figure 58

(1) Atmospheric-induced, as well as pilot-induced, loads in excess of the design flight envelope may be encountered during normal operation of the general aviation fleet.

(2) All types of operations are flown above the design cruising speed.

It is evident, therefore, that General Aviation should be classified into different roles. Needless to say, the fatigue load spectrum will be different for each role.

Estimation of Fatigue Life

The estimation of fatigue life using the "Miners" Cumulative Damage Rule involves the calculation of damage incurred on the airplane as a direct result of its operating environment.

Generally speaking the operating environment for a light airplane, regardless of its type of utilization such as executive, personal, instructional or commercial survey operation, can be defined as follows:

(1) Gust Environment - The airplane while in steady flight encounters a specified number of positive and negative gusts of varying intensities defined by the gust spectrum for the airplane.

(2) Maneuver Environment - The airplane is subject to a specified number of positive and negative maneuvering loads of varying intensity defined by the maneuvering spectrum for the airplane.

(3) Ground-Air-Ground Environment (G.A.G.).- At least once per flight the airplane is subject to loads associated with the following conditions.

a) Taxi condition at maximum take-off weight.
b) Steady lg Flight Cruise Condition at minimum landing weight.
c) Landing impact loads at maximum landing weight.

From a structural design aspect it is apparent that before any design fatigue load spectrum can be developed and before any safe life prediction can be made, it is necessary to define not only in what roles that airplane is going to be utilized, but also for how long it is going to be utilized in one role before being used in another role. This is obvious when one is confronted by the following statements:

(1) Landing Impact Acceleration for instructional-type airplanes is more severe and more frequent, approximately 4 per 30-minute flight, than on any other category light airplane and will account for a considerable amount of damage in the fatigue life of the airplane.

(2) Commercial Survey Aircraft have the longest flight duration, therefore less G.A.G. damage is inflicted on the airplane. They have more

severe gust experience than other types of usage, since 97% of the time they are in rough air.

Pressurization Considerations

The effect of pressurization produces a stress configuration consisting of hoop stress and longitudinal stress in addition to the in-flight shear, bending moment, and torque loads on the fuselage structure. It then follows that the weight of the basic pressurized fuselage will be higher than that of an unpressurized fuselage. From a minimum weight standpoint, the optimum structure is cylindrical with the elimination of flat or slab panels.

Sealing requirements demand that careful consideration be given to the number and spacing of rivets, particularly at longitudinal and transverse skin splices and at the attachment of pressure bulkheads and canopy structure. Likewise, more care must be taken in the fabrication, inspection, and quality control of the fuselage structure, particularly in the region of cut outs in the structure for windows, entry doors and access doors, at the attachment of the floor structure to the frames of the fuselage, and at the intersection of the wing and fuselage.

Entry doors and their locking and operating mechanisms should be designed on the fail safe concept to insure that the door structure and the sealing qualities are adequate should a simple failure in one of the latches or shear pins occur.

The use of metal-to-metal adhesive bonding, particularly to reinforce areas where high stress concentrations are present, increases the fatigue life of the fuselage. It demands good quality control and considerable component testing. Materials exhibiting low crack progagation characteristics are important. As an example, it has been shown (ref. 34) that 7075-T6 aluminum alloy is more prone to explosive fracture than 2024-T3 alloy.

From a structural standpoint, it is highly probable that any fatigue crack, once started, will tend to run longitudinally along the fuselage. This is due to the fact that in a pressurized fuselage the stringers are fairly closely spaced and the hoop tensile stress is twice the longitudinal stress. For this reason, circumferential reinforcing rings are placed at intervals along the fuselage to arrest the crack propagation of a fatigue crack and to reduce the hoop stress in the skin.

The spacing and cross section of the reinforcing rings are important Williams (ref. 35) states that rings spaced more than 30" apart, while locally restricting the radial expansion of the skin, allow unrestricted expansion in the area midway between the rings; with a 10" spacing the radial expansion of the skin nowhere exceeds that of the rings by more than a small percentage, so that the maximum hoop stress in the skin is equally reduced by material added to the rings as by the weight added to the skin.

Material Fatigue Properties

Many mechanical devices are subjected to forces that vary in magnitude and, often, in direction. If this variation occurs a relatively small number of times and the stresses do not exceed the yield strength of the material, design studies can be made safely on the basis of the static properties of the material. Unfortunately, this is not true in the design of airplanes since the structure usually experiences many repeated loadings (magnitude and direction) in its service lifetime.

This section summarizes and compares the fatigue properties of some of the basic materials as previously selected for aircraft structural applications. This data has been compiled and evaluated to present a qualitative picture of the fatigue characteristics associated with the material.

For the most part, complete information was not available for the materials; therefore, various methods were utilized in extending the data to provide information which could not be obtained directly. All the fatigue data shown represents axial loading tests and is ultimately plotted as standard S-N curves whereby the points along the curve represent the number of loading cycles a material may endure at a particular max stress before failure.

S-N curves for notched and unnotched sheet specimens representing stress Ratios (R) of -1.0 and +0.25 are shown in Figures 59, 60, 61, and 62. For the most part, these curves are derived by means of averaging the results directly from several references as shown in the respective tables.

Where basic information in the reference did not provide data representing correct stress ratios from which comparisons could be made, the basic data is expanded through use of an approximate Modified Goodman diagram. This method is described in reference 36.

The reference literature (ref. 37) associated with the 4130 and 4340 materials provided fatigue data in terms of alternating and mean stress. With use of modified Goodman Diagrams it is possible to reconstruct S-N curves (Figures 63 & 64) as a function of maximum-stress and any stress ratio desired.

Figure 60 illustrates that the fatigue strength of the higher strength aluminum alloy (7075-T6) actually is inferior to the lower strength alloys. This would suggest that increases in static strength have been obtained at the expense of an actual reduction in fatigue strength.

This is not true in the comparison of 4340 and 4130 steels; however, the difference in the static strength of these two materials is much greater than the difference in the fatigue strengths (ref. Figures 63 and 64).

Comparison S-N curves for plastic laminates reinforced with unwoven glass filaments are presented in Figures 65, 66, and 67. The curves represent three constructions: (1) all plies parallel, (2) alternate plies $\pm 5^{\circ}$ to the

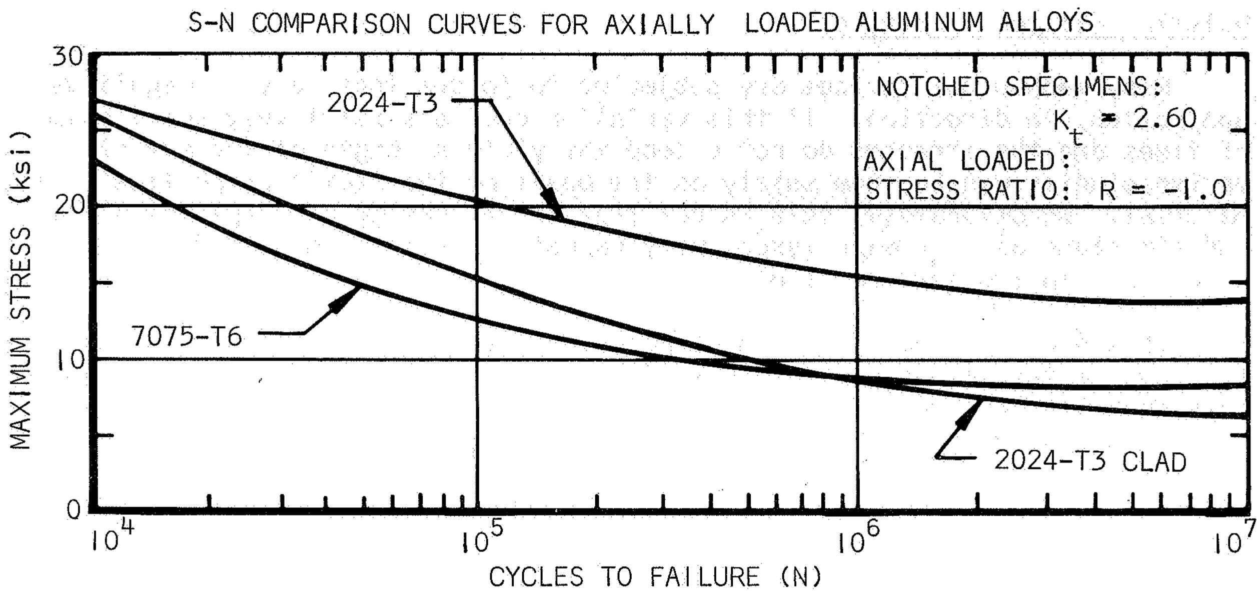


Figure 59

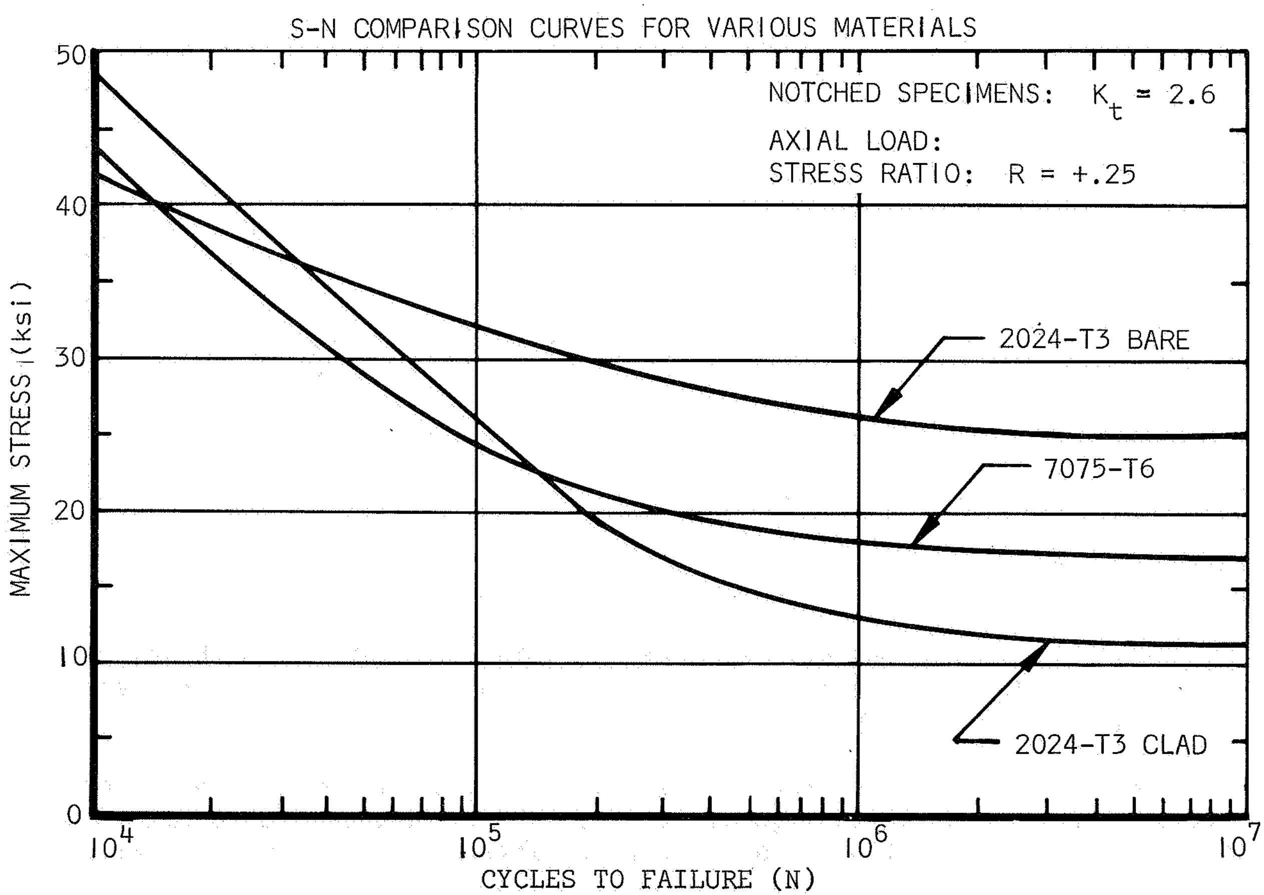


Figure 60

S-N COMPARISON CURVES FOR AXIALLY LOADED ALUMINUM ALLOYS

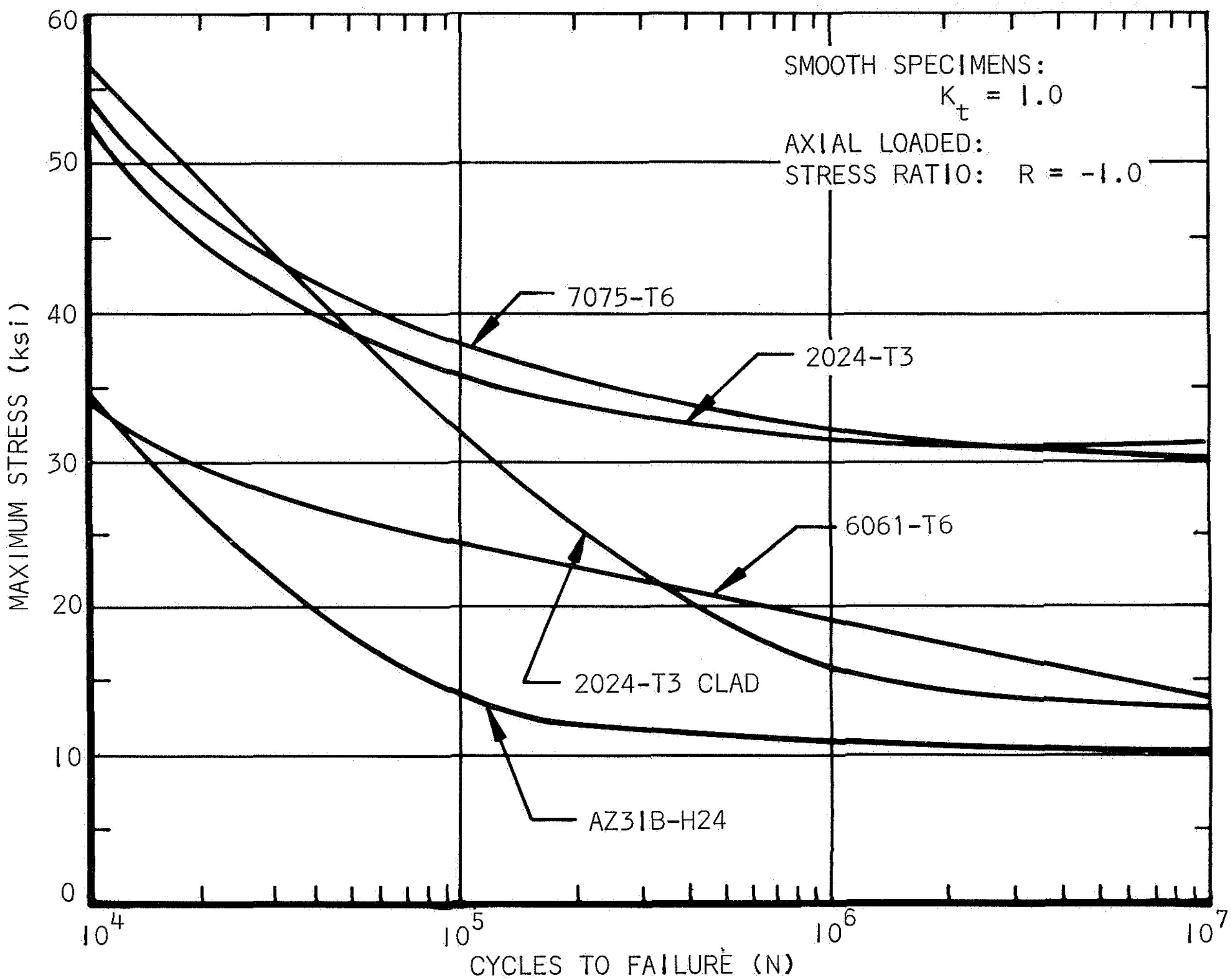


Figure 61

principal axis, (3) alternate plies 0° and 90° to the principal axis. All indicate the fatigue strength of the S-glass filaments to be superior to the E-glass type. It also appears (Figures 66 and 67) that the fatigue characteristics of the S-glass laminates may be even further improved with the use of different resins.

In recent years, more and more consideration is being directed toward the fracture characteristics of materials. Acceptance is given to the fact that fatigue failures could occur as a result of one or a combination of several loading environments. These environments include normal working loads, noise induced vibrations, and accidental damage. When a crack originally develops in a structure, it creates a point of high stress concentration, and subsequent application of normal service loads will cause further extension of

S-N COMPARISON CURVES FOR VARIOUS MATERIALS

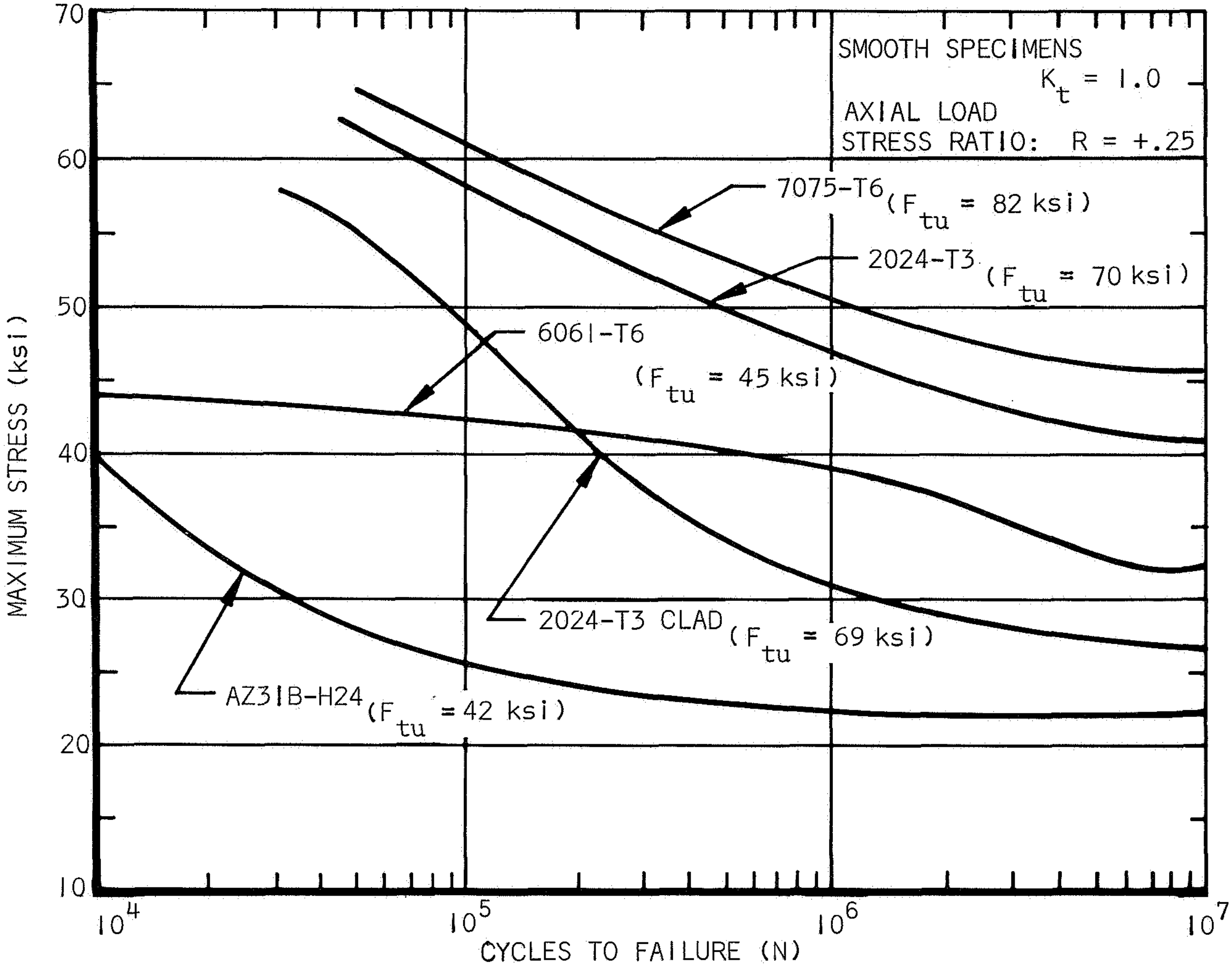


Figure 62

the crack. This extension, of course, largely depends upon the load/stress level and the inherent crack-propagation characteristic of the material. It is extremely important these cracks be detected before they can extend to a length which would cause a catastrophic failure. Structural inspections take place periodically, and consist of frequent visual examinations to detect any obvious defects, together with a detailed overhaul about once a year.

Two questions which still need answering are as follows:

(1) How long must a crack be before it can be detected?
(2) How long can it become before it leads to serious failure?

The ideal condition would be such that a defect which is approaching a detectable length would not become catastrophic prior to the next scheduled

S-N COMPARISON CURVES FOR AXIALLY LOADED 4130 & 4340 STL. ALLOYS

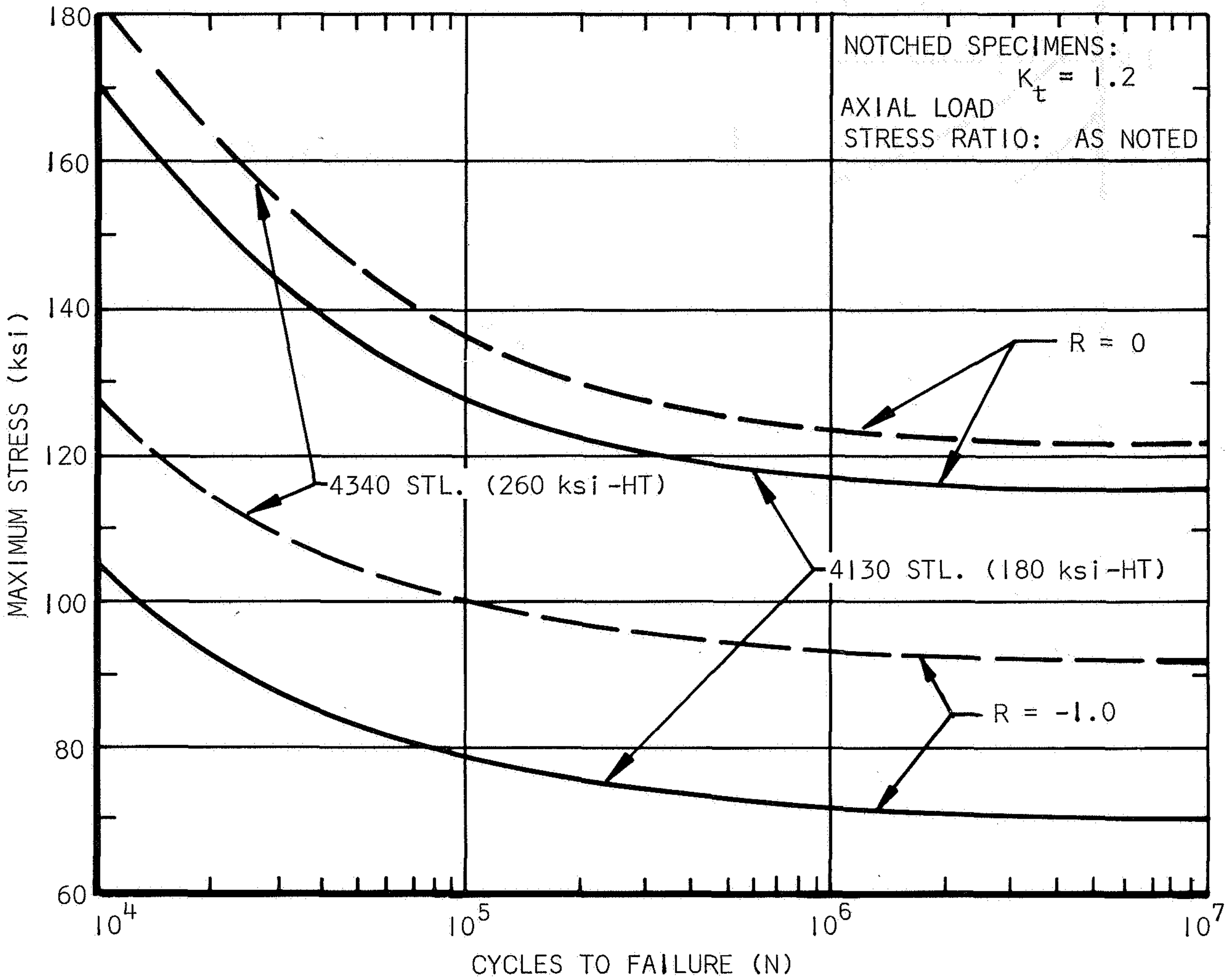


Figure 63

inspection. A good design would therefore consider a material which would satisfy these requirements; i.e., low crack-propagation rate to allow sufficient time for crack detection and high notch resistance to insure adequate strength at any crack location. These requirements actually have led to a return to the use of lower strength aluminum alloys, particularly in fatigue critical areas.

S-N COMPARISON CURVES FOR AXIALLY LOADED 4130 & 4340 STL. ALLOYS

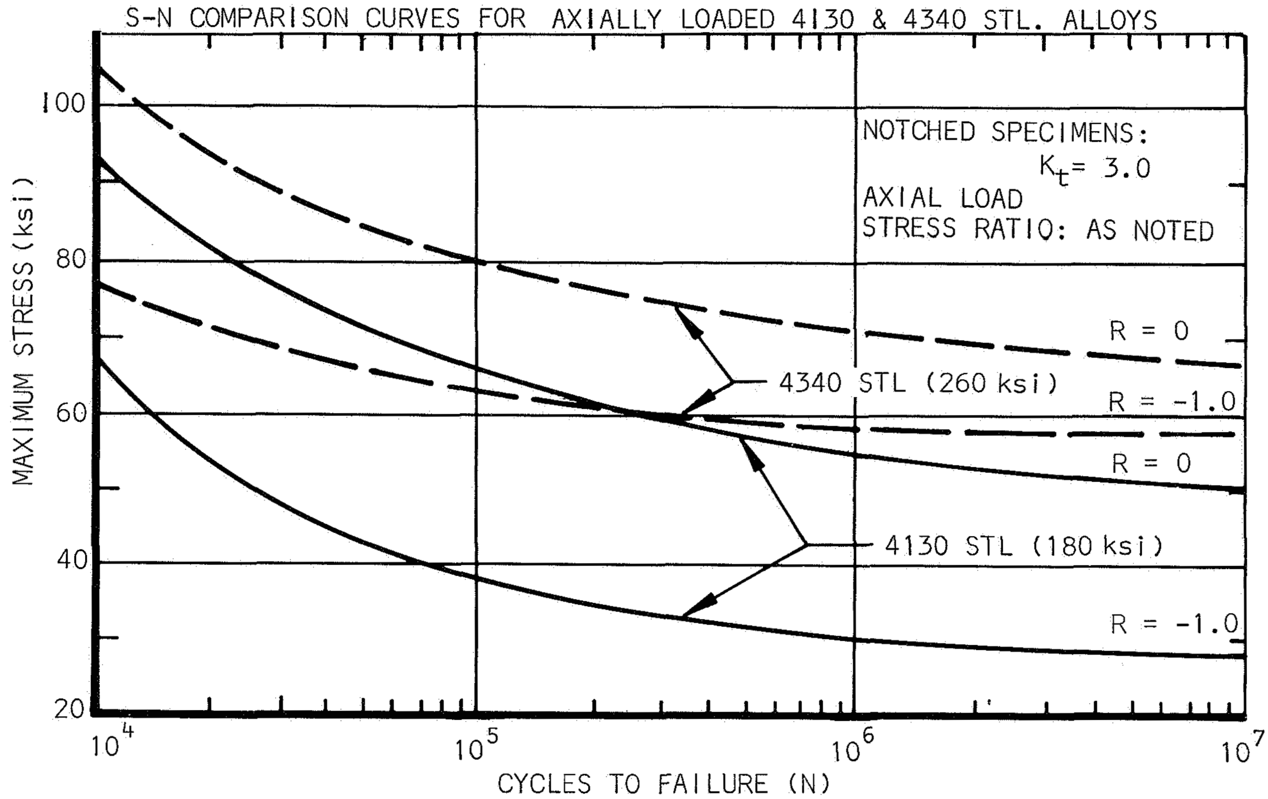


Figure 64

S-N CURVES OF LAMINATES MADE OF SCOTCHPLY 1002 RESIN AND UNWOVEN GLASS FIBERS, ALL ORIENTED PARALLEL TO THE PRINCIPAL AXIS

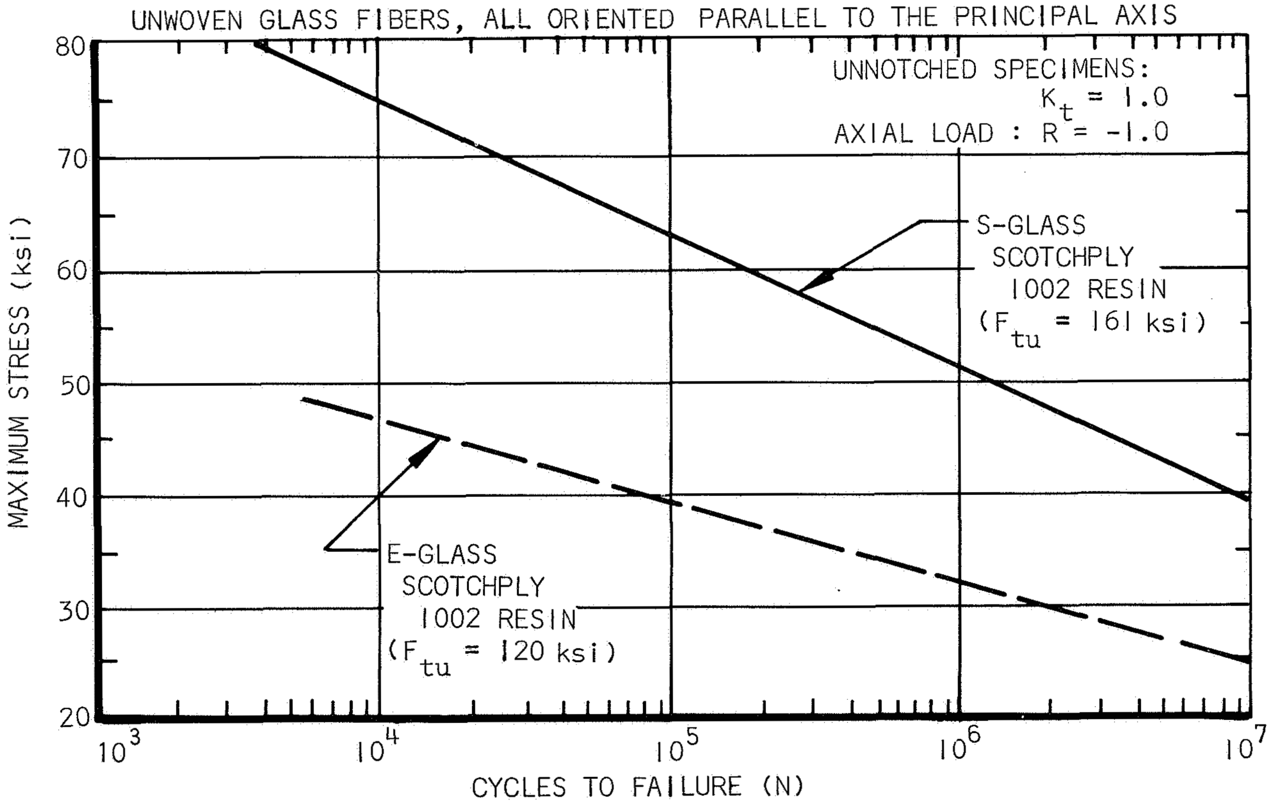


Figure 65

S-N CURVES OF LAMINATES MADE OF SCOTCHPLY RESINS AND UNWOVEN GLASS FIBERS HAVING ALTERNATE PLIES AT 0° and 90° TO THE PRINCIPAL AXIS

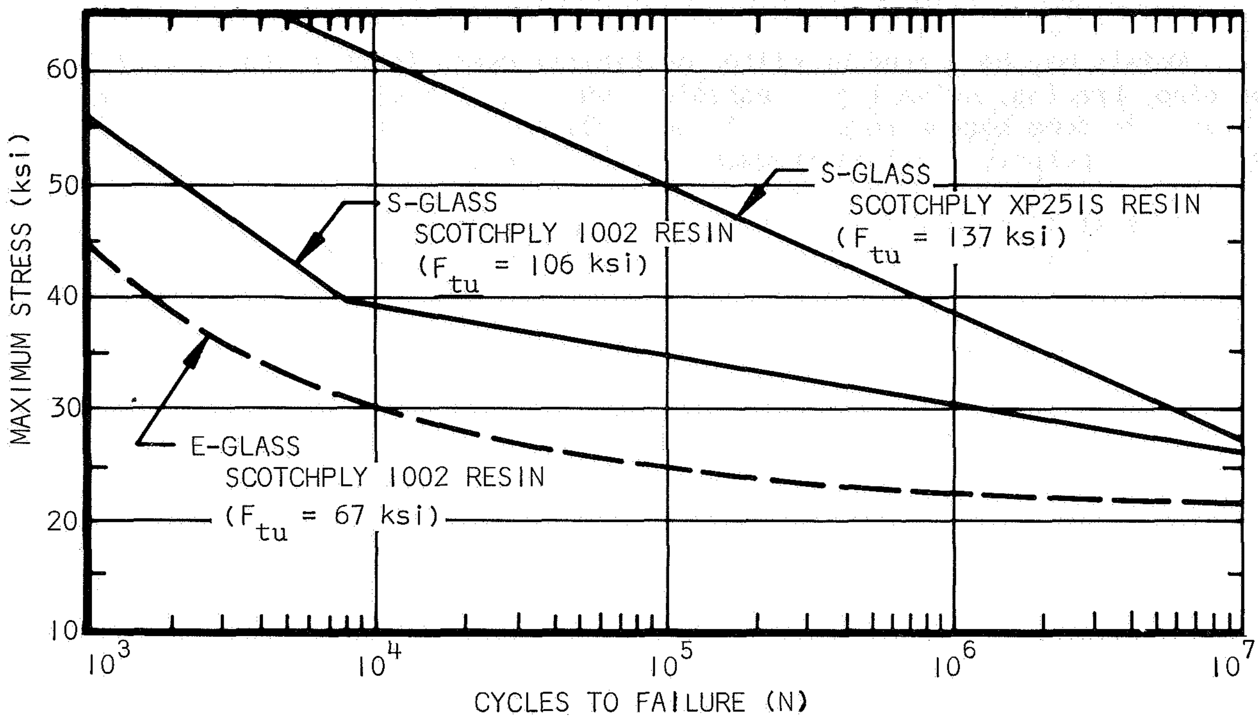


Figure 66

S-N CURVES OF LAMINATES MADE OF SCOTCHPLY RESINS AND UNWOVEN GLASS FIBERS HAVING ALTERNATE PLIES ORIENTED AT ±5° TO PRINCIPAL AXIS

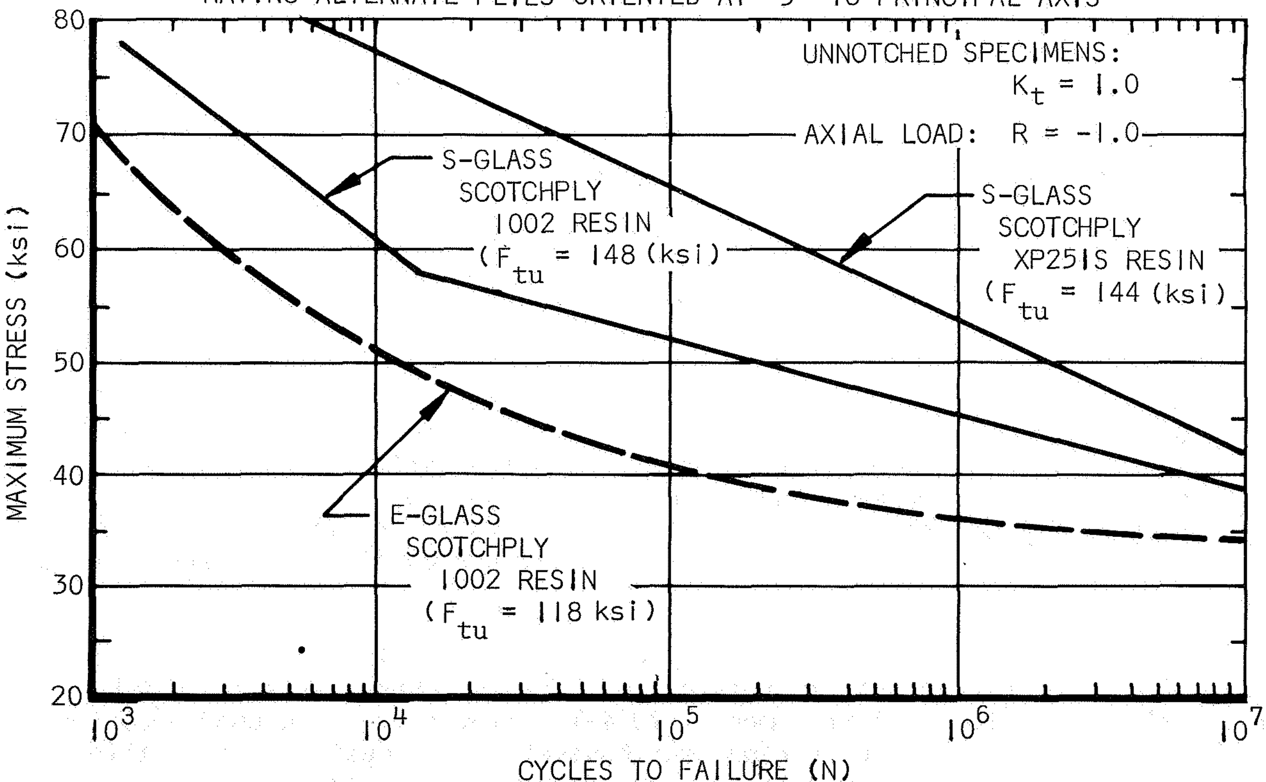


Figure 67

FASTENING DEVICES AND METHODS

Metals may be joined by either mechanical means (such as bolting) or by welding, brazing, soldering or adhesive bonding. All of these methods may be used to some degree in aircraft construction. Soldering is never used for structural purposes, but is frequently used in electrical work.

This section includes a discussion of the various joining processes adaptable to aircraft construction. Each method is presented in the following manner:

(1) A brief description.

(2) Illustrations are provided as necessary to clearly define the method of construction.

(3) Typical allowable strengths are given where applicable.

(4) Some comparisons (Fatigue and Static Strengths) are made between two or more of the techniques used.

(5) Advantages and disadvantages of each method are listed.

(6) Typical applications in aircraft manufacturing are given for each joining process.

Riveting

Rivets play an important role in the light aircraft industry. At the present it is the primary method of joining aluminum. Riveted construction is readily controlled and inspected, and it does not require the application of heat that might partially anneal or significantly impair the corrosion resistance of the heat-treated alloys used. The limited heating required in dimpling sheets of some alloys, and tempering before riveting does not impair essential properties. Sheets less than 0.050 inch thick generally are dimpled for countersunk head fasteners. Thicker material is machine countersunk.

Countersunk head rivets are used primarily for attaching outer skins whereas universal-head (modified round) rivets are used extensively in interior structures where protruding heads are not objectionable. Surface skin panels often are riveted by automatic machines (as illustrated in Figure 68) made to form one or both heads of the rivet. The machines are fed with rivets or slugs; and the heads are usually shaved flush with the exterior surface.

Rivet alloy 2117-T4 is the most popular for general use, especially for automatic riveting, because it retains good driving characteristics indefinitely after solution heat treatment. 2024-T4 alloy rivets are used occasionally where higher strength is required; however, these must be used within 30 minutes after heat treatment, or refrigerated until used.

STANDARD AUTOMATIC RIVETING MACHINE

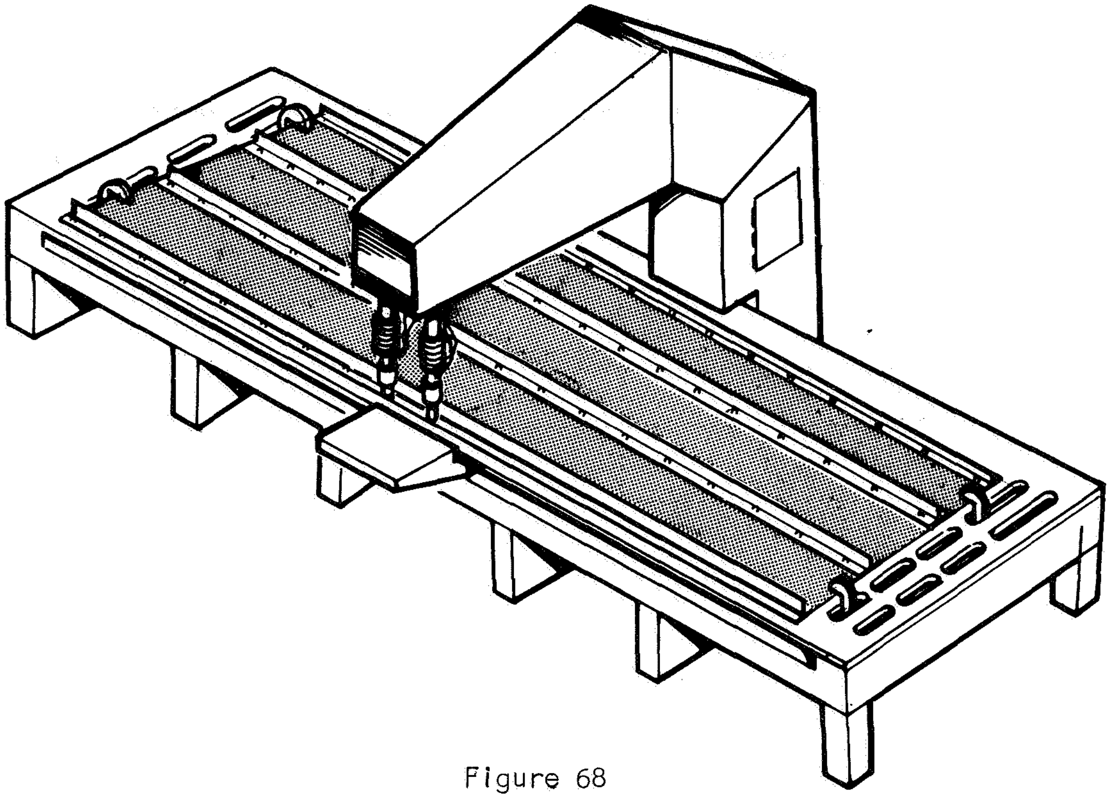


Figure 68

Specifications for the design of aluminum-alloy structures generally designate the rivet alloys to be used. Table XVII lists some combinations of structural and rivet alloys that combine satisfactorily in many applications. Compatability from the standpoint of electrolytic corrosion could be one requirement. Alloy 2213 is generally specified where rivets are to be used at elevated temperatures; however, this probably would not apply in the light aircraft field.

It is considered poor practice to use a large rivet in relatively thin metal or a small rivet in thick metal. Furthermore, a loss in shear strength can result when a relatively soft rivet is driven in a hard, thin plate. Tests indicate reductions in shear strengths of approximately 30 percent when the rivet diameter is four times greater than the sheet being joined.

The type of rivet to be driven generally governs the selection of the driving method. All standard rivets require backing up, pressure, or impact, and a driving-set or head-forming fixture. Blind rivets require special tools. Common practice is to drive solid aluminum rivets with either squeeze riveters or pneumatic hammers. The cup in a rivet set must conform to the style of the manufactured rivet head. Bucking bars or pneumatic backups used in hammer riveting should have sufficient force to counteract the hammer blows.

TABLE XVII
ALUMINUM - SATISFACTORY COMBINATIONS
OF STRUCTURAL AND RIVET ALLOYS

STRUCTURAL ALLOYS	RIVET ALLOYS
1 xxx SERIES	1100
3 xxx SERIES	6053,6061
5 xxx SERIES	5056,6053,6061
6 xxx SERIES	6053, 6061, 7277
2 xxx and 7 xxx SERIES	2017,2024,2117,2219 6061,7075,7277
Magnesium Base	5056

Flush-riveted joints require countersunk head rivets. Either the manufactured or the driven head can be countersunk; however, in most instances the manufactured countersunk head is used. Countersinking the metal for flush rivets is done by machine countersinking in heavy gages, or by predimpling or dimpling in thinner gages, as is common in aircraft construction. In a predimpling operation, dies are used to press countersink the metal, whereas in dimpling, the rivet is used with a die. For some alloys, heated dies must be used. Countersinking can also be accomplished by spinning rather than pressing. Either technique used is influenced by the thickness and strength of the alloy, rivet size, hole diameter, and countersink angle.

It is important that all driving sets have smooth polished surfaces, so the metal can flow easily while being formed. As a rule the diameter of the driven head should not be less than 1.3 times the diameter of the original shank. The rivet length should be sufficient to fill the hole and form a satisfactory head.

Tubular, semitubular, and split rivets are usually driven with high-speed automatic or semi-automatic riveting machines.

Driving equipment required for blind rivets depends on the rivet type. The drive-pin type can be driven with an ordinary hammer; the explosive type requires a heat source such as a soldering iron. Most manufacturers of blind rivets provide the driving equipment needed.

Careful attention to details in rivet design and fabrication pays big dividends in fatigue life. When a fatigue failure occurs in a structure, it is usually at a point of stress concentration which could have been improved with little or no added expense.

To meet the requirements of large volume production demands, automatic riveting machines must be used to insure high quality with reasonable costs. Commercial and Military aircraft manufacturers have been using automatic riveting for more than five years. It has been estimated fatigue life is increased

RIVETING EQUIPMENT

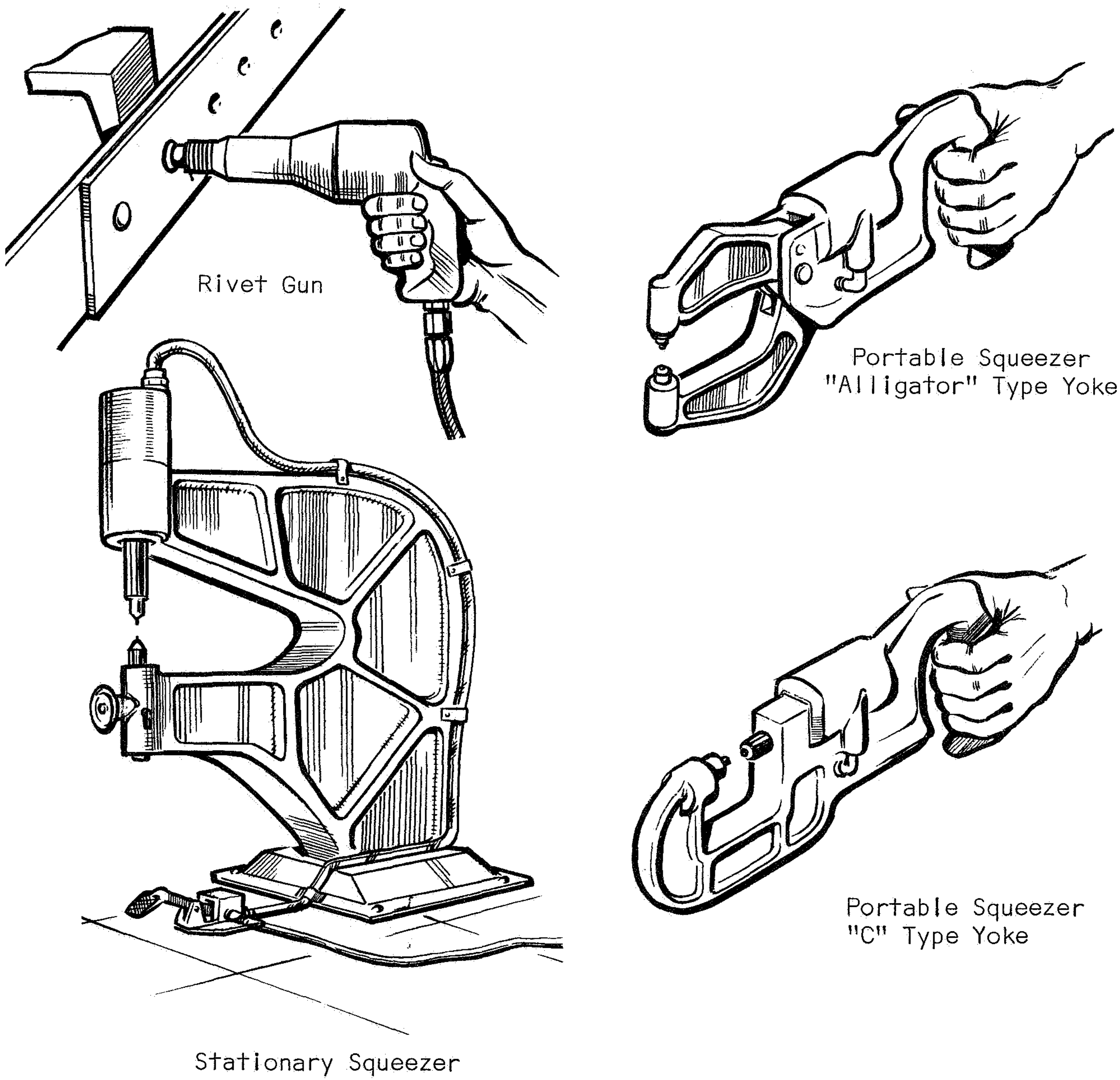


Figure 69

by approximately 200 percent over hand riveting. This increase is attributed to riveting uniformity, something impossible with hand riveting.

A large commercial aircraft manufacturer is installing one of the world's largest automatic riveting machines at its plant. Riveting will be performed at the rate of six seconds per rivet.

This machine is equipped with an automatic-sensing device, whereby riveting is performed to tolerances of 0.005 inch while maintaining consistent repeatability. Normality sensors automatically determine the contour of the wing surface; and guide the angle of the drill accordingly so all holes are exactly alike. All operations of this system are preplanned on perforated tape to automatically cycle from hole to hole while drilling, countersinking, pressure squeezing, impacting, and shaving the rivet to a smooth surface corresponding to the panel contour.

Automatic riveting machines can be set up to travel over the panel or remain stationary while the work, held in a fixture, moves past the machine.

The size and shape of the assemblies determine which method is more suitable. Tack rivets are used to temporarily fasten the sheets together, and later are replaced by permanent hand-driven types.

Design-allowable strengths.-The strength of a riveted joint is governed by the shear strength of the individual rivets, the bearing strength of the sheet, and the efficiency of the sheet in tension. Some typical ultimate shear strengths of single rivets are given in Table XVIII based on values shown in MIL-HDBK-5 (Strength of Metal Aircraft Elements). Due to the light loadings anticipated, joint strengths will probably be based on the bearing strength of the sheet, or the shear strength of the rivets.

TABLE XVIII

ALUMINUM RIVET ULTIMATE SHEAR STRENGTH (single shear in lbs)

Rivet Size / Sheet Gage	Protruding Head (3/32") AD3	Protruding Head (1/8") AD4	Dimpled Sheet 2024-T3,T42,T81 (3/32") AD3	Dimpled Sheet 2024-T3,T42,T81 (1/8") AD4	Countersunk Sheet 2024-T42 and higher Structural Aluminum (3/32") AD3	Countersunk Sheet 2024-T42 and higher Structural Aluminum (1/8") AD4
0.020	202		209	299	132	163
0.025	210		235	360	156	221
0.032	217	374	257	413	178	272
0.040		386	273	451	193	309
0.050		388		484	206	340

Sheet gage is thickness of thinnest sheet in a single shear application. Bearing strength of particular sheet used must also be checked.

A fatigue life comparison of a well designed riveted joint to several adhesive bonded joints is shown in the section on bonding in Figure 76. It appears, at least from this standpoint, better performance would be expected from a bonded joint; however, considering all the parameters (cost, reliability or quality control, production schedules, etc.), the automatic riveting concept could prove most worthy.

<u>Electric Welding</u>

Electric welding is often used in aircraft construction. It is the only welding method used for joining structural corrosion-resistant steel; and has been generally adopted for most aluminum alloys. Six basic resistance welding processes are commonly used with aluminum: spot, seam welding to make lap joints, upset and flash-welding for butt joining, percussion welding to attach studs to surfaces.

These processes are rapid and economically justified for high volume production. With proper material preparation consistent weld quality may be achieved automatically by the welding equipment. This technique is independent of operator skill; and one machine may be used to weld a range of thicknesses and sizes

<u>Spotwelding</u>. - Used primarily in shear applications; however, it is not recommended in the following areas:

(1) attachment of flanges to shear webs

(2) attachment of spar caps or shear web flanges to wing skin

(3) attachment of ribs to spars or shear webs

(4) at truss panel points in spars or ribs

(5) at junction points of stringers or stiffeners with ribs, unless a stop rivet is used

(6) at ends of stringers or stiffeners, unless a stop rivet is used

(7) on each side of a joggle, or wherever there is a possibility of a tension load component, unless a stop rivet is used

(8) splices exposed to the airstream should be so designed that flow of the airstream would not tend to pry it apart

Anodically treated surfaces cannot be spotwelded; consequently the faying surfaces of a spotwelded seam must be left unprotected prior to welding. The assembled parts are anodically treated or painted after welding. For this reason there is some doubt about the advisability of spotwelding aluminum alloys, other than 5052 or clad materials, if the assemblies are subject to severe corrosion. It is possible to spotweld through wet zinc-chromate primer applied to the faying surfaces.

A French aircraft company, has recently developed a series of light aircraft, using spotwelding quite extensively. This company set out to incorporate mass-production techniques, and in so doing reduced costs accordingly. The number of parts is reduced by using certain components in several applications. Standard joining techniques are employed in fabricating major subassemblies (wing section, forward fuselage section, aft fuselage section, etc.) These are mated on the final assembly jig as in an automobile assembly line.

Normal riveting is limited only to primary joints; whereas all the remaining connections are spotwelded with automatic welding machines. These machines are programmed with perforated tape to perform the complete welding operation; consequently the operator stands by and only takes over in the event of any malfunctioning.

Fuselage welding is performed in two stages. By using this fabrication technique, the main structural elements of the fuselage are welded by machine in about ten hours.

The fuselage consists of a forward section and tail cone joined by a riveted skin splice. The longerons also extend out from the rear section, and are spliced with rivets to the longerons of the forward section. This constitutes an all riveted primary joint. Figure 70.

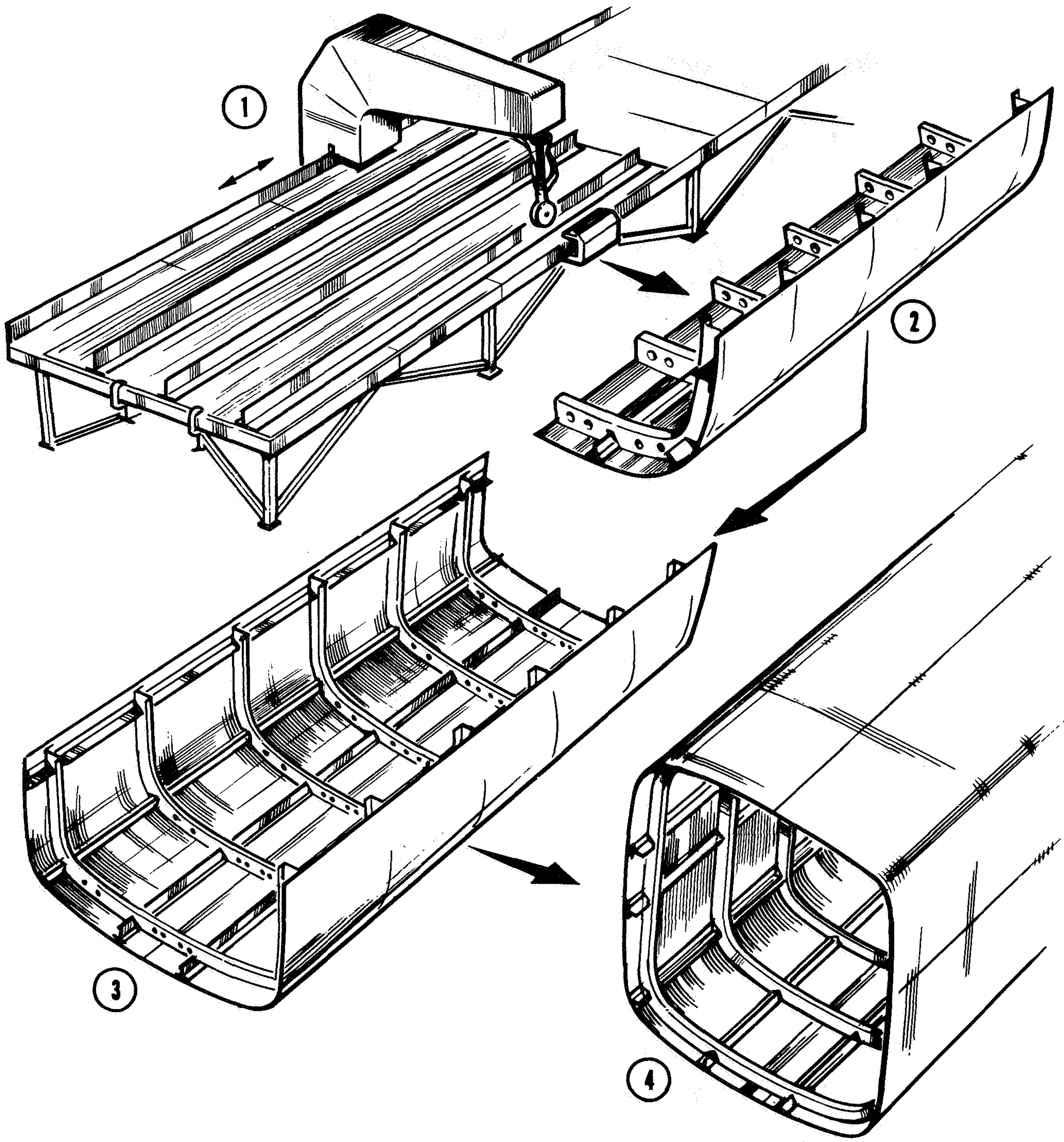
Fabrication of the wing is performed in a very similar manner whereby riveting is used only on the wing spars, ribs to stiffeners, stiffener to spar cap attachments, all these considered as primary joints.

The ailerons and flaps have identical profile. The skin is formed over the contour and spotwelded to the ribs and bent-up sheet metal longeron. (Ref Fig. 71) The trailing edge is constructed with beaded sheet metal skins spotwelded to the ribs, the longeron and at the trailing edge. This process applies to all movable surfaces on the aircraft.

Design-Allowable Strengths of Resistance Spotwelds: The strength of a spotwelded joint is governed by the shear strength of the individual spots, and the effect of the spotwelds on the tensile strength of the basic sheet. Therefore, both the shear strength of the spotweld, and the tension efficiency of the spotwelded sheet, must be considered in determining the strength of a spotwelded joint.

The allowable ultimate shear strengths of single spotwelds are given in Table XIX. Values are reproduced from MIL-HDBK-5. The allowable strength of a spotweld between two sheets of different material or thickness is the lower of the allowables for the individual sheets, as determined from the tables.

CURRENT LIGHT AIRCRAFT SPOT WELDED FUSELAGE CONSTRUCTION



The longitudinal stiffeners are first welded to the skins. The transverse members are then welded to the panel which is sufficiently flexible to be fitted into the second stage jig without any shaping.

Figure 70

CURRENT LIGHT AIRCRAFT SPOT WELDED FLAP CONSTRUCTION

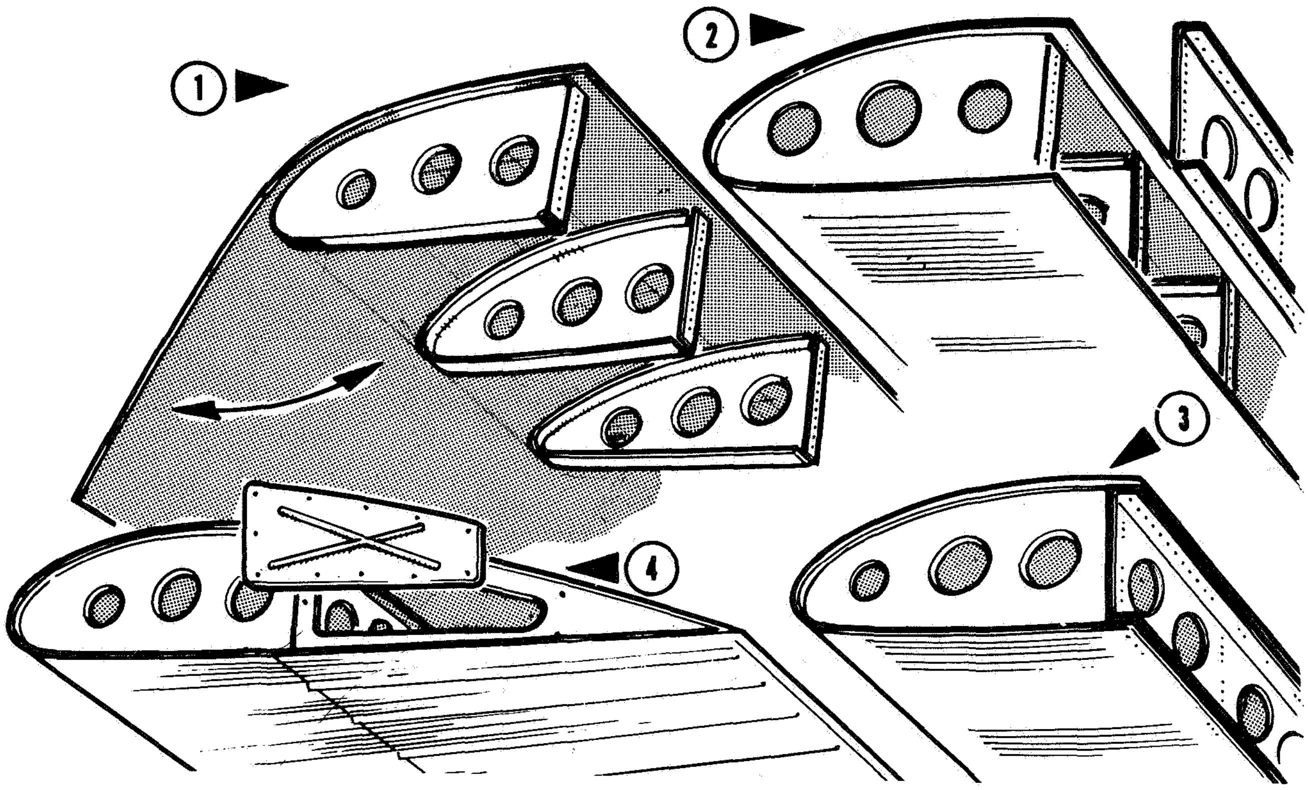


Figure 71

TABLE XIX

ALLOWABLE ULTIMATE SHEAR STRENGTHS OF SINGLE SPOTWELDS (ALUMINUM ALLOYS)
(Pounds per Spotweld)

Sheet Thickness (inches)	Aluminum Alloys, Clad or Bare Ultimate Tensile Strength of Material - psi			
	Below 19,500	19,500-27,999	28,000-55,999	56,000 & Above
	3003-0	3003-H14 5052-0	6061-T4 6061-T6	2024-All Tempers 7075-T6 7178-T6
0.012	16	24	52	60
0.016	40	56	80	88
0.020	64	80	108	112
0.025	88	116	140	148
0.032	132	168	188	208
0.040	180	240	248	276
0.050	236	320	344	372

Due to the anticipated light loadings involved with this type of aircraft, the joint strengths would be based primarily on the shear strengths of the spotwelds.

Seam welding. - Identical to spotwelding, except for the use of power-driven rollers as electrodes. A continuous airtight weld can be obtained at the rate of two to seven feet per minute by this method.

Some advantages of electric resistance spot and seam welding:

(1) Spotwelding is faster than riveting because no layout and drilling of holes are necessary. Numerous spotwelds can also be made in the time required to insert and head one rivet.

(2) Spot and seam welding do not add weight to the structure.

(3) Seam-welded watertight joints do not require the insertion of tape and a sealing compound. Weight and expense are saved.

(4) The drag of rivet heads is eliminated on exterior surfaces.

Butt welding. - Butt welding is applicable to almost all metals. The work to be welded is clamped in large copper jaws also serving as electrodes. One of the jaws is movable. At the proper time, pressure is applied to the movable jaw to bring the work in contact. When the electric current is applied after the parts are pressed together it is called upset butt welding. In flash welding the edges are brought close enough together to start arcing, and when they reach fusion temperature, the current is turned off and pressure is applied. All wrought alloys of solid cross-section up to about 0.5 square inch in cross-sectional area can be upset butt welded. Square-cut abutting surfaces, free of lubricant, are required for optimum welding results. Shearing or sawing the ends just before welding is adequate preparation.

Arc welding. - Arc welding is based on the heat generated in an electric arc. Variations in this process are metallic arc welding, carbon arc welding, atomic-hydrogen welding, inert-arc welding (heliarc), and multiarc welding.

Arc welding to a limited extent has been used for many years in aircraft fabrication. Probably the flexibility and general all-around good results obtained with gas welding retarded its extensive use; however, in recent years, its use is increasing rapidly as its economics and advantages become more apparent. In arc welding, the applied heat is more concentrated, resulting in a quicker welding with less expansion and warping as compared to gas welding. This makes it possible to hold closer tolerances on parts requiring machining after welding. An allowance of 1/16 inch is usually sufficient for most assemblies.

By using the heliarc (inert-arc) welding process, satisfactory welds may be made with aluminum, and if argon is used for a shielding gas, no flux is required. Dispensing with flux is a definite advantage because flux removal from aluminum welded joints is extremely important to avoid corrosion. Many types of welded joints cannot be made when using welding methods requiring fluxing. Corrosion-resisting steel as thin as 0.010 inch can be welded by this process. Steel, copper, and many alloys can be readily welded by this process.

Parent material weld allowables: Allowable ultimate tensile stress in alloy steels for material adjacent to the weld, when structure is welded after heat treatment, is shown in Table XX.

TABLE XX

ALLOWABLE ULTIMATE TENSILE STRESSES NEAR FUSION WELDS
IN 4130, 4140, 4340, OR 8630 STEELS

Section Thickness 1/4 Inch or Less	
Type of Joint	F_{tu} (ksi)
tapered joints of 30° or less	90
all others	80

For alloy steel members subjected to bending, the allowable modulus of rupture when welded after heat treatment should not exceed the F_b equivalent to that for steel having a F_{tu} = 90,000 psi.

Strength of Weld Metal (Welding Rods)

Table XXI indicates allowable weld metal strengths for various steels. These are based on 85 percent of respective minimum tensile ultimate test values.

TABLE XXI

WELD METAL STRENGTHS FOR WELDED JOINTS (Welding Rods)

Material	Heat Treatment After Welding	F_{su} ksi	F_{tu} ksi
Carbon and alloy steels	none	32	51
		32	51
Alloy steels	none	43	72
Alloy steels	stress relieved	50	85
Alloy steels	stress relieved	60	100
Steels	quench & temper		
4130	125 ksi	63	105
4140	150 ksi	75	125
4340	180 ksi	90	150

Welding Considerations

There are many general considerations all designers should be familiar with in designing welded joints. The following apply particularly to arc welding.

(1) Straight tension welds should be avoided because of their weakening effect. When a weld must be in tension, a fishmouth joint or finger-patch should be used to increase the length of the weld and to put part of it in shear.

(2) A weld should never be made all around a tube in the same plane. A fishmouth weld should be made. This situation arises frequently when attaching an end fitting to a strut.

(3) Two welds should not be placed close together in thin material. Cracks will result because of the lack of metal to absorb shrinkage stresses.

(4) Welds should not be made on both sides of a thin sheet.

(5) Welds should not be made along bends, or cracks will develop in service.

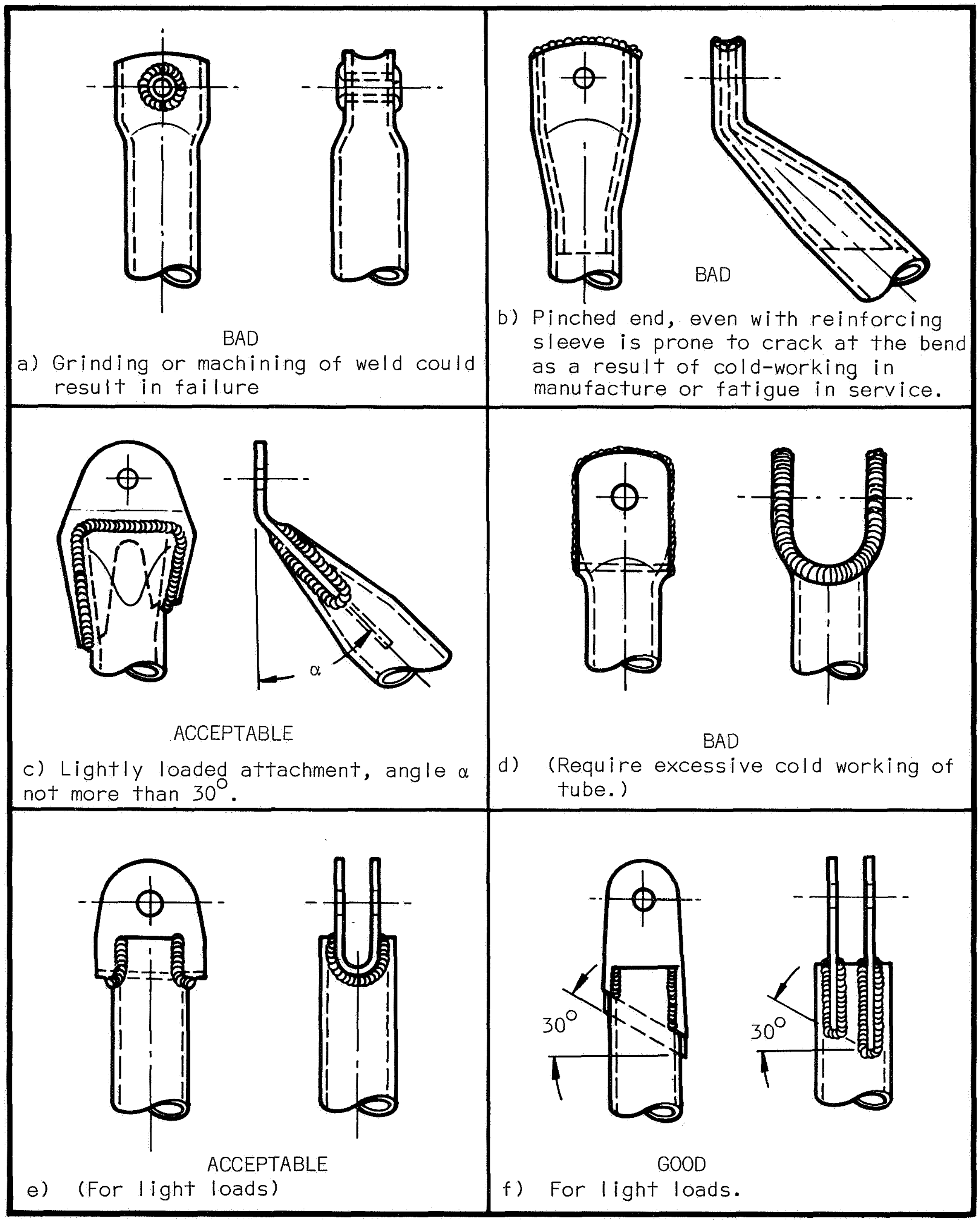
(6) Welded reinforcements should never end abruptly. The sudden change of section will result in failures by cracking when in service.

(7) Aircraft bolts should never be welded in place unless they are made of weldable material and are going to be welded to a similar metal. Furthermore, welding will destroy the heat-treated condition of the bolt. This has to be considered in the design/analysis. The same comments are valid for aircraft nuts. However, when required, tack welding in three places is usually all that is necessary to position them.

(8) When possible, welded parts should be normalized or heat treated after completion, to refine the grain and relieve internal stresses caused by shrinkage.

If welded parts are not normalized they could develop cracks in service, particularly if subjected to vibrational stresses. This is because weld material is cast metal lacking the strength, ductility, or shock resistance of wrought metal. The internal stresses are also seeking to adjust themselves. Sharp bends or corners, or rapid changes of section in the vicinity of welds are especially liable to cracking.

In the design of tubular joints, care should be taken to make all welds accessible. Figure 72 illustrates industry accepted design practices. These configurations provide proper stress distributions through the joints and should be followed as much as possible.



BAD
a) Grinding or machining of weld could result in failure

BAD
b) Pinched end, even with reinforcing sleeve is prone to crack at the bend as a result of cold-working in manufacture or fatigue in service.

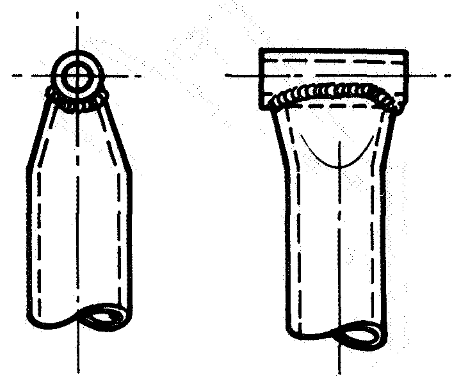
ACCEPTABLE
c) Lightly loaded attachment, angle α not more than 30°.

BAD
d) (Require excessive cold working of tube.)

ACCEPTABLE
e) (For light loads)

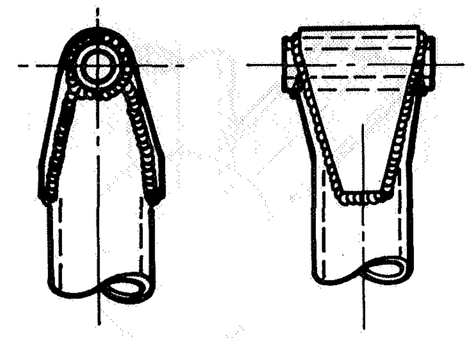
GOOD
f) For light loads.

Figure 72



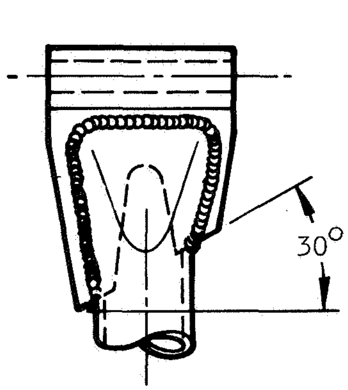
LIGHT LOADS

g) (Not very strong in compression due to crushing under spacer tube).



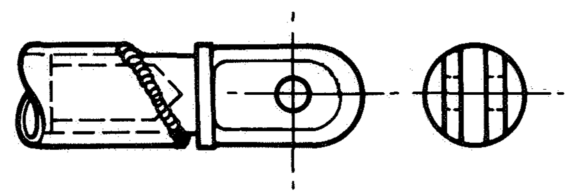
LIGHT LOADS

h) (This is an improvement on "g" Provide enough material on bushing to avoid machining of weld).



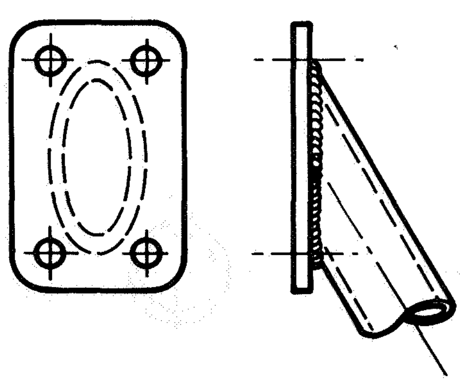
HEAVY LOADS

i) Stagger termination of welds at opposite sides of tube.

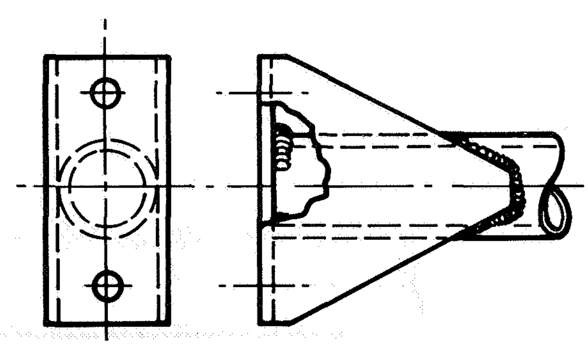


ACCEPTABLE

j) Welding of standard clevis

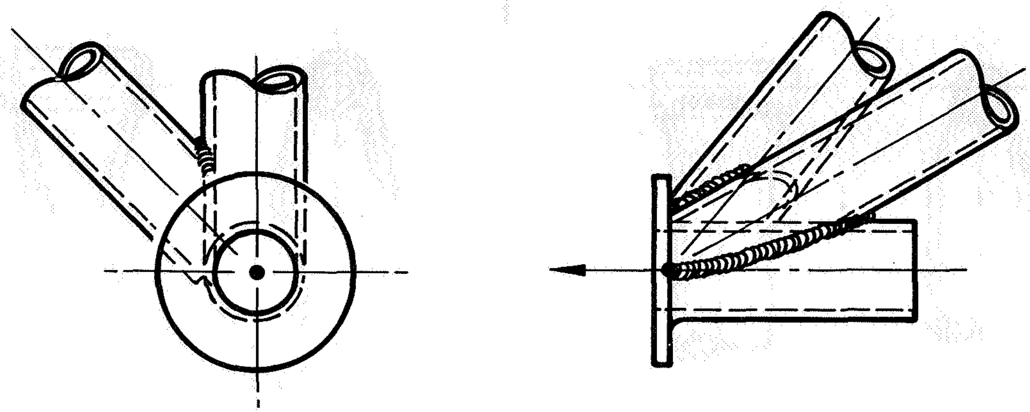


k) Satisfactory for a fixed end attachment



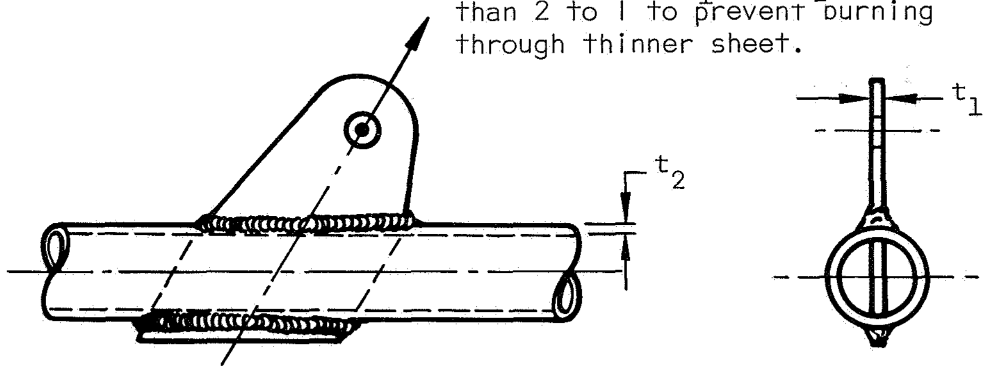
l) Satisfactory for a fixed end attachment

Figure 72-Continued.



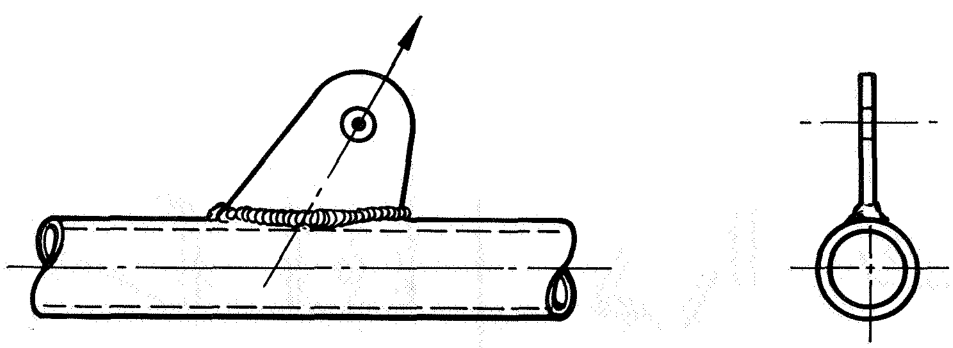
m) Minimize eccentricity between tube center lines and loads.

Note: applicable to any weldment: Keep thickness ratio between two welded parts (t_1 and t_2) less than 2 to 1 to prevent burning through thinner sheet.



HEAVY LOADS

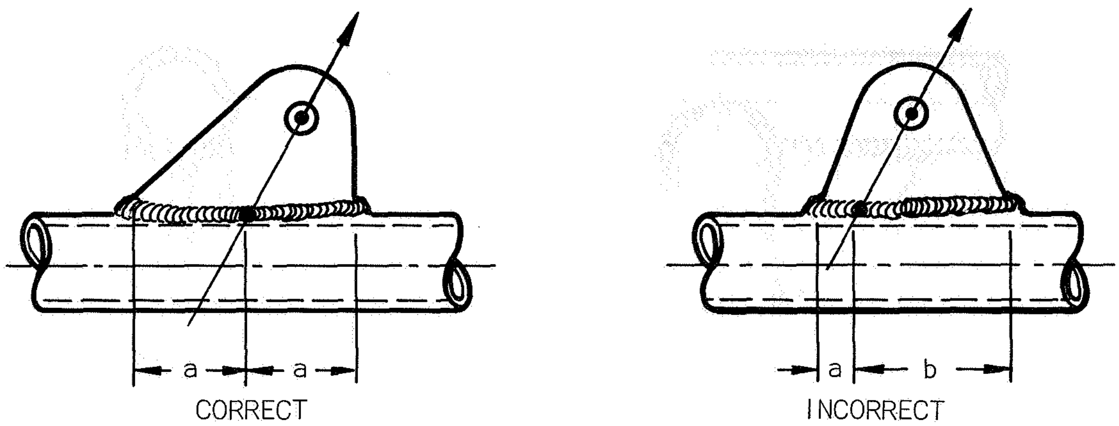
n) Fitting plate attachment.



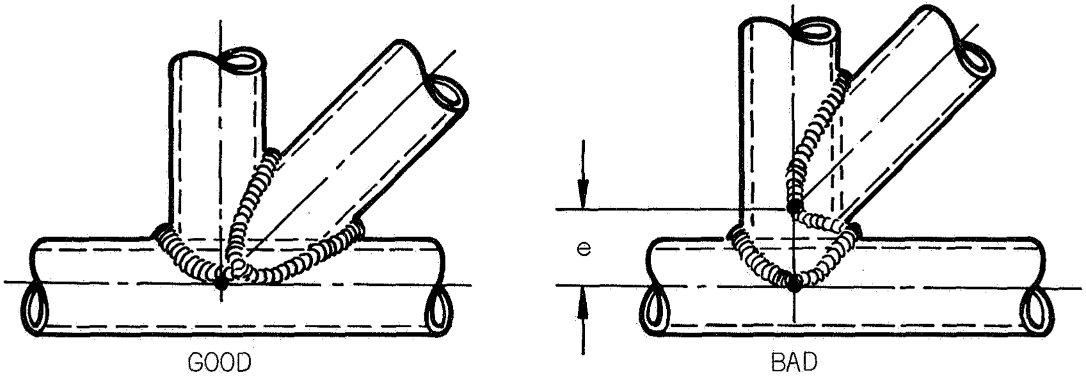
LIGHT LOADS

o) Fitting plate attachment.

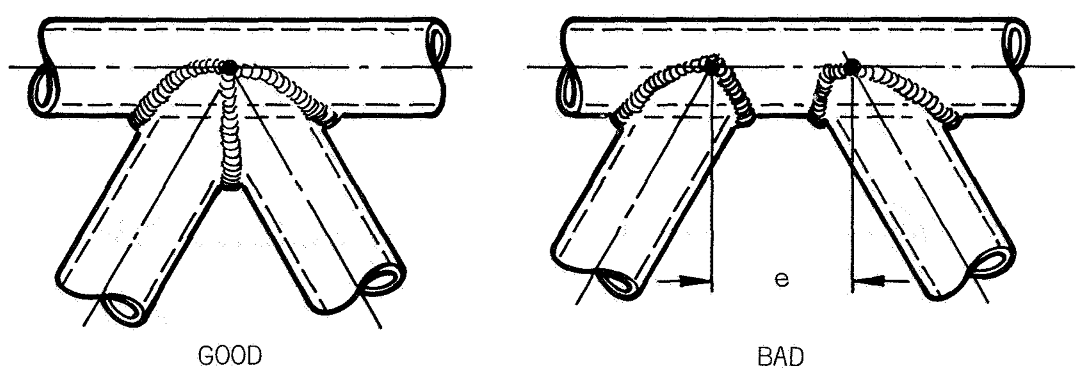
Figure 72 -Continued.



p) Fitting plate attachment

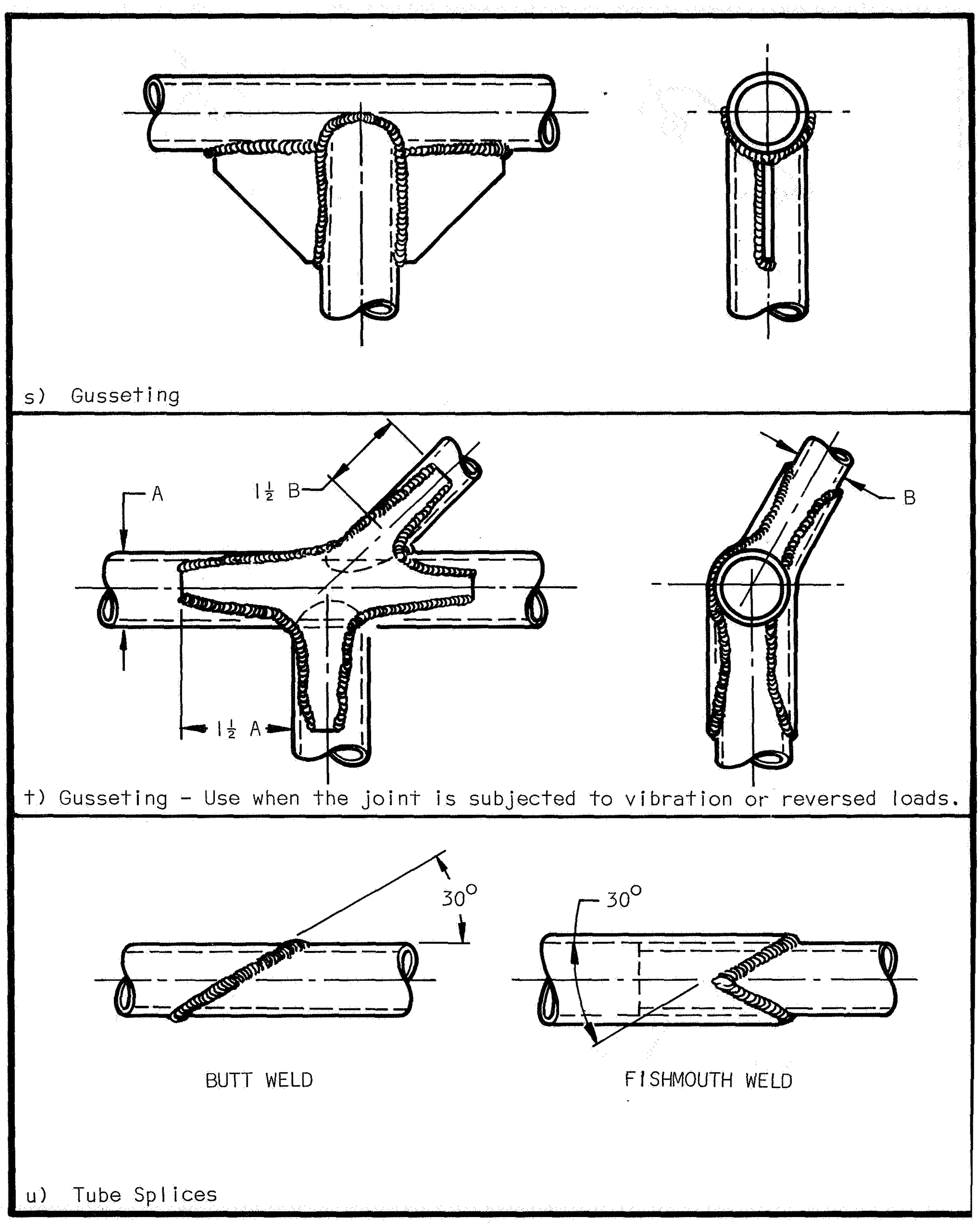


q) Tube Cluster - (Keep eccentricities to minimum)



r) Tube Cluster - (Keep eccentricities to minimum)

Figure 72 -Continued.



s) Gusseting

t) Gusseting - Use when the joint is subjected to vibration or reversed loads.

u) Tube Splices

Figure 72 -Concluded.

Brazing

Brazing is a method of metal joining, using a filler metal having a melting temperature less than the parent material being joined. Brazing is primarily used for joining assemblies for use at normal atmospheric or slightly elevated temperatures because the usual brazing alloys are compositions which soften readily at relatively moderate temperatures. Brazing is distinguished from soldering by the melting point of the filler metal (filler metal for soldering has a much lower melting point), and differs from welding in that no substantial amount of the base metal is melted. Thus, the temperatures for brazing are intermediate between those for welding and soldering. The strength and corrosion resistance characteristics of a brazed assembly also generally fall between those of welded and soldered assemblies.

Brazing aluminum.-Nonheat-treatable wrought alloys brazed most successfully are the 1xxx and 3xxx series, and the low-magnesium 5xxx series. Alloys containing a higher magnesium content are more difficult to braze by the usual flux methods, because of poor wetting by filler metal and excessive penetration. Filler metals are available that melt below the melting temperature of all commercial-wrought nonheat-treatable alloys.

Of the heat-treatable alloys, those most commonly brazed are the 6xxx series. The 2xxx series may be brazed quite satisfactorily; however, the 7xxx series is low melting and, therefore, not normally brazeable, with the exception of 7075 and x7005.

Material Combinations - Aluminum

(1) It is desirable from a production standpoint to design assemblies in their entirety from 2xxx or 3xxx alloys, or combinations of these two materials.

(2) Combinations of alloys (2xxx to 61xx, 2xxx to 53xx, etc.) are difficult to braze and should be avoided.

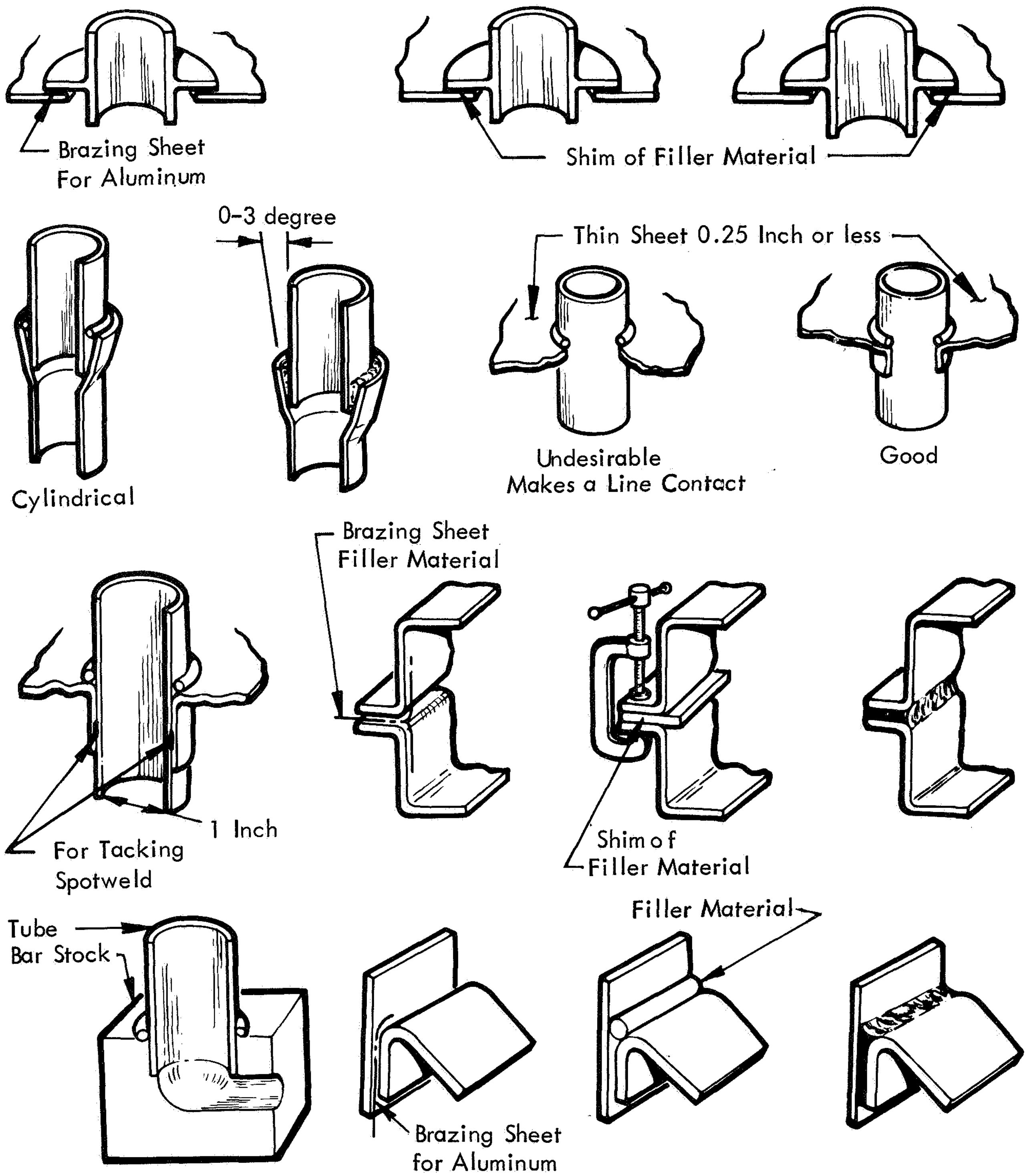
(3) Combinations of 61xx or 53xx to 61xx are satisfactory.

(4) Brazing sheets must be used in combination with 2xxx or 3xxx alloys only.

Brazing sheets should be used where a large number of joints are necessary in flat or formed sections of sheet; or possibly for ducts, tanks, or other assemblies where it would be difficult to secure wire or other forms of filler material adjacent to the joints. This material would also be used in an area requiring brazing in a position other than one allowing gravity flow of the filler material. Typical examples of brazed joints are shown in Figure 73.

Brazing steel.-Joining steel parts into single units may be done by brazing with copper or silver alloys.

TYPICAL EXAMPLES OF BRAZING



Duct and Tank Applications

Figure 73

When copper alloys are used, brazing is performed within a furnace (copper furnace brazing), having a controlled heat of 2050°F. This is above the melting point of copper (1985°F); therefore this may be accomplished by induction, torch, resistance, furnace, or dip methods.

The selection of the brazing method depends upon the materials involved, the shape and size of the parts, whether heat treatment after brazing is required, the number of parts, etc.

Materials for brazing steel: Most steels may be brazed by either method; however, corrosion-resistant steel may not be copper-furnace brazed. Only the stabilized grades of 18-8 stainless steel (321 and 347) can be silver brazed as the temperatures involved impair the corrosion resistance of the unstabilized grades (302 and 303). The physical properties of heat-treated and cold-worked materials are reduced by the temperatures required for brazing.

Heat treatment may be performed on copper-furnace-brazed assemblies; however, due to the low melting point of the silver alloys, it is not possible to heat treat steel assemblies after silver brazing has been performed.

Fusion welding after brazing is normally prohibited within three inches of a brazed joint.

The same general design guide illustrated for various joints in Figure 73 should also be used for steel materials.

Allowable stresses.-F_{su} = allowable ultimate shear stress for the brazed area = 15000 psi (this applies to all conditions of heat treatment for all applicable materials).

Because of decarburation occurring during brazing, the strength of the parent material in most cases is reduced as follows:

TABLE XXII

EFFECT OF BRAZING ON ALLOWABLE STRENGTH

Material	Allowable Strength
heat-treated material including normalized used in as-brazed condition	mechanical properties of normalized material
heat-treated material (including normalized) reheat-treated during or after brazing	mechanical properties corresponding to heat treatment performed

Advantages of brazing.-

(1) parts too thin to weld may often be brazed.

(2) heavy sections may be joined to thin sheets.

(3) warpage and distortion are reduced.

(4) brazed joints are vacuum tight.

Disadvantages of brazing.-

(1) assemblies made of 2xxx and 3xxx aluminum alloys are fully annealed during brazing, and cannot be restored to the original hardness; steels must be heat treated again to obtain original strengths.

(2) series 53xx and 61xx aluminum alloys must be heat treated and artificially aged after brazing to obtain the condition required.

(3) brazed assemblies cannot be put into the furnace for a second brazing unless there is a filler material with a lower melting point than used in the previous brazing.

(4) resistance to corrosion of aluminum alloys generally is not impaired by brazing; however, if flux is not completely removed, the residue will cause corrosion (interdendritic attack on the fillets, and intergranular attack on the base metal); if flux is not removed, it causes rapid pitting in the presence of moisture.

(5) when two aluminum alloys are brazed together, exposure to salt water or some other electrolyte may result in attack on the more anodic part; this condition is aggravated if the anodic part is relatively small compared to the other piece.

(6) furnace brazing causes a certain amount of diffusion of a clad surface reducing its corrosion resistance; Brazing Sheet No. 100 must be used for such applications (filler metal on one side and a special alclad alloy on the other side).

Applications of brazing.-

(1) Controls and mechanisms for:

(a) accessories.

(b) electrical system.

(c) fuel and oil system.

(d) heating, ventilating, and de-icing systems.

(e) power plant controls.

(f) hydraulic equipment.

(2) Supports and attachments for:

(a) accessories, instruments, radio, etc.

(b) antenna masts and housings.

(c) pitot masts.

(d) landing gear doors or entrance doors.

(3) Miscellaneous.

(a) landing gear up-lock systems.

(b) handles (assist, door, pump, seat adjustment, etc.)

<u>Bonding</u>

Many times, adhesives are called the modern tool for joining assemblies; however, the only modern aspect is that bonding agents have been greatly improved. There is much historical precedent associated with this technique back to the era when wood aircraft structure was first glued together. The old Mosquito bomber of the early 1940's used plywood wings bonded with wood glue.

Although much research was conducted prior to 1940, the initial successful adhesives were not developed until the early 1940's. A group of phenolic resin-synthetic rubber hybrids were developed by one United States automobile manufactuer which maintained high strength over a wide range of temperatures. About this same time an adhesive manufacturing company in England was experiencing success with an adhesive formulation based on a phenolic resin-polyvinyl combination.

The American developed adhesives were single component systems which could be easily applied with simple tools (brush, roller, etc.), whereas the British system was a more sophisticated two-part system. With this process, it was necessary first to apply a liquied phenolic resin to the adherends,followed by a layer of powder over the liquid film. The powder, a polyvinyl formal, developed the necessary toughness or elasticity in the bonded joint, while the phenolic resin provided the proper adhesion characteristics.

Due to the apparent simplicity in applying the single-component system, further development of these adhesives were more closely followed in the United States and abroad.

Coincidental with the development of these newer adhesives, the airplane was playing a mojor role in the fighting of World War II. The aircraft industry was, therefore, desperately in search of unique manufacturing techniques to save weight or provide smoother airfoil surfaces. This urgency led to the immediate acceptance of adhesive bonding for use in aircraft structure. In the United States, the government approved the single component adhesive system as an aircraft structural bonding agent while England began utilizing the double component system for joining metal to wood in the De Haviland Hornet.

Within a few years, vinyl-phenolic bonded-sandwich structures became more predominant for use in wing panels and fuselage sections of the B-57 and Matador missile. By the mid 1950's, structural adhesive bonding was used extensively in the manufacturing of the B-58. Since then, new epoxy adhesive systems have been used more consistently and more daringly. Bonding of aluminum to itself, and to other metals and non-metals, has become common practice. Because of the great potential in weight reduction, the major technical effort to develop reliable adhesive bonding data has been restricted to aluminum alloys used in aircraft such as bare and alclad 2020-T6, 2024-T3, T6, T86, and 7075-T6.

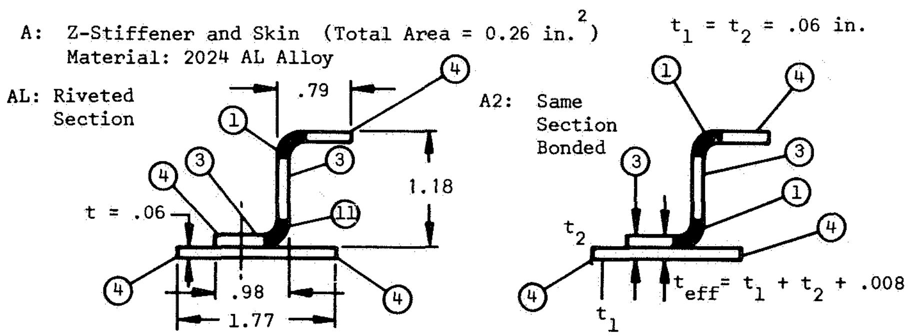
A dramatic example in present-day application of adhesive bonding is the supersonic F-111 fighter-bomber. Most of the entire exterior skin is an adhesive-bonded honeycomb-sandwich structure. Another prime example of complex bonded structures being made today is associated with helicopter rotor blades. The Bell Helicopter (model UH-ID) uses an adhesive to bond an aluminum honeycomb core and doublers to the main spar, a brass nose bar, and a stainless-steel leading edge. This 22-foot long all-bonded assembly is cured at 120 psi and 350 degrees F.

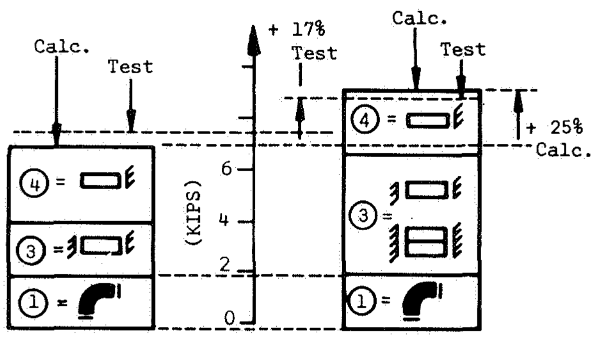
It is apparent that adhesive bonding has a definite place in the aircraft industry. The crippling strength of compression panels is significantly improved due to the integral stiffening effect of the bonded laminates (ref. Fig. 74).

The fatigue strength of compression panels is increased thru the use of good bonded design. Figure 75 compares three configurations and reveals that the one with insufficient skin width to stringer bond is inferior to the riveted configuration beyond 10^4 load cycles thus demonstrating the importance of proper bonded design.

Fatigue strength comparisons of Redux bonded single and double lap joints with a riveted joint are made in Figure 76. Here again, the superiority of well designed bonded joints is evident. Results of box beam fatigue tests involving riveted, bonded, and integrally stiffened construction are presented in Figure 77. The advantage gained by using scarf joints in lieu of lap joints is shown in Figure 78 where the S-N curves for both configurations are plotted.

COMPARISON OF CRIPPLING STRENGTH OF BONDED AND RIVETED BUILT-UP COMPRESSION ELEMENTS



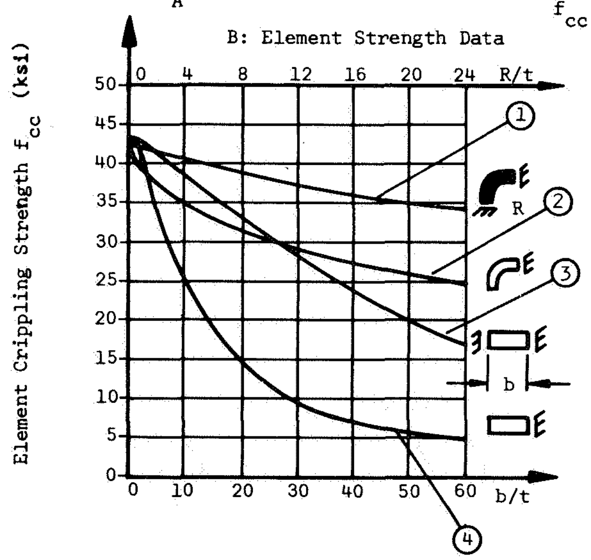


Relative Strength

Calculation of Crippling Strength

$$F_{cc} = \frac{\sum_{1}^{n} a \cdot (f_{cc})}{A}$$

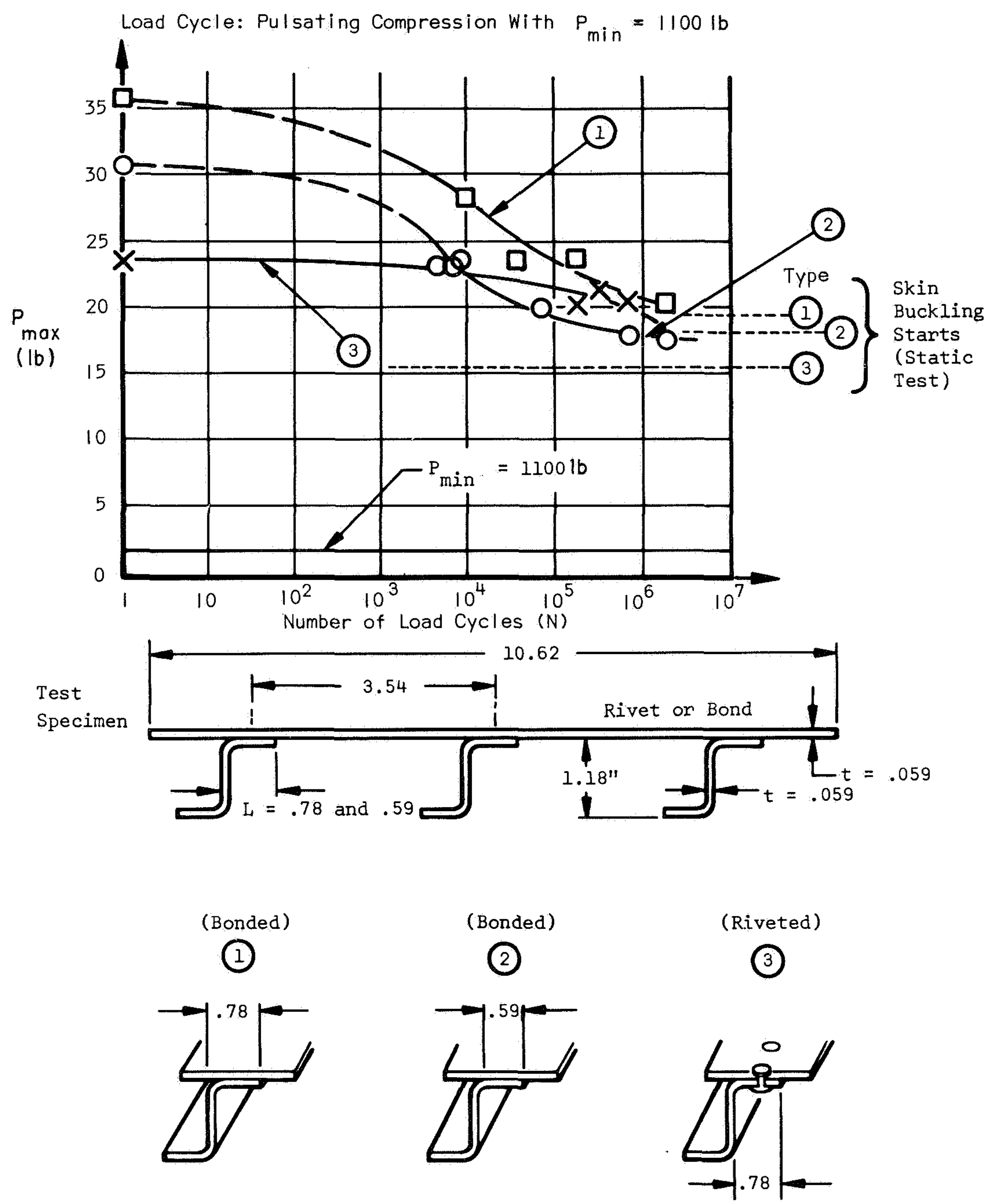
Where: a = Area of individual elements
A = Total area of section
f_{cc} = Crippling stress of elements according to above curves.



Data Extracted from Article Written by L. Jungstrom; <u>Design Aspects of Bonded Structures</u>; Bonded Aircraft Structures Published by C.I.B.A. (A.R.L.) Limited; 1957

Figure 74

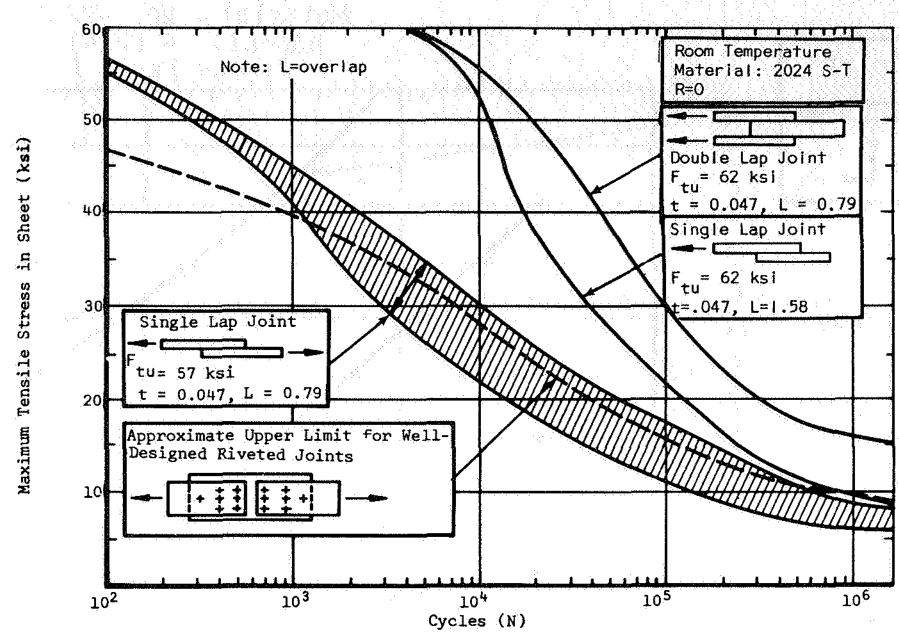
EFFECT OF WIDTH OF SKIN TO STRINGER BOND ON
FATIGUE STRENGTH OF COMPRESSION PANELS



Data Extracted from Article Written by O.L. Jungstrom; Design Aspects of Bonded Structures; Bonded Aircraft Structures Published by C.I.B.A. (A.R.L.) Limited 1957

Figure 75

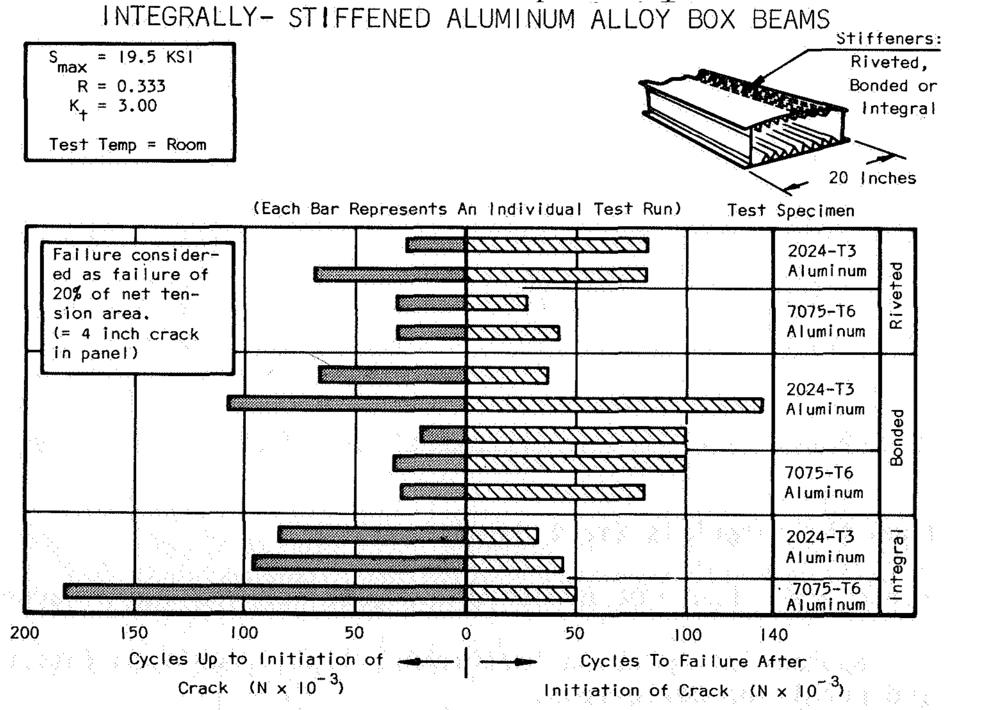
COMPARISON OF FATIGUE STRENGTH OF BONDED SINGLE- AND DOUBLE-LAP JOINTS WITH A RIVETED JOINT



Data taken from: FFA Report HU-226 and FFA Medd. No 30

Figure 76

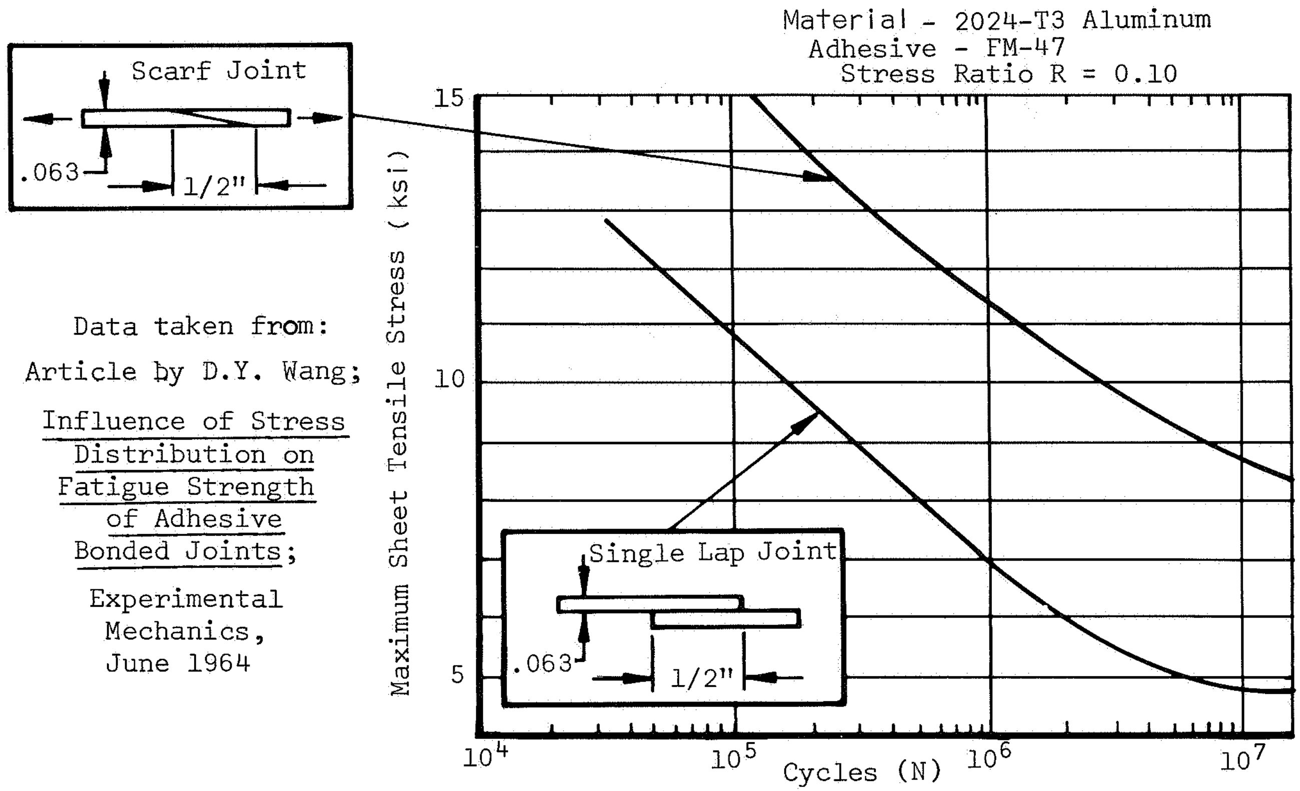
COMPARISON OF RIVETED, BONDED, AND INTEGRALLY- STIFFENED ALUMINUM ALLOY BOX BEAMS



Data Extracted from NACA-TN-3856, August 1956; Fatigue Crack Propagation in Aluminum-Alloy Box Beams

Figure 77

COMPARISON OF FATIGUE STRENGTH OF A SIMPLE LAP JOINT AND A SCARF JOINT



Data taken from:
Article by D.Y. Wang; Influence of Stress Distribution on Fatigue Strength of Adhesive Bonded Joints; Experimental Mechanics, June 1964

Figure 78

Higher strength-to-weight ratios are possible with sandwich materials. Often it is the only way to join thin-gage sheets; the adhesive bond can double as a seal; dissimilar metals can be fastened without corrosion effects and irregular shapes or complex sections can be fastened comparatively easily. Helicopters, for example, because of vibration, require the damping provisions provided by the nitrile rubber-epoxy adhesive system. Table XXIII lists the many advantages as well as the limitations occurring through the use of bonded structures.

General design and production philosophy associated with bonded structures.-

(1) Know the materials (test data).

(2) Structures should be properly designed for the use of adhesives.

(3) Use appropriate prebond treatments, tightly written instructions, and permit no deviations.

(4) Insist that the recommended process or specifications be rigidly adheared to when:

(a) Applying and curing the adhesive.

(b) Handling, fitting, and jigging of the parts.

(5) Train personnel to understand the importance of good workmanship and its influence on joint strength and life.

(6) Set up a quality-control system to maintain a high standard of reliability. Destructive test specimens should be frequently processed concurrently with production bonds.

Initial strength of a joint does not constitute a good reliable bond which will satisfy its intended service life. The adherend surface preparation is an important prerequisite in the permanence of joints subjected to simultaneous stress and adverse environment. Joints made with poorly prepared adherends may exhibit the same initial breaking strength as those made with adherends having undergone an elaborate chemical cleaning process. The bonds made with the minimum surface treatment, however, will prove inferior with respect to permanence. Elaborate metal-cleaning procedures might be alleviated by using a pre-priming operation incorporated in the material production line at the mill. This method is already used by a honeycomb panel manufacturer in the United States. A primer is applied to both surfaces of sanwich facing material, accomplishing the following:

(1) provides proper substrate for primary honeycomb bonding

(2) maintains clean surface for a later secondary bond if necessary

(3) primer acts as an additional corrosion-resistant barrier to all exposed surfaces of the adherend whether or not a secondary bond is made

This process could easily be incorporated as an additional step at the mill; however, the basic material cost could increase as much as 20 percent.

Repairs for bonded construction - Repairs to damaged panels and surfaces might be necessary either during production or after they are in service for some time. Consequently, effective repair methods must be developed to maintain the original contour, insure structural integrity, and prevent damage propagation.

Repairability requires: (1) the damaged part, dependent upon the extent of the damages, must be removable, if necessary, by some means that will leave the remaining parts undamaged; (2) the damaged part must be capable of being repaired, using mechanical fasteners, adhesive bonding, or a combination of both, without loss of properties to the remaining bonds.

Quite often repairs are made with materials differing from the material of the damaged structure. Therefore, a repair adhesive must be capable of

TABLE XXIII

ADVANTAGES AND LIMITATIONS OF BONDING

DESIGN FACTOR	ADHESIVE BONDING ADVANTAGES	LIMITATIONS
Aerodynamic Smoothness	Smooth exterior contours greatly improved.	
Cost	Savings achieved through bonding of large assemblies which have been properly designed for bonding or by weight savings.	Special tools and facilities are required for contoured parts.
Corrosion of Dissimilar Material Joints	Versatility of joining dissimilar materials is greatly improved. Corrosion in faying surfaces is reduced. Metals may be readily joined to non-metallics.	Differential coefficient of expansion must be considered due to the build-up of residual stresses.
Stress Concentration	More uniform distribution of stress through a bonded joint along entire length. Greatly reduces stress concentration.	Residual stresses may be induced during heat cure.
Fatigue Resistance	Great improvement--10 to 1 over rivets. Reduces crack propagation.	
Static Strength	Adhesives exhibit high strengths when stressed in shear. The more efficient adhesives either approach or surpass the sheet metal strength at an L/t ratio between 20 and 30. L = Lap length; t = adherend thickness.	Production adhesives are generally limited to 350°F.
Design Factor Weight and Size	Reduction of weight and size may be obtained. Greater capability for joining thin or brittle materials. In properly designed bonded structures, the following weight savings could be achieved over riveted structures: (1) Compression members: up to 25 percent (2) Tension members: 10 to 15 percent (3) Tension members designed by fatigue criteria: up to 20 percent (4) Some miscellaneous weight may be saved by eliminating the necessary local reinforcements usually required with conventional fasteners. NOTE: A typical overall weight savings for civil aircraft is 3 to 6 percent of the total structure weight	
Production	Many details may be eliminated which simplifies the overall design. Large areas may be bonded in a single operation.	A close tolerance between mating parts is essential. Special skills and personnel training are usually required.
Inspection	Non-destructive test techniques are available to insure good reliability.	Extensive quality control must be exercised, since the strength level of bonded joints may not be fully determined through non-destructive testing.
Sealing	Internal fuel cells and pressurized cabins are automatically sealed when bonded.	Bacteria growth in fuel may attack the adhesive. Components may require additional protective coating in these areas.
Electrical Insulation	Excellent.	Jumpers are mandatory for electrical continuity.
Miscellaneous	Compared to welding, thermal damage to parent metals is greatly reduced. Field repair is easily performed.	Proper surface preparation is mandatory for good quality bonds. Work areas for bonding must maintain a high standard of cleanliness.
Experience	Adhesives have been successfully used on military and commercial aircraft for over 10 years.	

satisfactorily bonding a variety of materials, preferably under the same conditions of temperature and pressure. Another requirement for any repair adhesive must be that it displays an apparent forgiveness for less efficient cleaning methods in the field as compared to those used in the initial manufacture of the part. Regardless of whether the damaged assembly was made with a combined riveting and bonding technique, or by bonding alone, a repair can usually be made by using follow-up pressure-type mechanical fasteners. Another means of pressure application would be fabricated-in-place vacuum-bag blankets with portable vacuum pumps.

The following summarizes the main requirements of a repair adhesive:

(1) Since ovens, autoclaves, and special equipment will not be available at most field facilities, the repair adhesive must satisfactorily cure at near room temperature.

(2) It must also be capable of easy application within the temperature range of 40 to 100 degrees F.

(3) It must give good bond strength initially and after environmental exposure, for materials cleaned by methods not yielding the best possible surfaces for bonding.

(4) The effects of repeated cure on the original bond must not affect its integrity.

(5) It must withstand exposure to cleaning fluids used in service operations.

(6) It should have a good shelf life (at least 3 months), remain acceptable through a wide range of storage conditions, have at least 2 hours, and preferably 10 hours, of open assembly time.

CONCLUDING REMARKS

The cost analysis of an all plastic Far Term airplane, shown in Appendix A, and the comparative analysis of a conventional sheet metal aircraft with equivalent requirements, shown in Appendix C, indicates the obvious advantage of reduced labor. The reader should bear in mind that this illustration of cost analysis is based on several more-or-less arbitrary assumptions and statistics. Even with present day technologies, cost analysis is a mixture of art and science, often times tempered by personal experience.

APPENDIX A

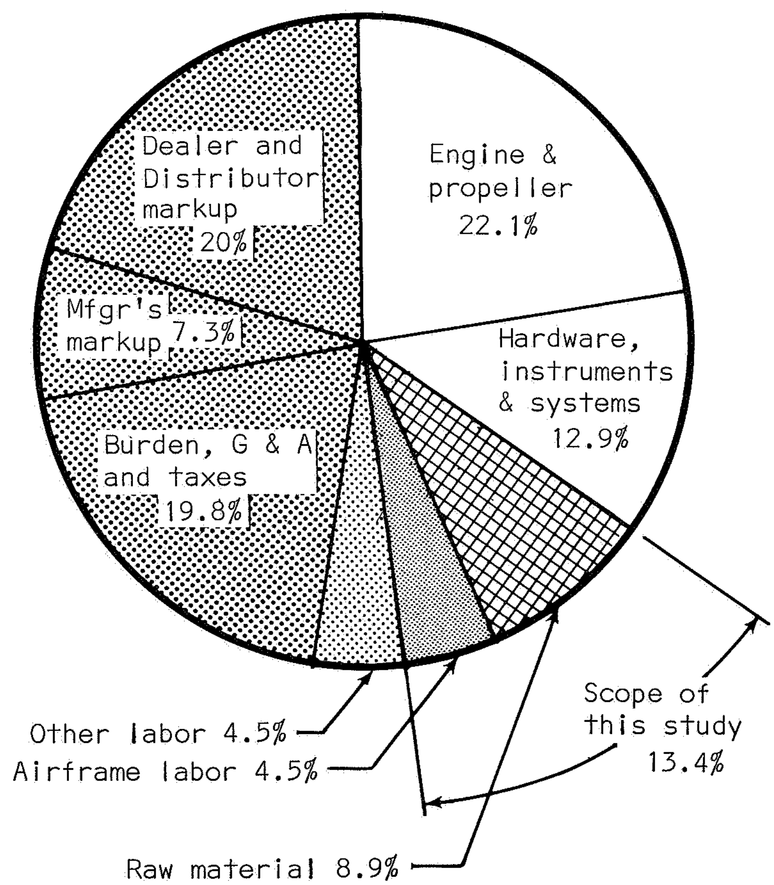
CONSUMER PRICE BREAKDOWN OF FAR TERM AIRPLANE (ESTIMATED)

An estimated total consumer price breakdown of the Far Term airplane has been determined by combining:

(1) the estimated costs of the primary structural components; i.e., the vertical tail, horizontal tail, wing and fuselage.

(2) the estimated cost of the remaining items such as burden, manufacturer and dealer markup, engine, hardware, etc., based in part on previous breakdowns of contemporary airplanes.

It is estimated that the reinforced plastic Far Term airplane of the 1980's, produced in six-figure quantities, will sell for approximately $10,973.00. A breakdown of this price is illustrated in the following pie chart.



The following table breaks the consumer price down in further detail. Following the table is a list of assumptions upon which the pie chart and the table are based.

CONSUMER PRICE BREAKDOWN FOR THE FAR TERM AIRPLANE

	Item		Dollars	Percent Total
(1a)	Direct labor (structure)	$ 238.62	$ 734.62	6.7
(1b)	Direct labor (other)	496.00		
(2a)	Overhead (structure)	310.21	955.21	8.7
(2b)	Overhead (other)	645.00		
(3a)	Material (structure)	811.25	978.25	8.9
(3b)	Material (retractable L.G., other)	167.00		
(4)	Molding time charge (not labor)		258.10	2.3
(5a)	Equipment (Engine & propeller	2425.00	3842.00	35.1
(5b)	(L.G., wheels, instruments, etc)	1417.00		
	Sub-Total		$ 6768.18	61.7
(6)	Direct, Sales, and G&A expenses		1211.92	11.0
(7)	Manufacturing cost		$ 7980.10	72.7
(8)	Factory profit (10% of Mfg. cost)		798.01	7.3
	Total dealer's cost		$ 8778.11	80.0
(9)	Distributor and dealer mark-up		2194.53	20.0
(10)	Total Estimated cost to consumer		$10972.64	100.0

AIRFRAME FABRICATION COST ANALYSIS

(11)	Airframe labor	$ 486.62
(12)	Airframe share of overhead	1434.64
(13)	Raw material	978.25
(14)	Molding time charge	258.10
(15)	Airframe fabrication cost	$ 3157.61

(16) AMPR weight is estimated to 1038 lbs.

(17) Unit airframe cost: $\frac{\$3157.61}{1038} = \$\ 3.02/\text{lb}$

Direct labor - (structure)	$ .72	fin labor	4 Plastic Parts	fab. time estimated at 4 min/part
	.90	rudder labor	5 Plastic Parts	
	2.88	horiz.stab.labor	16 Plastic Parts	
	36.36	wing labor	202 Plastic Parts	
	5.76	fuselage labor	32 Plastic Parts	
	192.00	other labor	(Estimated,other than plastic parts)	

$ 238.62 = @ $2.70/hr.ave.wage = 89 hours

Direct labor - (other)	$1700.00	(total direct labor from Table I)
	-1360.00	(airframe labor from Table I)
	$ 340.00	
	+ 156.00	(10% Table I $1560 est. for retractable L.G.)
	$ 496.00	

Overhead (structure) - \$ 310.21 = \$238.62 x 130% from Table I

Overhead (other) - \$ 645 = \$496 x 130%

Material (structure) -

\$ 8.27	vertical fin	(13.13 lb x .63 \$/lb)
5.36	rudder	(8.50 lb x .63 \$/lb)
79.12	horiz. tail	(36.06 lb x 2.00 \$/lb)
328.50	wing	(Fig. 55, bar 6; Fig. 56)
390.00	fuselage	(2.00 \$/lb x 195 lb of primary struc)
\$ 811.25		

Material (other) - \$ 167.00 = (\$70 est. for retractable L.G. material; \$97 est. for seats, upholstery, interiors, etc.)

Molding (time charge) -

\$ 13.20	vertical fin	$\left(\frac{1{,}320{,}000\ \$}{100{,}000 \text{ units} \times 4 \text{ parts}} = 3.30\$/\text{part Table XV}\right)$
16.50	rudder	(3.30 \$/part x 5 parts)
52.80	horiz. stab.	(3.30 \$/part x 16 parts)
70.00	wing	(estim., 202 parts, multicavity tooling)
105.60	fuselage	(3.30 \$/part x 32 parts)
\$ 258.10		

Equipment (Engine & propeller) - $\$2425.00 = 80\%^{*} \times \frac{250 \text{ HP}}{230 \text{ HP}} \times \$2795.00_{\text{Table I}}$

Equipment (L.G., etc.) - $\$1417.00 = \$1305_{\text{Table I}} \times 80\%^{*} + (24.2\%_{\text{Table I}} \times \$1560^{\Delta})$

Direct sales and G & A - \$1211.92 = Overall burden - Mfg. overhead = (2.95 from p.13 x labor) - \$955.21 = (2.95 x 734.62 - 955.21)

Distributor & dealer - \$2194.53 = dealer cost x 25% = 8778.11 x .25 (used 25% instead of 33% due to high volume sales, e.g. auto industry)

Airframe labor - $\$ 486.62 = 238.62 + \frac{1}{2} \times 496$ (Airframe labor = Direct structural labor + $\frac{1}{2}$ other labor

Airframe share of overhead - $\$1434.64 = \frac{\text{airframe labor}}{\text{all labor}} \times$ (overhead + direct, sales, G & A expenses) $= \frac{486.62}{734.62} \times (955.21 + 1211.92)$

---

* Assumed quantity-price improvement 15 years hence.

Δ Estimated price of retractable landing gear and other additional equipment, 15 years hence (\$3000 x 52%). (52% = mass production factor as determined on page 15).

APPENDIX B

VALUE OF A POUND SAVED

To determine the worth of a pound saved on a light airplane, two contemporary light airplanes, having identical powerplants and cruise speeds but different gross weights, were compared (for a twenty-year service life and a 333-hours-per-year utilization rate).

Airplane B is 140 pounds lighter than airplane A, by virtue of a greater design effort expended on a greater quantity of individually lighter detail parts. Additionally, these parts are likely made from structurally more efficient, and more expensive, materials.

The direct operating cost of the heavier airplane A is \$0.09 more than that of the lighter airplane B, due entirely to the greater fuel consumption of airplane A. See Table XIX.

The indirect operating costs of the lighter airplane B are greater since they are identical respective functions of a higher consumer price.

The higher consumer price of the lighter airplane B is solved for by equating the total operating costs for the two airplanes, in terms of consumer price for airplane B. I.e.,

Assuming: No interest after 5 years and,
Depreciating to 5% (scrap value),

$$(T.O.C.)_A = (T.O.C.)_B$$

$$\left.\begin{array}{r} 20\text{yr}(333\text{hr} \times \frac{\$9.57}{\text{hr}} + .0477 \times \$17000 + \$695) \\ + 5\text{yr}(.12 \times \$17000 + .044 \times \$17000) \\ + 15\text{yr}(.35/15\text{yr} \times \$17000) \end{array}\right\} = \left\{\begin{array}{l} 20\text{yr}(333\text{Hr} \times \frac{\$9.48}{\text{hr}} + .0477 \times \text{price} + \$695) \\ + 5\text{yr}(.12 \times \text{price} + .044 \times \text{price}) \\ + 15\text{yr}(.35/15/\text{yr} \times \text{price}) \end{array}\right.$$

$$\left.\begin{array}{r} \$\ 93\ 854 \\ 13\ 940 \\ 5\ 950 \\ \hline \$113\ 744 \end{array}\right\} = \left\{\begin{array}{l} .954\ \text{price} + \$77\ 037 \\ .820\ \text{price} \\ .350\ \text{price} \\ \hline 2.124\ \text{price} + \$77\ 037 \end{array}\right.$$

$$2.124\ \text{price} = \$113\ 744 - \$77\ 037$$

$$\text{Price} = \$17\ 284$$

$$\frac{\$284}{140\ \text{lb}} = \$2.03/\text{lb}$$

The price differential for airplane B, at which the higher indirect operating costs exactly compensate the lower direct operatings costs, represents the dollar amount that can be spent for its 140 pounds of weight saved. I.e., \$17284 - \$17000 = \$284 for 140 pounds saved, or \$2.03 per pound.

TABLE XIX - VALUE OF A POUND SAVED (for a 20 year service life)

SPECIFICATIONS	AIRPLANE A (HEAVIER)	AIRPLANE B (LIGHTER)
Weight Cruise speed Engine hp Fuel consumption Consumer price	3014 lb 150 mph 250 hp 11.47 gph $17,000	2875 lb 150 mph 250 hp 11.25 gph $XX,XXX (see below)
DIRECT OPERATING COSTS (HOURLY)		
Fuel and oil	$5.34	$5.25
Maintenance	$2.35	$2.35
Engine overhaul	$1.88	$1.88
Total D.O.C	$9.57/hr	$9.48/hr
INDIRECT OPERATING COSTS (YEARLY)		
Hangar rent	$480	$480
Insurance(4% + $215)	.04 x $17000+$215 = $895	.04 x Price + $215 =
Depreciation(5yr,40% residual)		
First 5 years	.12 x $17000 = $2040	.12 x price =
Last 15 years	.35/15 x $17000 = $ 397	.35/15 x price =
Tax ($7.70/1000 value)	.0077 x $17000 = $ 131	.0077 x price =
Interest (first 5 years only @ 80% x 5.5%)	.044 x $17000 = $ 748	.044 x price =
Total I.O.C.		
First 5 years	.1717 x$17000+$695	.1717 x price+$695 =
Following 15 yrs	.3607 x$17000+$695	.3607 x price+$695 =

Note that the $2.03/pound is for a 333 hours/year utilization rate and a 20-year service life. The worth of a pound saved is directly proportional to service life and utilization rate. Refer to Figure 40 for dollar value per pound saved, for service lives and utilization rates, ranging from five to thirty years and from 100 to 900 hours per year, respectively.

Additionally, the same process was repeated for both piston-powered and turbine-powered helicopters, the results of which are illustrated in Figures 79 and 80.

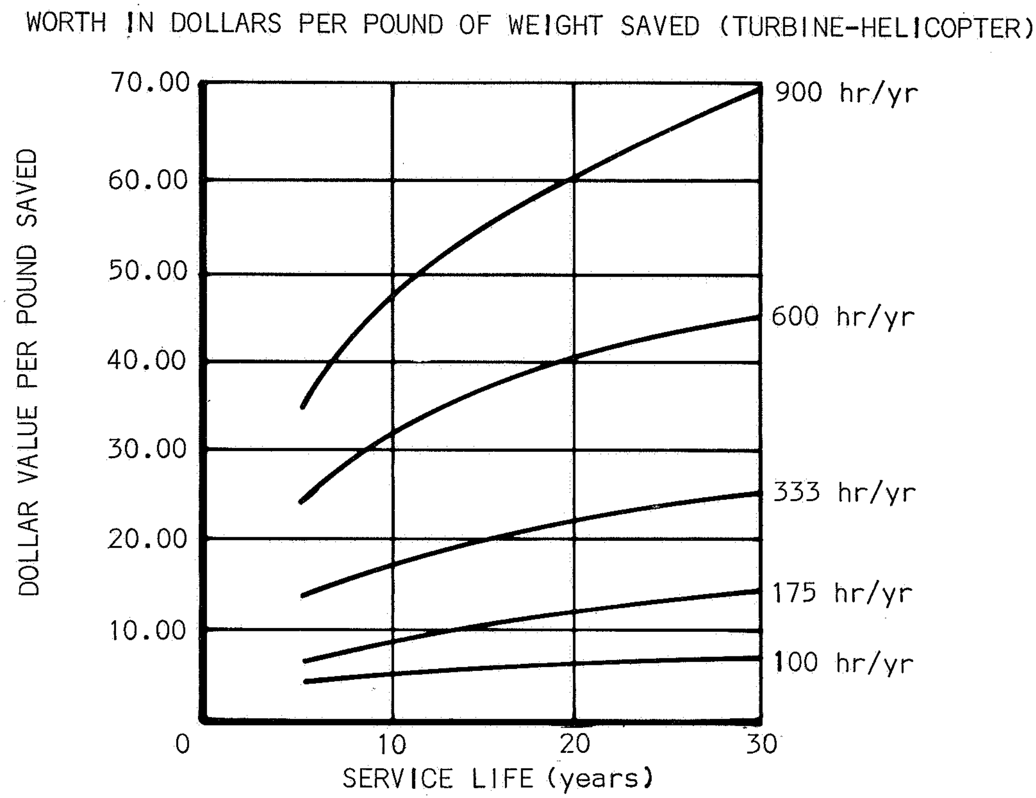


Figure 79

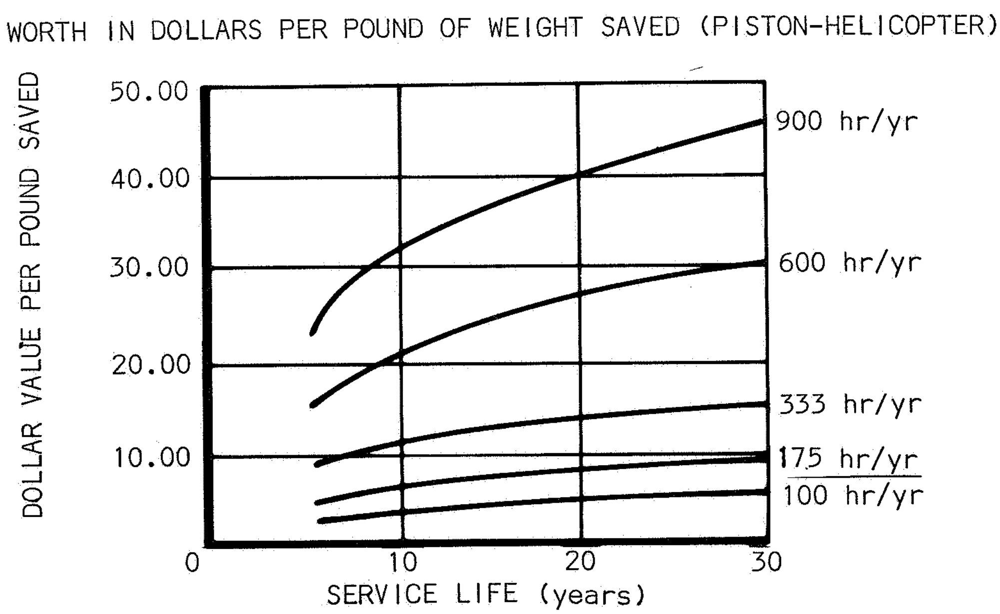


Figure 80

APPENDIX C

ESTIMATED COST OF CONVENTIONAL SHEETMETAL AIRPLANE AT 100,000th UNIT

Today's sheetmetal airplane, comparable to the NASA guideline Far Term airplane on an empty weight basis, would cost $12.50/lb* x 1609 lb, or $20,150.00.

Since 1956, one of the large light airplane manufacturers in the U.S. has produced a cumulative total of 25,000 airplanes, at an average present day price of $19,080. This is a line of airplanes which approximates the Far Term airplane and is fairly near the hypothetical $20,150.00 airplane.

From Table I (page 12), direct labor amounts to 10% of the consumer price. In this case it would be 10% x $20,150.00 or $2015.00.

The labor cost on the 100,000th unit is determined using a constant (linear) 80% learning curve. It is very conservative to use a constant 80% since, according to the U.S. Airforce Project Rand Report R-291**, there is apparently a minimum below which the labor cannot be reduced. This leveling off of the labor cost apparently occurs not long after the 300th unit. The following values are points on a constant 80% curve.

Labor Cost	Quantity (cumulative)
$ 2015	25,000
1612	50,000
1290	100,000

The consumer price of the 100,000th conventionally produced airplane can then be compared to the "Far Term" airplane as follows:

Item	Sheetmetal	"Far Term" (plastic)#
Labor	$ 1290 ##	$ 734.62
Overhead @ 130%	1677	955.21
Material (structure)	906(765/17000x20150)	811.25
Material (other)	167	167.00
Molding Time Charge		258.10
Engine and Propeller, L.G.,etc.	3842	3842.00
	7882	6768.18
Direct, Sales, G&A ###	2129	1211.92
Manufacturing Cost	10011	7980.10
Factory Profit @ 10%	1001	798.01
Dealer Cost	11012	8778.11
Dir. & Distr. Markup (25%)	2753	2194.53
Estimated Consumer Price	13765	$10972.64

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* See Figures 6 and 7.

** U.S. Air Force Project Rand Report (R-291), July 1, 1956, Cost-Quantity Relationships in the Airframe Industry.

\# See Appendix A. ## Airframe labor = 80% total labor = $1032.

\### From page 13, (direct + sales + G&A) = 2.95 x direct labor - overhead = (2.95)(1290) - 1677 = 2129.

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