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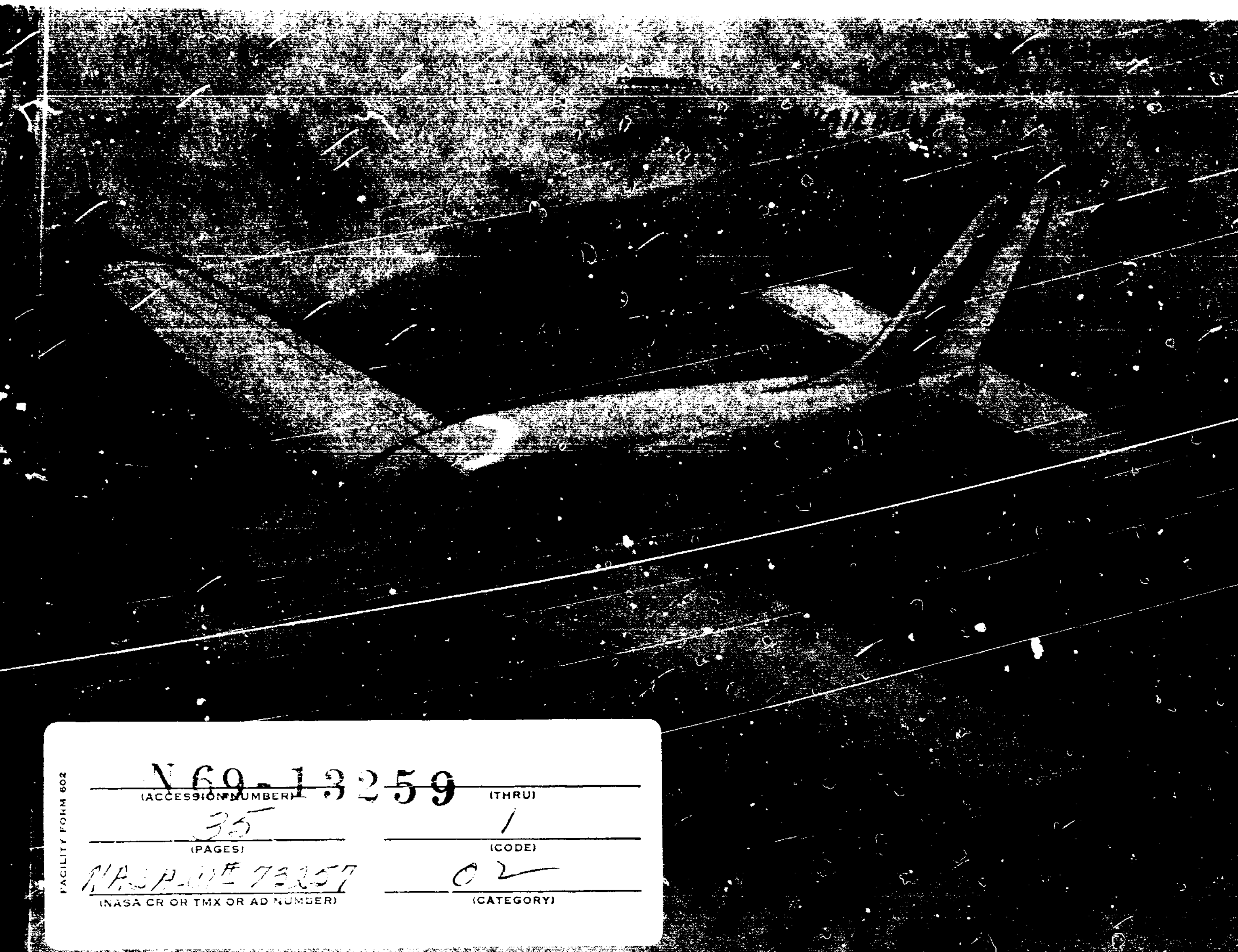

POTENTIAL STRUCTURAL MATERIALS AND DESIGN CONCEPTS FOR LIGHT AIRPLANES

SUMMARY REPORT
AUGUST 1968

SAN DIEGO AIRCRAFT ENGINEERING, INC.
San Diego, California 92101

NASA CONTRACTOR REPORT

NASA CR-73257

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By L. Pazmany, H. Prentice, C. Waterman and F. Tietge

October 1968

SUMMARY REPORT

Prepared under Contract No. NAS 2-4423 by
SAN DIEGO AIRCRAFT ENGINEERING, INC.
San Diego, California 92101

For THE MISSION ANALYSIS DIVISION
OFFICE OF ADVANCED RESEARCH AND TECHNOLOGY
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
AMES RESEARCH CENTER
MOFFETT FIELD, CALIFORNIA

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PREFACE

San Diego Aircraft Engineering, Inc. was responsible for executing the NASA study of potential structural materials and design concepts for light aircraft, as well as for the preparation of this report which describes the evaluation and application of several of these materials to the conceptual design of a single-engine, four-place airplane of the 1980's.

NASA contract NAS 2-4423 authorized these tasks which were performed for the Mission Analysis Division, Office of Advanced Research and Technology at the Ames Research Center of National Aeronautics and Space Administration, 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 & 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. i.e.:

Aircraft Owners and Pilots Association
Aluminum Company of America
American Aviation Corporation
Beech Aircraft Corporation
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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 Fiberglass Corporation
Piper Aircraft Corporation
Pixie Mold and Tool Corporation
Ryan Aeronautical Company
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Union Carbide Corporation
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CONTENTS

DESCRIPTION AND SCOPE OF AREA STUDIED

The expansion and competitive position of general aviation in the field of transportation depends upon improving the safety and utility of light aircraft while 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 the study were:

(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 effect 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.

Basic Assumptions and Methods of Approach

In pursuing these objectives the contractor was to consider two levels of technology and two types of light aircraft, fixed and rotary wing. The levels of technology were classified as "near term," 5 years hence, and "far term," 15 years hence. The conceptual designs were to meet the contract guidelines summarized in Table VI.

The study was performed in two phases. Phase I was concerned with researching, correlating, and evaluating available information on (a) operational characteristics; (b) material properties; (c) structural concepts and capabilities; (d) manufacturing and cost considerations; as they apply to light four-place airplanes and helicopters. The intent of Phase II was to select the more promising structural materials and concepts and apply them to the two conceptual designs for the two levels of technology. However, upon completion of Phase I, the results indicated (a) that the economic gains associated with improved light aircraft structural design would be more significant for "far term" aircraft; (b) that light fixed wing and light helicopters structures are similar; (c) the need for a more definitive analysis of the fabrication cost of the selected materials and concepts.

Thus, it was decided (with the agreement of NASA) to eliminate from consideration in Phase II the "near term" airplane and helicopter, and the "far term" helicopter. Phase II concentrated on establishing detailed structural design, cost, and fabrication analyses for those materials and concepts that showed the most promise of reducing labor hours and facilitating mass production as applied to the "far term" light airplane conceptual design.

A major aim of the study was to identify key problem areas where additional research would increase the potential of the more promising materials and concepts and lead to safer and more economical light aircraft.

SIGNIFICANT RESULTS OF THE STUDY

Evolution of the post WW II Light Aircraft Industry

Safety.- 60% - 70% of all accidents are improvable, i.e. they have causes associated with design, material selection, configuration, etc., which can be affected by the designer. A very small percentage (1%) of all accidents were/are caused by structural failure.

Utilization.- The contemporary single engine airplane is utilized at an average of 175 hours per year. A nation wide industry Survey indicated the following needs in future light airplane designs:

Simplicity in design to reduce maintenance costs
Improved safety features
Improved flight characteristics

Cost.- The prices of typical General Aviation aircrafts have shown the following trends in the last decade:

Price of low-price aircraft is fairly constant to declining.
Price of middle-price aircraft is fairly constant to rising.
Price of high-price aircraft is generally rising.

Performance.- In general, the aerodynamic efficiency of the typical General Aviation aircraft has increased very slightly during the last 20 years. Maximum speeds and landing speeds have increased gradually mostly as a result of increase in engine power and wing loading.

Design Considerations

Airplane Weight.- The structural weight of Typical General Aviation type aircraft is approximately 46% of the empty weight.

Analysis of three different wings with the same area, but different plan-form (I Rectangular, II Rectangular inboard panel with tapered outboard panels, and III Tapered.) resulted in the lightest wing being configuration III Tapered.

A study of weight trade-offs between wing area versus high lift devices to obtain a given stall speed resulted in the wing with double slotted flap as the lightest.

The structural weight of typical General Aviation piston and turbine powered helicopters are approximately 50% of the empty weight.

Crashworthiness.- During the course of this study, the subject of crash-worthiness and its relation to aircraft design was considered on many occasions and from every possible view point. The results of an outstanding research program made by the FAA were published in a report entitled "Aircraft Design - Induced Pilot Error".

Cost.- Operating cost: It cost approximately 15¢ per mile to operate a typical four-place airplane. (Based on 333 hrs/year utilization.). It cost approximately 55¢ per mile to operate a typical reciprocating engine powered light helicopter, and 80¢ per mile for a turbine engine helicopter (based on 300 hr/yr utilization). When comparing retractable gear versus fixed gear airplanes operating costs, break even points could be as low as 170 hr/yr. At the present, turbo prop-powered light airplanes are far more expensive to operate than piston engine-powered. Break even points are non existent.

Airframe cost: Reciprocating engine powered helicopters cost approximately $70.00 per pound of empty weight and the average cost of a typical light airplane is $12.00 per pound of empty weight.

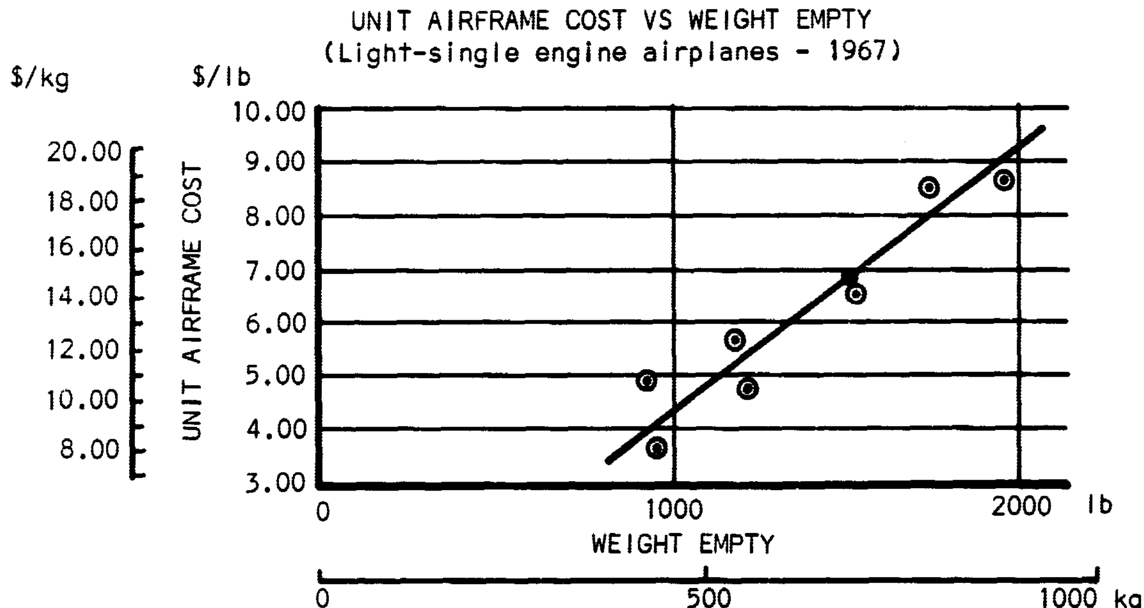

Figure 1

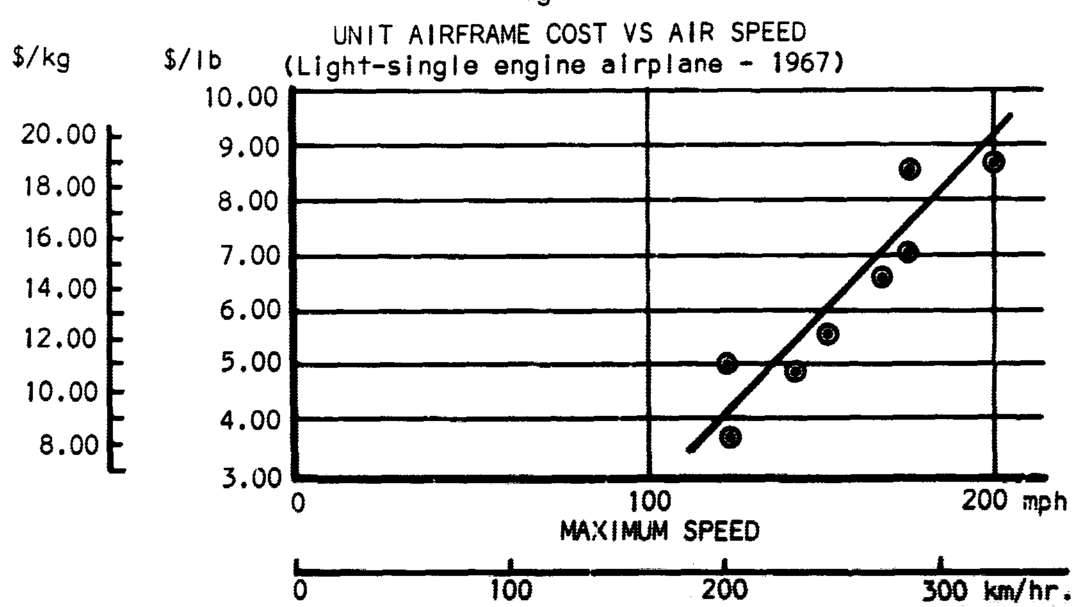

Figure 2

Airframe cost, representing several single engine light airplanes, is shown in Figures 1 and 2 as a function of empty weight and maximum speed. For the study \$7.00 per pound was used as a typical value for conventional aluminum construction. During the study the worth of a pound saved was estimated for light airplanes. The results are shown in Figure 3.

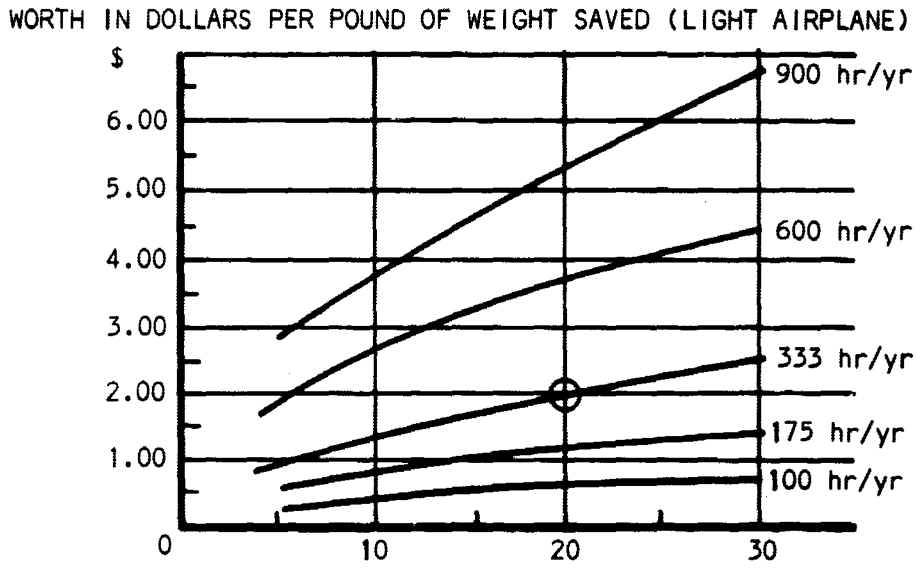

Figure 3

The typical breakdown of a four-place single engine airplane as shown in Figure 4, indicates that airframe labor and raw material is only 12.5% of the consumer price.

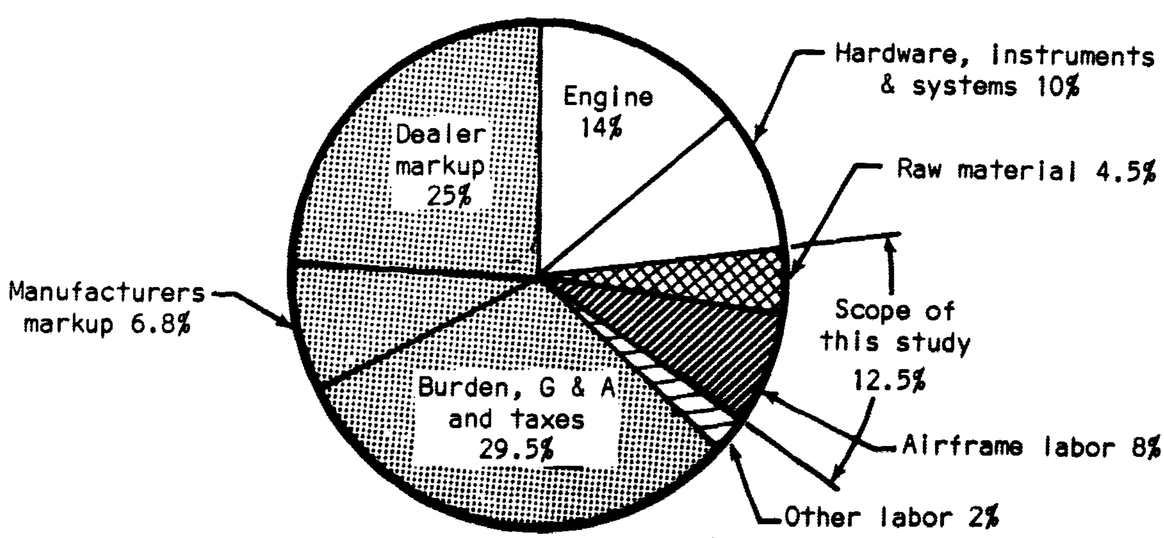

Figure 4

Effect of mass production on cost.- The cost of labor involved in manufacturing a light airplane affects other portions of the total price. The change in consumer price resulting from reductions in airframe fabrication labor is shown in Figure 5 and indicates that 50% reduction in labor will reduce the consumer price by approximately 25%. As labor approaches zero, the resulting consumer price approaches a limit of 42%. Obviously, the 100% savings in labor can only be approached through automation.

PRICE EFFECT OF LABOR SAVING

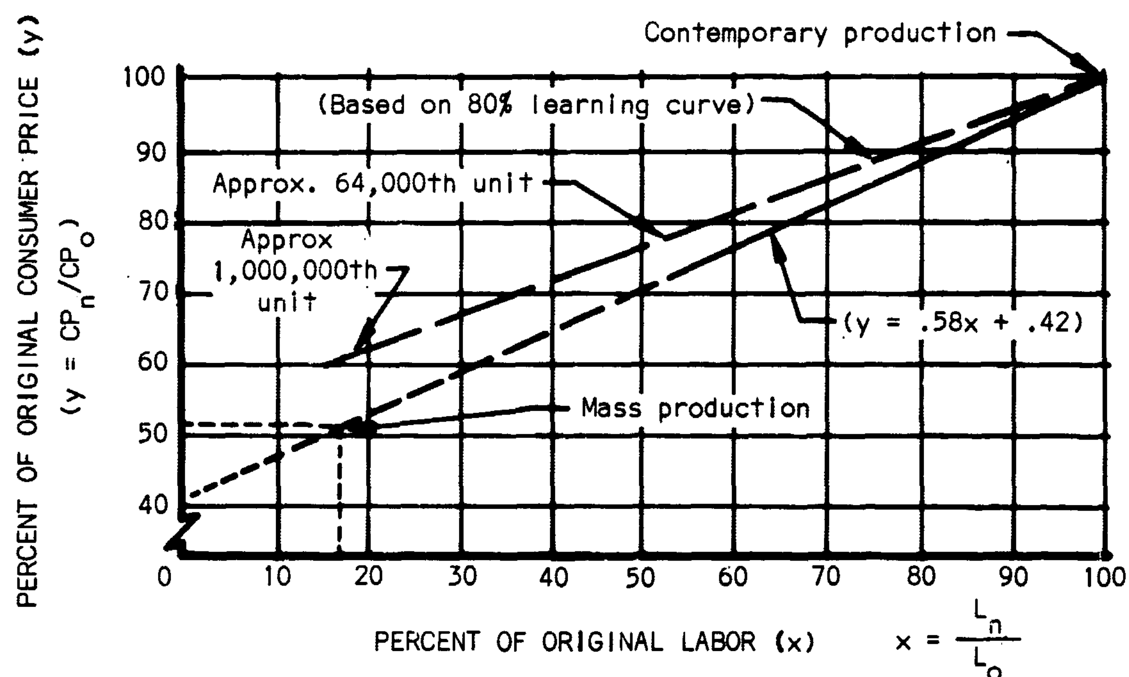

Figure 5

Performance.- Airplane: For the purpose of optimizing a configuration to meet the given set of guidelines, this study was limited to the parameters which would be affected by structural materials and at the same time have a major influence on the performance. In the final analysis, the basic parameters to be optimized were reduced to wing loading, W/S; and power loading, W/P.

A parametric study of factors affecting maximum speed indicates that sensitivity to parameters such as wing area, aspect ratio, wing thickness ratio, extent of laminar boundary layer, gross weight, and fuselage frontal area are rather small when compared to the effect of retractable vs fixed gear or a 10% change in power available.

It should be noted that although the effect of delaying the boundary layer transition does not justify a large expense to achieve abnormal surface smothness on conventional wings, it may be an important consideration for new materials such as plastics where an extremely smooth surface may be achieved at no extra manufacturing cost.

Potential Structural Materials.- Initial selection: 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 over 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.

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 was used.

Comparative structural efficiencies are presented in Tables I and II for those materials selected as promising candidates.

Evaluation of promising candidate materials. -The promising candidates were compared on the basis of types of members and concepts.

Composites: Anisotropic composites require some consideration 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

TABLE I

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. $e/10^5$	COMPARATIVE STRUCTURAL EFFICIENCIES $\frac{F_{tu}}{w}$	$\frac{F_{tu}}{w\ \$/LB}$	$\frac{\sqrt{E_c}}{w}$	$\frac{\sqrt{E_c}}{w\ \$/LB}$	$\frac{\sqrt[3]{E_c}}{w}$	$\frac{\sqrt[3]{E_c}}{w\ \$/LB}$
	(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	-	-
4130(Normalized)	N	95	75	75	55	29	12	.283	FAIR	0.92 (1)	GOOD	.63	336	365	19	21	-	-
6061-T6	N	42	35	34	27	10.1	12	.098	EXCEL	0.70 (1)	GOOD	1.30	428	617	32	46	-	-
BAR (t=1.00 in)																		
4130 (180HT)	N	180	163	179	109	29	12	.283	FAIR	0.13	GOOD	.63	635	4900	19	146	-	-
4340 (260HT)	N	260	217	242	149	29	10	.283	FAIR	0.16	FAIR	.63	919	5750	19	119	-	-
25 Ni (Maraging)	N	319	284	-	-	24	8	.296	GOOD	2.25	FAIR	.59 (5)	1078	480	17	8	-	-
FORGING																		
6181-T6	N	44	37	39	28	10.3	10	.098	EXCEL	-	-	1.28	450	-	33	-	-	-
2014-T6	N	65	55	55	39	10.7	7	.101	POOR	-	-	1.25	643	-	32	-	-	-
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
2024-T3 CLAD	N	60	45	37	38	10.2	15	.100	GOOD	0.65	GOOD	1.29	600	910	32	48	22	33
5086-H32	N	40	28	26	24	10.4	6	.096	GOOD	0.53	EXCEL	1.32	417	787	34	64	23	43
5456-H343 (4)	N	53	41	39	31	10.4	6	.096	GOOD	0.60	EXCEL	1.33	552	920	34	57	23	38
6061-T6	N	42	36	35	27	10.1	10	.098	EXCEL	0.54	GOOD (2)	1.30	428	794	32	60	22	41
X7005-T6	N	47	38	39	-	10.5	-	.101	GOOD	0.65	GOOD	1.32	465	716	32	49	22	33
7075-T6	N	76	66	67	46	10.5	7	.101	POOR	0.71	GOOD	1.29	752	1060	32	45	22	31
7178-T6	N	83	73	73	50	10.5	7	.102	POOR	0.71	GOOD	1.30	814	1145	32	45	21	30
AZ 31B-H24	N	39	29	24	18	6.5	6	.064	POOR	1.10	GOOD	1.40	610	555	40	36	29	27
EXTRUSION(t≤.250)																		
2014-T6	N	60	53	55	35	10.7	7	.101	POOR	0.97	GOOD	1.25	590	608	32	33	-	-
2024-T4	N	60	44	39	32	10.7	12	.100	POOR	1.12	GOOD	1.29	600	535	33	29	-	-
6061-T6	N	38	35	34	24	10.1	10	.098	EXCEL	0.44	GOOD	1.30	388	1710	32	73	-	-
7075-T6	N	81	73	74	45	10.5	7	.101	POOR	1.39	GOOD (2)	1.29	802	577	32	23	-	-
7075-T73	N	66	58	58	-	10.6	-	.101	GOOD	1.42	GOOD	1.29	655	462	32	23	-	-
7178-T6	N	88	79	79	47	10.5	5	.102	POOR	1.49	GOOD	1.30	863	579	32	21	-	-
Mg Yttrium-T5	P	55	50	50	30	6.5	4	.067	POOR	(6.00) (3)	GOOD	1.40 (5)	820	(137)	38	(6)	-	-
CASTING																		
A356-T61	N	38	28	28	27	10.5	5	.097	GOOD	-	-	1.19	392	-	33	-	-	-
356-T6	N	25	16.5	16.5	25	10.3	3	.097	GOOD	-	-	1.19	258	-	33	-	-	-
359-T61	N	45	34	34	31	10.7	4	.097	GOOD	-	-	1.16	463	-	34	-	-	-
ZK 61A-T6	N	34	23	-	-	6.5	2	.065	FAIR	-	-	1.40 (5)	523	-	39	-	-	-
ZE 63A-T6	N	38	24	-	-	6.5	4	.065	FAIR	-	-	1.40 (5)	585	-	39	-	-	-
AZ 91C-T6	N	27	14	14	-	6.5	2	.065	FAIR	-	-	1.40	416	-	39	-	-	-

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

TABLE II

PROMISING CANDIDATE MATERIALS - NON-METALLIC

MATERIAL	APPLICATION	F_{tu}	F_{ty}	F_{cu}	F_{su}	E_c	e	w	WEATHERABILITY	MATERIAL COST ②	THERMAL CO-EFF. $\alpha/10^5$	$\frac{F_{tu}}{w}$	$\frac{F_{tu}}{w\ \$/LB}$	$\frac{\sqrt{E_c}}{w}$	$\frac{\sqrt{E_c}}{w\ \$/LB}$	$\frac{\sqrt[3]{E_c}}{w}$	$\frac{\sqrt[3]{E_c}}{w\ \$/LB}$
												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
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)
1" S-Glass/Epoxy	FT	45	-	62	8	7.8	-	.060	EXCEL	4.00 (2.00)	-	750	(375)	46	(23)	33	(16)
E-Glass/Polyester	NT	20	-	26	-	1.99	-	.070	EXCEL	0.63	1.20	286	454	20	32	15	29
CLOTH REINFORCED																	
DAP Prepreg	NT-FT	49 ①	-	-	-	2.6 ①	-	.070 ①	EXCEL	3.15 (1.58)	-	700	(446)	23	(14)	20	(12)
181 Cloth/E-Glass	NT-FT	45	-	45	-	3.3	-	.070	EXCEL	(1.00)	-	643	(643)	26	(26)	21	(21)
181 Cloth/S-Glass	NT-FT	94	-	65	-	4.2	-	.070	EXCEL	(2.00)	-	1340	(670)	29	(14)	23	(12)
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)
S-Glass	FT	210	-	120	13.6	7.6	-	.073	EXCEL	(2.00)	-	2880	(1440)	38	(19)	27	(13)
±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)
S-Glass	FT	17.7	-	37.3	50.0	2.5	-	.073	EXCEL	(2.00)	-	349	(174)	22	(11)	19	(10)
±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)
S-Glass	FT	81.8	-	64.9	39.5	4.2	-	.073	EXCEL	(2.00)	-	1120	(560)	28	(14)	22	(11)
±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)
S-Glass	FT	113.8	-	78.7	34.0	5.1	-	.073	EXCEL	(2.00)	-	1560	(780)	31	(15)	24	(12)
±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)
S-Glass	FT	133.1	-	86.9	30.5	5.6	-	.073	EXCEL	(2.00)	-	1825	(912)	32	(76)	24	(12)
WOOD				F_{cy}													
Sitka Spruce	NT	9.4	5.3	3.5	1.0	1.4	-	.015	POOR	0.67	-	626	935	79	118	-	-
Mahogany/Poplar Plywd	NT	6.7	-	2.6	1.9	.9	-	.020	POOR	2.05	-	335	167	48	23	48	23
Spruce - Staypak	NT	35.8	25.9	4.3	1.3	4.7	.75	.047	FAIR	④	-	760	④	46	④	-	-

NOTES: ① ESTIMATED ② ( ) = 1982 ESTIMATE ③ NT - NEAR TERM, FT - FAR TERM ④ EXPERIMENTAL, NO PRICE AVAILABLE the angle. For this reason, composite systems are normally found in various combinations of fiber-oriented layers.

Strengthwise, fiber-to-resin proportion is another important relationship. 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. For fabric laminates, 70 percent by volume is considered normal.

Tension members: The efficiency of tension members were compared using the following equation:

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

S-Glass/Epoxy, Graphite/Epoxy, followed by 4340 (260 HT) Steel, are the most efficient candidates.

Simple columns: Minimum weight round tube columns were plotted versus load intensity and column length. In general Graphite composites and Mg - Yittrium - T5 are the most efficient materials while 1025 steel and Sitka spruce are the least efficient.

Compression structures: 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 is a result of equating general and local instability formulas:

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

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:

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

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 plane construction were 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 III.

TABLE III

MINIMUM AREA EQUATIONS FOR OPTIMIZED WIDE COLUMNS AND COMPRESSION PANELS

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	$\frac{N_x}{LE} = 0.823\ (\bar{t}/L)^3$	$\frac{N_x}{bE} = 3.62\ (\bar{t}/b)^3$

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 were compared on a thickness basis. The curves were obtained through a substitution and division process of the shear buckling equation for flat plates.

Shear buckling: $T_{cr} = \frac{K_s E_c t^2}{b^2}$

$T_{cr} = N_{xy} / t$,

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

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

Therefore:

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

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

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

Minimum weights versus structural indexes for flat plate shear panel materials were derived by multiplying shear buckling equations, as modified for minimum thickness form, by material density, w:

$$wt/b \sqrt[3]{K_s} = w\,(N_{xy} / bE)^{1/3} \qquad \text{But: } W = wabt\ , \quad w = W/abt$$

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

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

Compression flanges: In reviewing condidate 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 which assumes b/t,flange width to thickness ratio, to remain constant.

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 was calculated versus service life.

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

A typical light aircraft was used as a baseline against which weights and

TABLE IV

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 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 (*)	(1.00)	1350	.41	15.95	16.95	8.85	9.85	Yes
S-Glass (*)	(2.00)	955	.63	9.25	11.25	8.85	10.85	Yes

(*) ± 45°, 0°, 0° layers

(   ) indicates 1982 estimate

TABLE V

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	$\frac{K_n W_n}{K_b W_b}$	BREAKEVEN C_{fn} FABR.	BREAKEVEN C_{in} INSTL.	ACTUAL C_{fn} FABR.	ACTUAL C_{in} INSTL.	FEASIBILITY	$\Delta\$_{PP}$	$\Delta\$_{OC}$	$\$_{Savings}$
	(2)				K_1K_2		(3)	(3)	(5)		BRKEVN ≥ ACT.	(6)	(7)	(4)
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, $W_b/b_L 2 = (10^{-4}) = 5.0$														
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, $W_b/b_L 2 (10^{-4}) = 9.8$														
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 I and II with some modification where core materials are concerned.

(3) C_w = \$2.00/Lb.

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

(5) 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

costs were 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.

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 IV and V. In the case of wide columns, non-optimum factors due to practical stringer spacing and joint reinforcement are accounted for in calculating breakeven costs.

Material/Concept Feasability.- The feasibility of the various material/concepts is evaluated by comparing 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 IV and V 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 year (6667 hr.) period, and the net overall savings realized.

Fatigue Evaluation.- 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.

Fastening Devices and Methods.- Materials may be joined by either mechanical means such as riveting, bolting, welding, brazing or adhesive bonding. All of these methods may be used to some degree in aircraft construction.

Each method was evaluated and presented in the following manner:

(1) A brief description.

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

(3) Typical allowable strengths were given where applicable

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

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

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

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. 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.

Study Guidelines.- The Mission Analysis Division of NASA, established the guidelines for the design of a typical General Aviation type airplane to be used on the Application Phase of this study. The airplane is a single-engine four-place configuration and is referred herein as the "Far Term Airplane". The guidelines are listed in Table VI.

TABLE VI
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.
Propulsion		V_{stall}	48 knots @ S.L.
		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.

Design Justifications.- Figure 6 illustrates the airplane which satisfies the contract guidelines. Table VII lists the dimensions and general data.

Certain major parameters 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.

TABLE VII

FAR TERM AIRPLANE GENERAL DATA

Gross weight	(W)	2850 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°
Mean aerodyn.chord	(MAC)	5.173 ft.	Airfoil		NACA 0009
Speep @ c/4	(Λ)	0°			
Dihedral	(Γ)	5°	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°
			Airfoil		NACA 0012

The material/concepts selected for the various airplane components are based primarily on the results of phase I of the Study, and were summarized in Tables IV and V.

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.

Figure 7 illustrates the wing construction. 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 slices and skins are made of compression molded high modulus graphite filament/epoxy; the ailerons are made of injection molded E-glass/nylon; the tip fairings and the flap hinge fairings 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.

THREE-VIEW OF FAR TERM LIGHT AIRPLANE

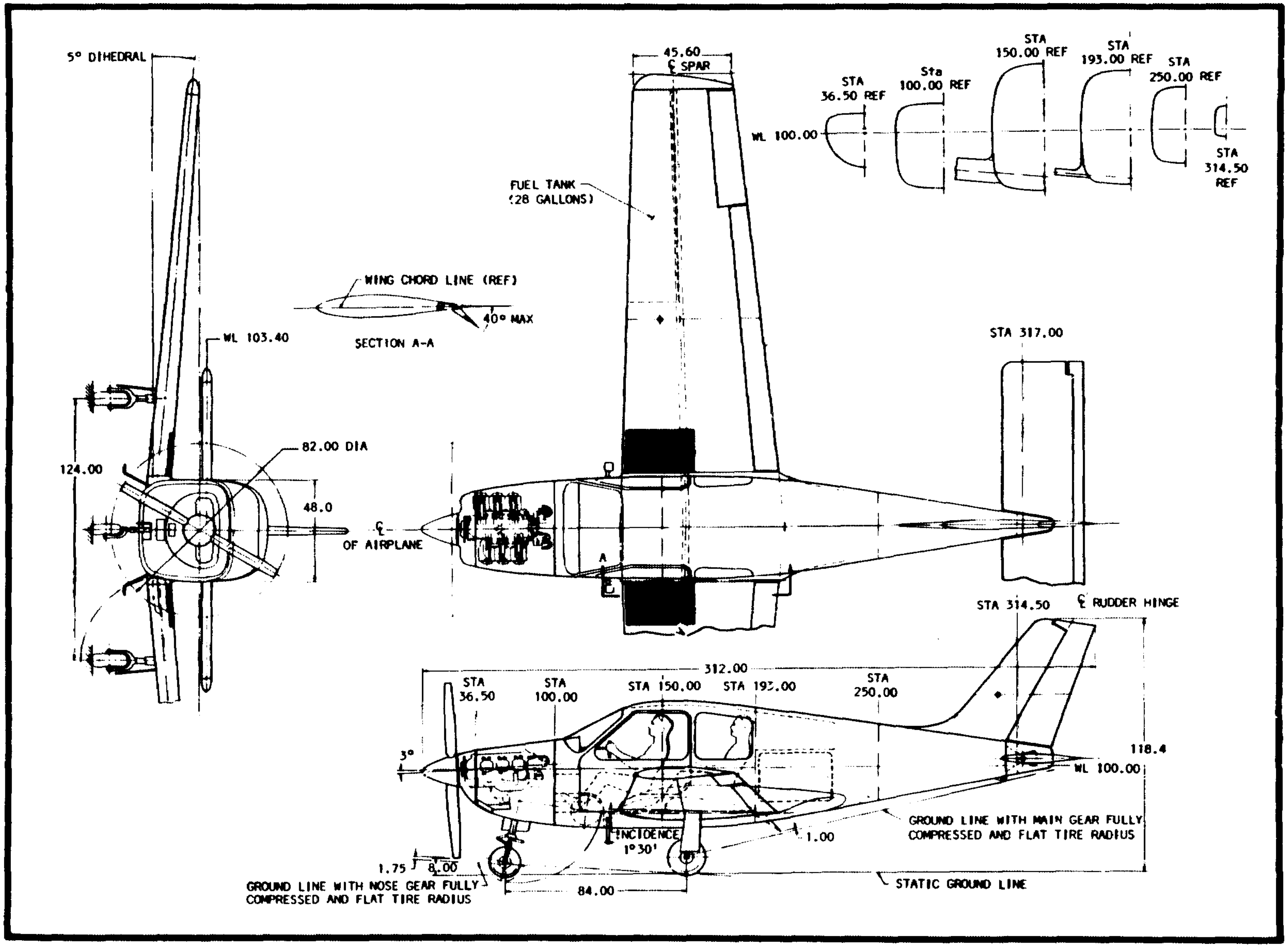

Figure 6

WING PLANFORM, FAR TERM LIGHT AIRPLANE

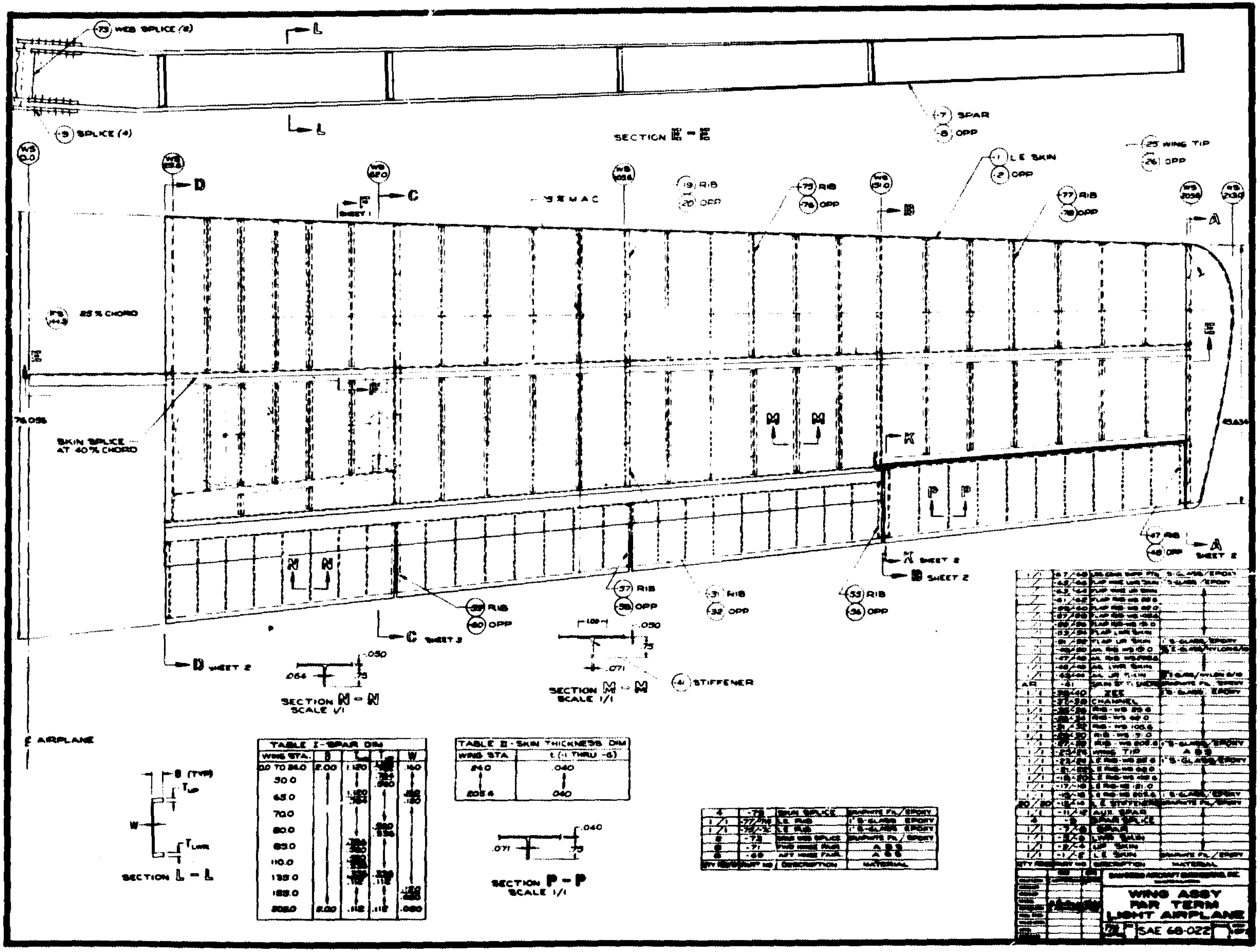

Figure 7

Two alternate wing construction concepts (designated II and III) were considered as possible weight and/or cost savers. 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 VIII. Configuration III is the same as II, except graphite is used in place of the S-glass. This concept (i.e.,III) leads 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).

TABLE VIII - WING WEIGHTS (POUNDS)

ITEM	FAR TERM LIGHT AIRPLANE			CONTEMPORARY AIRPLANE
	CONFIG. I Graphite Constr.	CONFIG. II S-Glass Constr. Foam Core Spar	CONFIG. III Graphite Constr. Foam Core Spar	ALUMINUM WING
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	26.9
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

Fuselage: 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. See Figure 8.

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.

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.

Figure 10 illustrates the vertical stabilizer design-based on a compression molded reinforced thermosetting plastic (1/2" E-glass/polyester available in the industry in .025 thick prepreg sheets). Minimum number of parts are characteristics of these concepts. The four-piece stabilizer and the six-piece rudder are entirely bonded. The rudder design was based on the same material/ concept.

FUSELAGE, FAR TERM LIGHT AIRPLANE

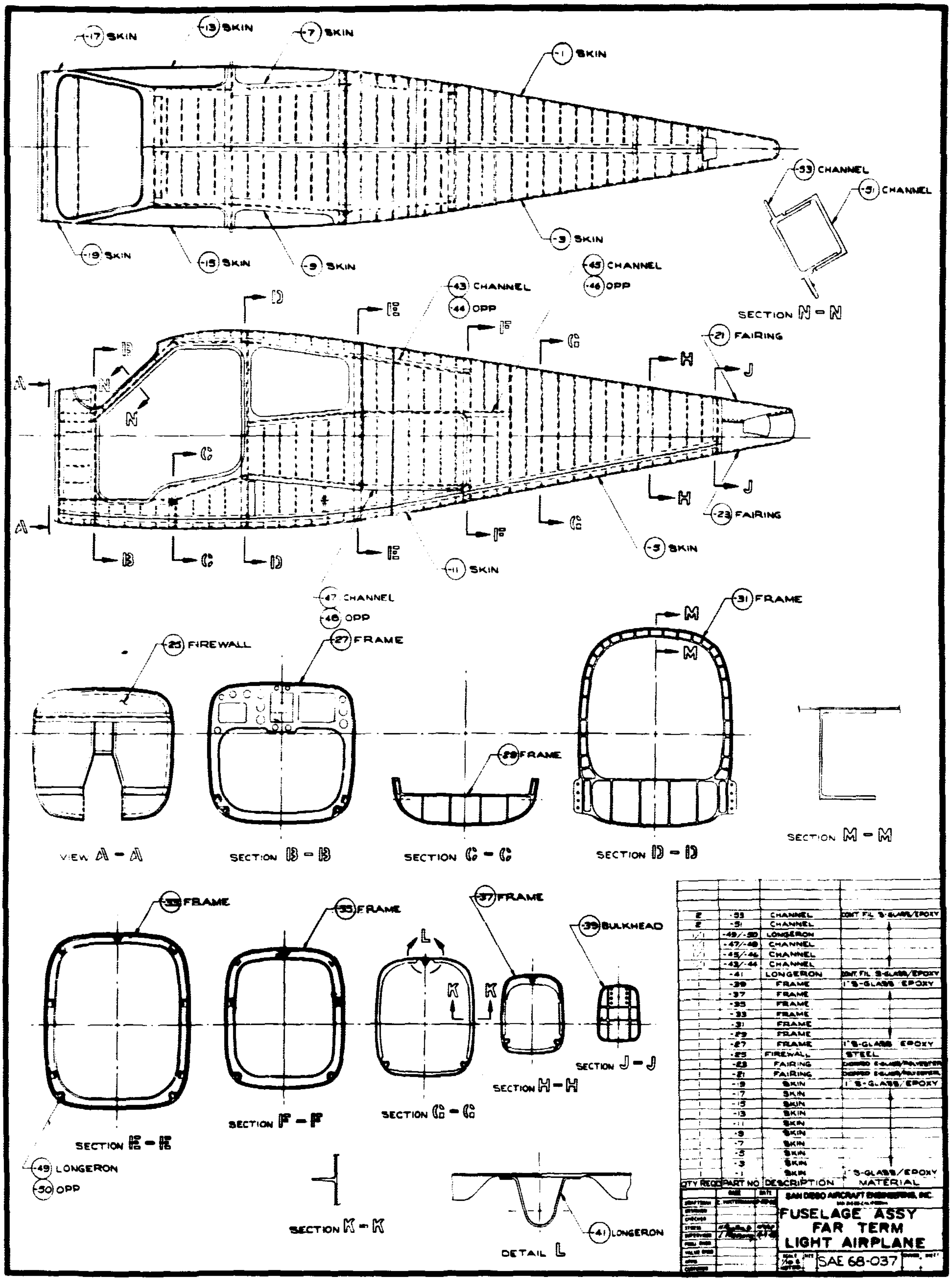

Figure 8

Figure 11 illustrates the horizontal tail design, which is also an all bonded construction of glass-reinforced plastic components, pressure molded from 1" S-glass/epoxy prepreg composite.

Table IX tabulates weights and unit weights for the empennage components which were comparable or lighter than contemporary sheet metal type construction.

TABLE IX - FAR TERM LIGHT AIRPLANE EMPENNAGE WEIGHTS

		COMPONENT	VERT. FIN		RUDDER		STAB	
		Area	(9.18 ft^2)		(6.66 ft^2)		(40.00 ft^2)	
Material	W	Weight	Total	Unit	Total	Unit	Total	Unit
	(Lb/in^3)	Fabrication	Lb.	Lb/ft^2	Lb.	Lb/ft^2	Lb.	Lb/ft^2
Nylon 6/10	.051	Inject.mold	14.44	1.58	9.35	1.40	NA	NA
E-glass/polyester	.070	Comp.mold	13.13	1.43	8.50	1.28	NA	NA
1"S-glass/epoxy	.062	Comp.mold	11.63	1.27	7.50	1.13	36.06	0.90
Aluminum sheet	.100	Riveted	-	1.47	-	1.10	-	1.07

Component Cost and Manufacturing Considerations.- The cost analyses were limited to the vertical stabilizer and the wing. The analyses demonstrated the magnitude of the potential savings associated with machine molded/high production rate construction concepts.

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. Of prime significance is the indication shown in Figure 9, that both injection molded and compression molded vertical stabilizers can be manufactured at a lower cost than conventional sheetmetal, even at current quantities.

VERTICAL STABILIZER UNIT COST VS PRODUCTION RATE

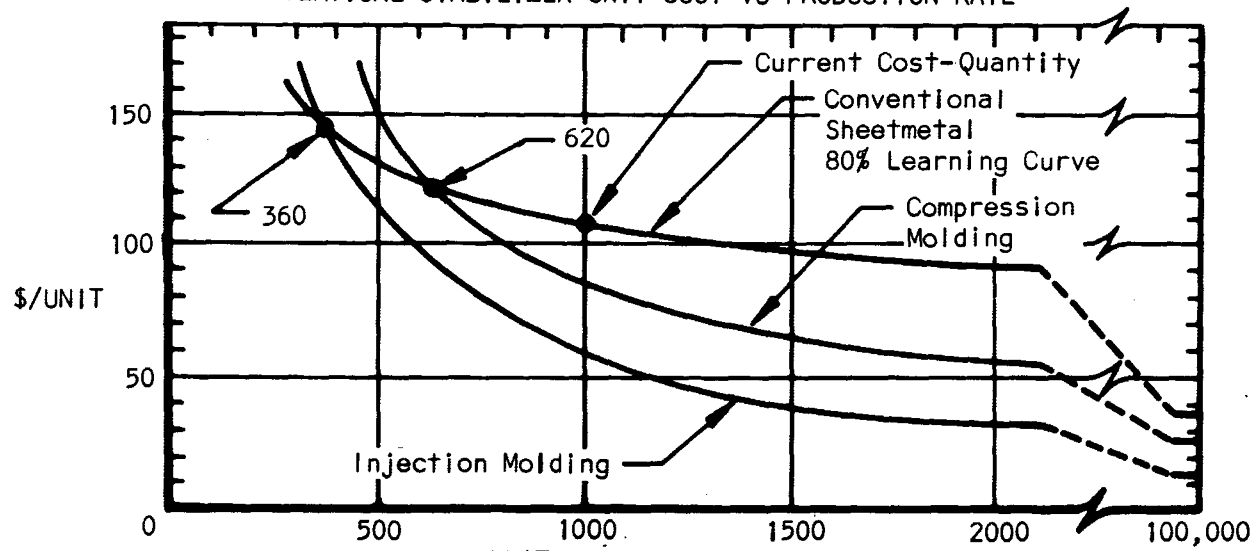

Figure 9

VERTICAL STABILIZER, FAR TERM LIGHT AIRPLANE

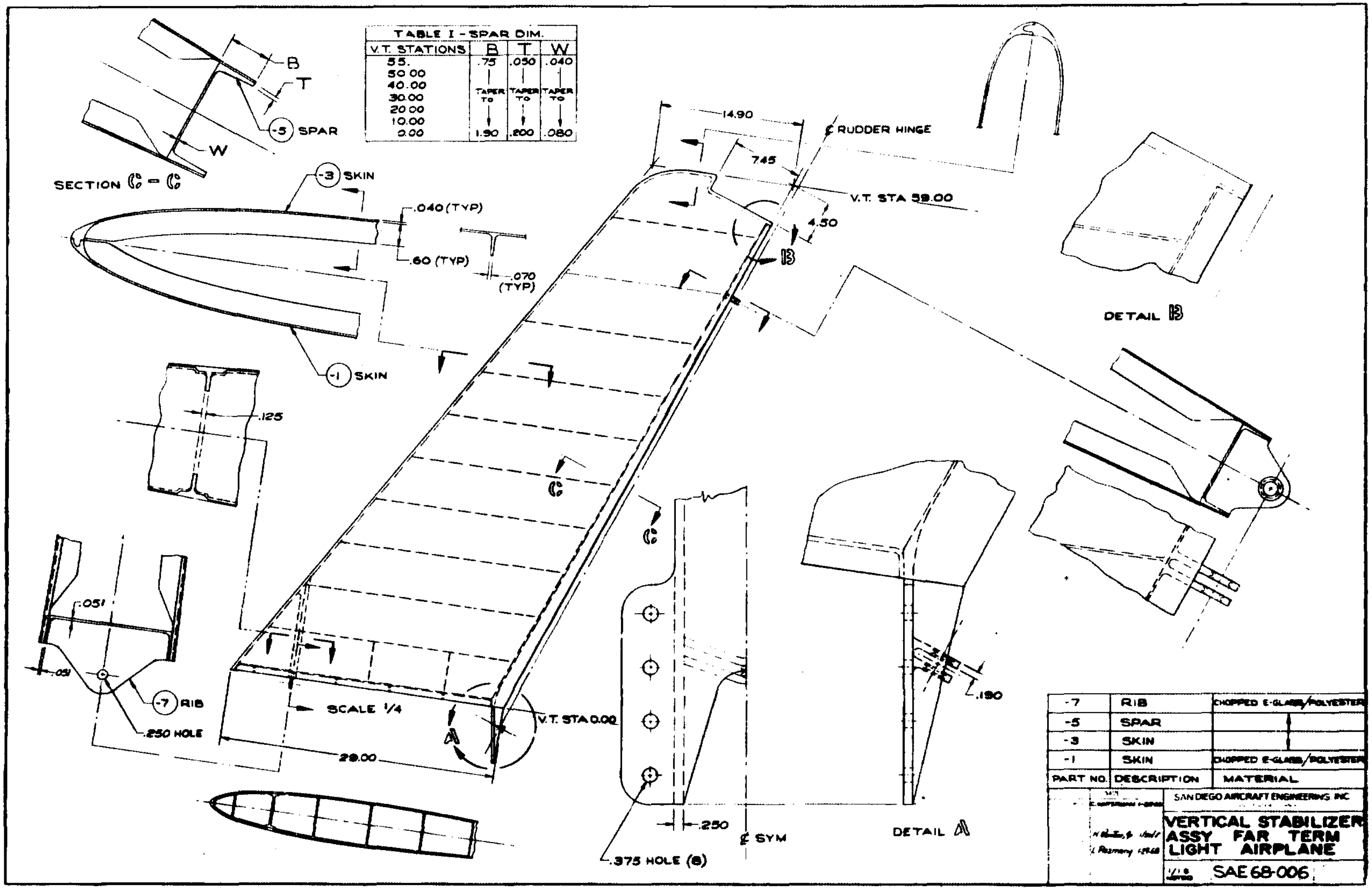

Figure 10

HORIZONTAL STABILIZER, FAR TERM LIGHT AIRPLANE

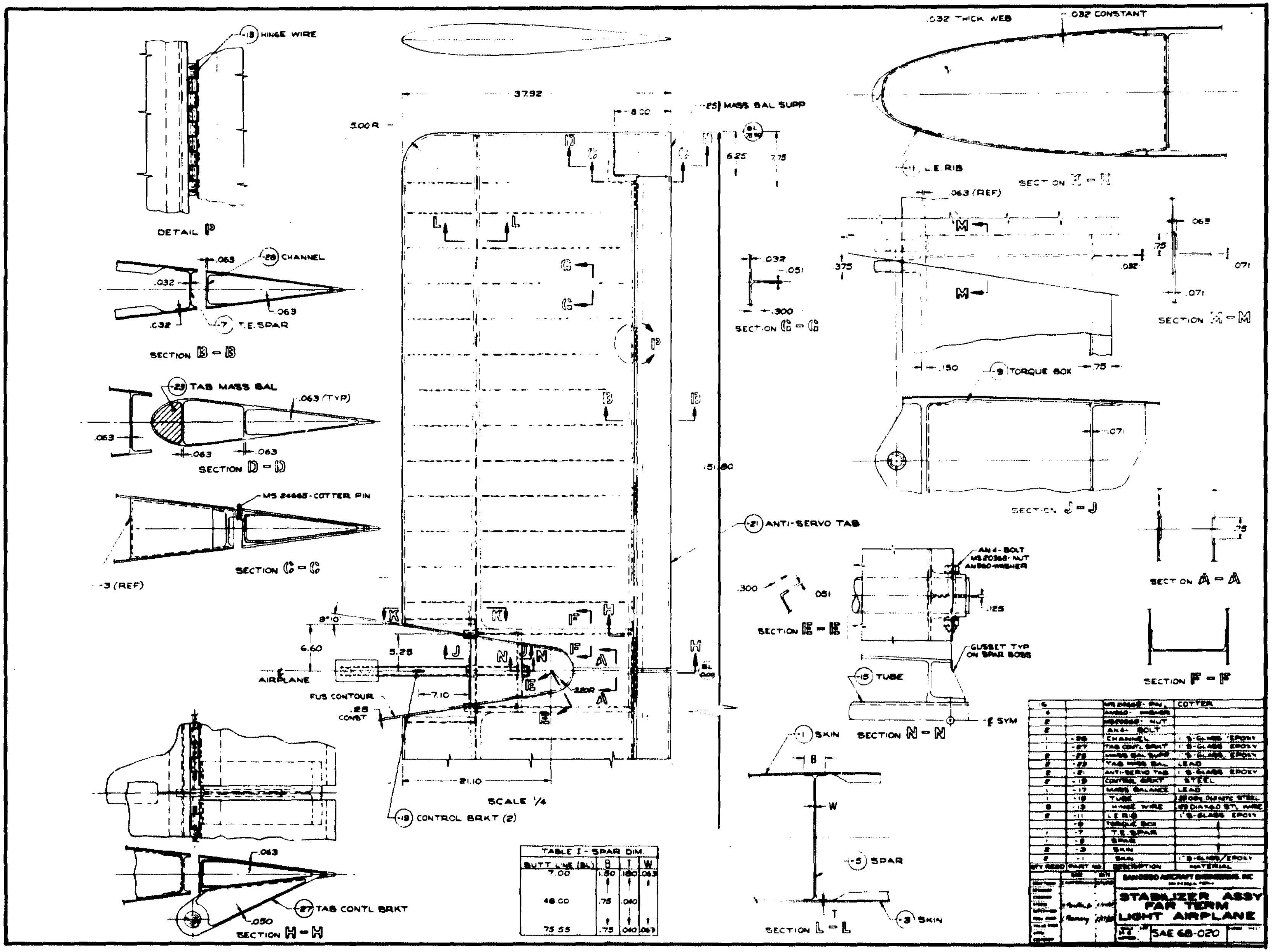

Figure 11

Wing: The first cost analysis, based on the tapered wing illustrated in Figure 7, assumed that each component would be machine molded individually in a press of appropriate capacity, considering both the 30-minute cure time for current epoxies and an estimated cure time of 15 minutes for future epoxies. Referring to Figure 12, 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.

Bars (3) and (4) represent the same wing as bars (1) and (2), respectively, except for the use of multi-cavity tooling. Examination of bar (4) made 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.

FAR TERM LIGHT AIRPLANE WING UNIT MANUFACTURING COSTS
(for 100,000 units/year production rate, except 9)

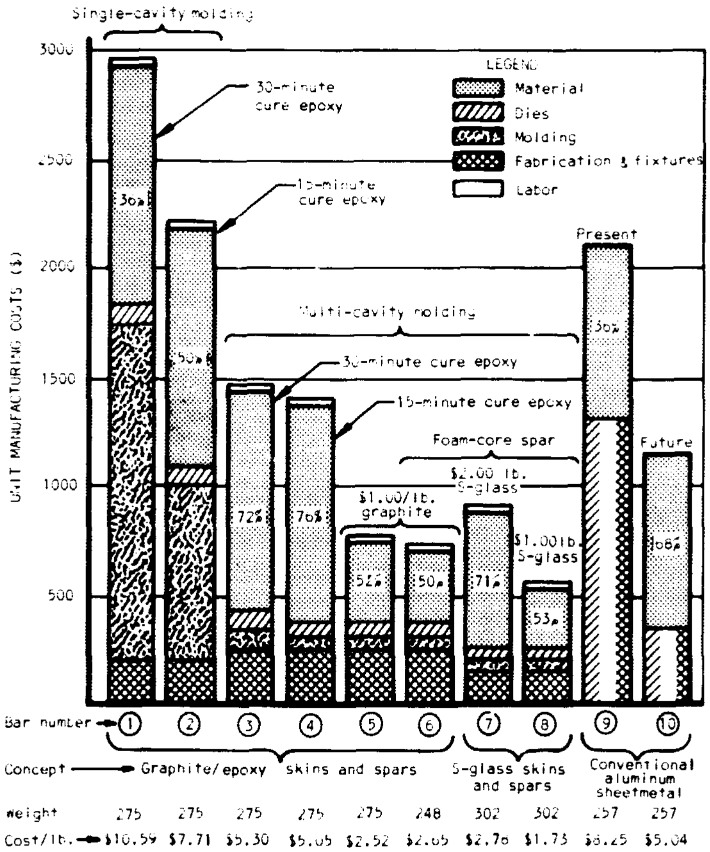

Figure 12

CONCLUSIONS

General

This study investigated every aspect of light airplane and helicopter design, manufacturing methods, and materials in use or of potential use as structural and non-structural components. Significant conclusions drawn from the investigation now follow.

Present helicopter construction is not different from typical light airplane construction but helicopter designers use more of the sophisticated techniques common in military aircraft and commercial transports, such as bonding and chemical milling, because weight savings are more important in a helicopter than in a light airplane.

Typical light airplane structure consists of relatively large sheet metal panels 0.025 to 0.032 inch thick, whereas helicopter fuselages have smaller panels 0.020 to 0.016 and even 0.012 inch thick, supported by many very lightweight formers and stiffeners. These lighter but more elaborate constructions, coupled with lower production rates, are conducive to higher airframe costs. The average helicopter cost is $30.00 per pound of empty weight, compared to $10.00 per pound of empty weight for light airplanes.

Because of the lower rate of helicopter production (586 helicopters versus 15,747 airplanes manufactured in 1966 - a ratio of 1 to 27), and assuming this ratio remains more or less constant, in fifteen years the light helicopter industry may be manufacturing 3,700 units per year, still below the present production rate of the light airplane industry. Consequently the mass production techniques visualized for producing 100,000 light airplanes may not be justified for light helicopter manufacturing.

There is no doubt, after comparing hundreds of different materials, and discussing the study with leading light airplane and helicopter manufacturers in the United States, that aluminum sheet metal airframe is here to stay. Aluminum is a very readily machinable, formable, and joinable material. It is one of the most economical; and manufacturing processes, techniques, and equipment are at hand. On that basis, it would be unwise to deviate much from this. A new material would mean new tooling, a learning period, etc.

Modern light airplanes and helicopters are made almost exclusively of 2024-T3 aluminum alloy. The study indicates some relatively new alloys, such as 6061-T6 and 5086-H32, can replace most of the skin material with resultant cost savings. Some manufacturers are aware of this. 6061-T6 is used in some models by leading aircraft manufacturers. Wooden construction is obsolete for mass produced airplanes.

Riveting is still the easiest, cheapest, and most inspectable way of joining two pieces of metal. Two women with an air gun, a bucking bar, and two hours of training can install perfectly acceptable rivets at a rate of 20 per minute; and, when the nature of the assembly permits it, automatic riveting machines can drill and squeeze rivets at the rate of 30 per minute with one operator.

Automatic spotwelding of aluminum sheet metal deserves a very careful look. The light aircraft industry is looking to structural bonding with great hopes. The idea is well proven: It works with military and commercial airplanes. Modern helicopter blades would not be feasible without metal bonding. Light aircraft manufacturers are beginning to use it.

The pre-priming of aluminum sheets at the mill might solve, economically, one of the biggest problems of bonding surface preparation. New fast-curing low-pressure adhesives are eagerly awaited by most light aircraft manufacturers. Automated, fully-reliable quality-control devices will make bonding more attractive. Human handling should be reduced as much as possible. At present, quality depends too much on the individual handling of each assembly. The advantage of the extensive use of bonding might be emphasized mainly on weight savings, (3 to 6 percent of structural weight). Fatigue life improvement due to bonding is a well known fact, but of little concern to the light aircraft manufacturer.

The use of fiberglass prepreg laminates and ultraviolet curing seems the most promising technique for lightly-loaded or non-structural parts. This concept has been proven in the mass production of drones. Just recently, several light airplanes made of glass reinforced laminates have reached the flight testing stage in the United States and Germany. They could be considered Near Term designs, and if the manufacturing cost are comparable to the sheet metal counterparts, they might lead the way to a revolution in light aircraft production.

Considering the raw material cost for a typical four-place airplane is only $ 765.00 (4.5 percent of consumer price), there is no doubt but that the only possibility for a radical improvement in price for a Far Term airplane will be in the reduction of airframe labor cost, rather than reduction of material costs.

In an opinion canvass of leading aircraft manufacturers, they were all in agreement that at the present, hand layup fiberglass construction is not cheaper than sheet metal construction. To this, can be added the fact that the basic raw material (fiberglass fabric (E-Glass) @$2.00 per pound at present) is three times more expensive than aluminum sheet metal. This makes the success of the all-fiberglass hand layup airplane very doubtful for the near term.

The Far Term airframe should be an injection or compression molded article, using a thermoplastic or perhaps a thermosetting material. The production of airplane parts by molding is not only feasible, but probable; the materials are existent; molding techniques and limitations are well known.

Nylon matrixes, reinforced with chopped glass could be used for injection molding components twice the size of a briefcase. The structural efficiency number for shear buckling is 23 compared with 22 for aluminum sheet. The price of reinforced nylons is $1.64 per pound today, but in fifteen years it may be as low as 65 cents a pound. The time required to mold a part is measured in seconds, at the most, one minute.

Glass-fiber-reinforced epoxy systems with a shear buckling index of 33, cost $4.00/lb at the present and it is forcast to be only $2.00 in fifteen years. Graphite-fiber-reinforced plastics, still in the development stage, are estimated by some of the industry to be commercially available in fifteen years for as low as $2.00 per pound.

Analysis of the effect of mass production revealed that the institution of automotive-type manufacturing methods could reduce the price of a typical and representative light airplane by approximately 48 percent. By using existing aircraft manufacturing methods (plus normal evolution) and the classic 80% (constant) learning experience, the price of this representative airplane could theoretically be reduced to $14,651 on the 100,000th unit.

The estimated consumer price for the "Far Term" airplane is compared to that of a conventional sheetmetal airplane in the table below. This table is based on a production quantity of 100,000 units.

Item	Sheetmetal	"Far Term" (plastic)
Labor	1290	$ 734.62
Overhead	1677	955.21
Material (structure)	906	811.25
Material (other)	167	167.00
Molding Time Charge		258.10
Engine, Propeller, L.G., etc.	3842	3842.00
	7882	6768.18
Direct, Sales, G & A	2773	1211.92
Manufacturing Cost	10655	7980.10
Factory Profit	1066	798.01
Dealer Cost	11721	8778.11
Dir. & Distr. Markup	2930	2194.53
Estimated Consumer Price	$ 14651	$ 10972.64

The cost per pound of empty weight for the reinforced plastic airplane would be 6.80 $/lb. and the cost of airframe 3.07 $/lb. These figures are nearly 50% of the values quoted on page 3 for present day production airplanes.

Recommendations for Future R&D Programs

As a result of this study, various areas are identified where additional research will enhance the possibility of a safer and more useful light aircraft. Some of the recommendations follow directly from the investigations performed during the study, while others are those suggested by various people within the industry. The recommendations are divided into two categories. Those indicated by the study and related to structural materials and concepts, and those beneficial to other design areas of future light aircraft. No attempt was made to rank the various recommendations.

Category I. - Structural Materials and Concepts.

(1) Fatigue characteristics of panels as related to panel size, rivet spacing, and material thickness.
(2) Fatigue characteristics of typical repairs on aircraft. For example oversize holes, patches or splices on spar caps.
(3) Materials for landing gear springs.
(4) Stress corrosion in regard to protection and corrective action.
(5) A structural adhesive which will cure at room temperature with high T peel strength (75 lbs/in), high shear strength (4500 psi), curing at 10 psi (vacuum bag) in 10 minutes, one phase, rolled on curtain coating, lack of sensitivity to surface contamination.
(6) An extruded helicopter blade, which combines heavy sections at the leading edge and very thin sections at the trailing edge.
(7) Data on fatigue of bonded structures.
(8) Test data on creep and fatigue of plain laminate and sandwich panels with representative fiber orientations. Test laminates to obtain F_{tu} vs ϕ for various fiber patterns. Also combined loading to confirm biaxial strength criteria.
(9) Mechanical properties of laminates as function of resin and void content.
(10) Develop aluminum sheet metal with prime coat ready for bonding without any further surface preparation except solvent cleaning. The coating should provide also corrosion resistance. The coating should be applied at the mill for low cost.
(11) Specifications for raw materials, resins reinforcements for composites. Standardize test methods, specimens.
(12) Tests for determining crack propagation characteristics of various fiber orientated composite laminates.
(13) Establish design criteria for plastic structural components: Maximum and minimum temperatures, humidity,hail stone,sand and dust errosion.
(14) Test data on non-continuous glass reinforced laminates (mechanical properties, and environment limitations or degradation).
(15) Test data on compression allowables of laminate plates and flange members for varying width/thickness ratios and also for different fiber orientations.
(16) Tests for determining attachment allowables in laminate composites varying fiber orientations, thickness and edge distance.
(17) Tests for determining effect of stress concentrations in composite laminates under static and sudden loading conditions. Vary thickness and fiber orientation. Also include bolted attachment configurations.

Category II.- Other Design areas.

Power Plant.-

(1) Cooling drag, optimum air inlets and exhaust designs for horizontal opposed power plants.
(2) Methods of reducing propeller noise.
(3) Design parameters of small diameter, multi-blade propellers in shrouded ducts.
(4) Development of ultra-low pitch blade settings for ground roll braking.
(5) Simple CO detectors and CO elimination.
(6) Improved fuel injection equipment (mass and flow sensing).
(7) Engine mounts, with lower frequency having less damping for better isolation in operating range.

Systems.-

(1) Improved braking methods.
(2) Improved flotation of tricycle gears on soft fields.
(3) Survey of landing loads, accelerations, sink speeds, etc. to determine if a more realistic design criteria is required.
(4) Oxygen systems for high altitude unpressurized aircrafts.
(5) De-icing of inlets, leading edges and control surfaces.
(6) Simple, inexpensive air conditioning system.
(7) A simple fuel system design, which would bleed fuel from wing tanks simultaneously to eliminate "fuel management", as required with present systems.
(8) More experimental data on crashworthiness of light airplanes and revised design requirements.
(9) More VGH data for fatigue evaluation of light airplanes. Perhaps FAR 23 should be subdivided according to type of operation (commercial survey, training etc.) or perhaps another subdivision should be made at 6000 lbs. gross weight.

Manufacturing.-

(1) Automated plexiglass forming.
(2) A core material for hollow laminated parts, which can be easily removed, (for instance water soluble).
(3) A better bagging material for laminate fabrication than the presently used PVA, (should be reusable).
(4) Establish processing techniques for Ultra Violet curing of plastics.
(5) An improved casting process to yield thinner walls and greater precission in the manufacturing of piston engine cylinders.
(6) Develop manufacturing techniques, tools, and establish design criteria for compression and injection molding of very large glass reinforced moldings and laminates.

Aerodynamics.-

(1) Control surfaces hinge moments experimental data.
(2) Additional basic data on laminar flow airfoils including effects of various types of moveable surfaces.
(3) Effects of different leading edge shapes on laminar flow airfoils.
(4) Improved effectiveness of vertical tails.
(5) T-tail characteristics.
(6) Nacelle shapes and locations for pusher engine installations. Also effects of wing and flaps on propeller.
(7) Flight path control with flap and power modulation.
(8) Simple methods of stability augmentation.
(9) Data regarding spoilers and vortex generators.
(10) Additional data on stall and section characteristics at low Reynolds numbers.
(11) Minimizing pitch changes with gear, flap and power changes by changing vertical placement of horizontal tails (full scale wind tunnel tests).
(12) Variable stability for production airplanes, heavy in cruise and light in low-speed flight.
(13) Means of getting usable C.G. ranges of 10 to 40% MAC. with reflexed airfoils, upward floating ailerons or flaps.
(14) Drag reduction of tricycle landing gear.
(15) Improvement in handling qualities at 1.1 Vs in landing approaches.
(16) Use of canard surfaces for supplemental longitudinal control.
(17) Practical methods of eliminating adverse yaw in low-speed airplanes.
(18) Summary of NASA-NACA data applicable to stability and control design for personal airplanes.
(19) Stabilator design for minimum pitch change with power and flaps.
(20) Improvement of spiral stability with upward floating ailerons.
(21) Condensed bibliography listing significant reports and summary reports from the beginning of NACA.
(22) Bibliography of STOL and high lift reports.
(23) Up-dating of many reports regarding airfoil data and structural data to take advantage of present state of the art.
(24) A method of automatic flight control from take-off to landing as applied to general aviation.
(25) Span load distribution for wing tips or various planforms and various section shapes.