

STRESS
ANALYSIS OF THE
WING STRUCTURE OF
A PRIMARY GLIDER
MICHIGAN STATE COLLEGE
THESIS FOR DEGREE OF B.S.

SVEN G. JOHNSON

1941

THESIS

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SUPPLEMENTARY
MATERIAL
IN BACK OF BOOK

STRESS ANALYSIS OF THE WING STRUCTURE OF A
PRIMARY GLIDER

A Thesis Submitted to

The Faculty of
MICHIGAN STATE COLLEGE

of
AGRICULTURE AND APPLIED SCIENCE

by

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Candidate for the Degree of
Bachelor of Science
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INTRODUCTION

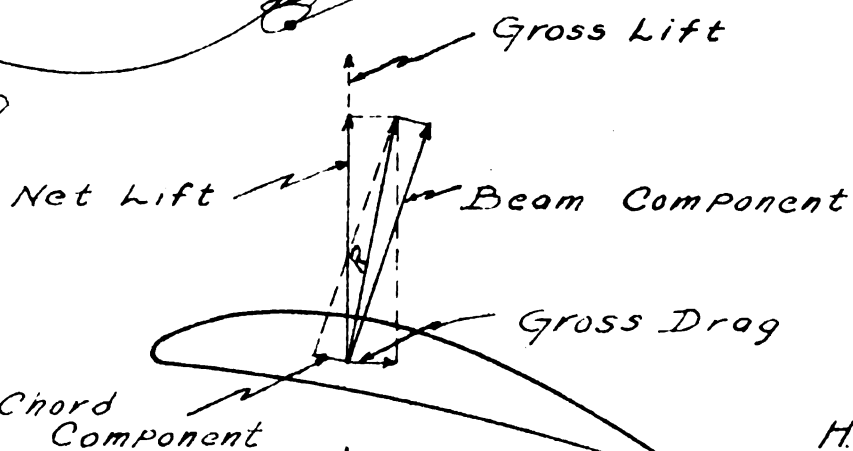
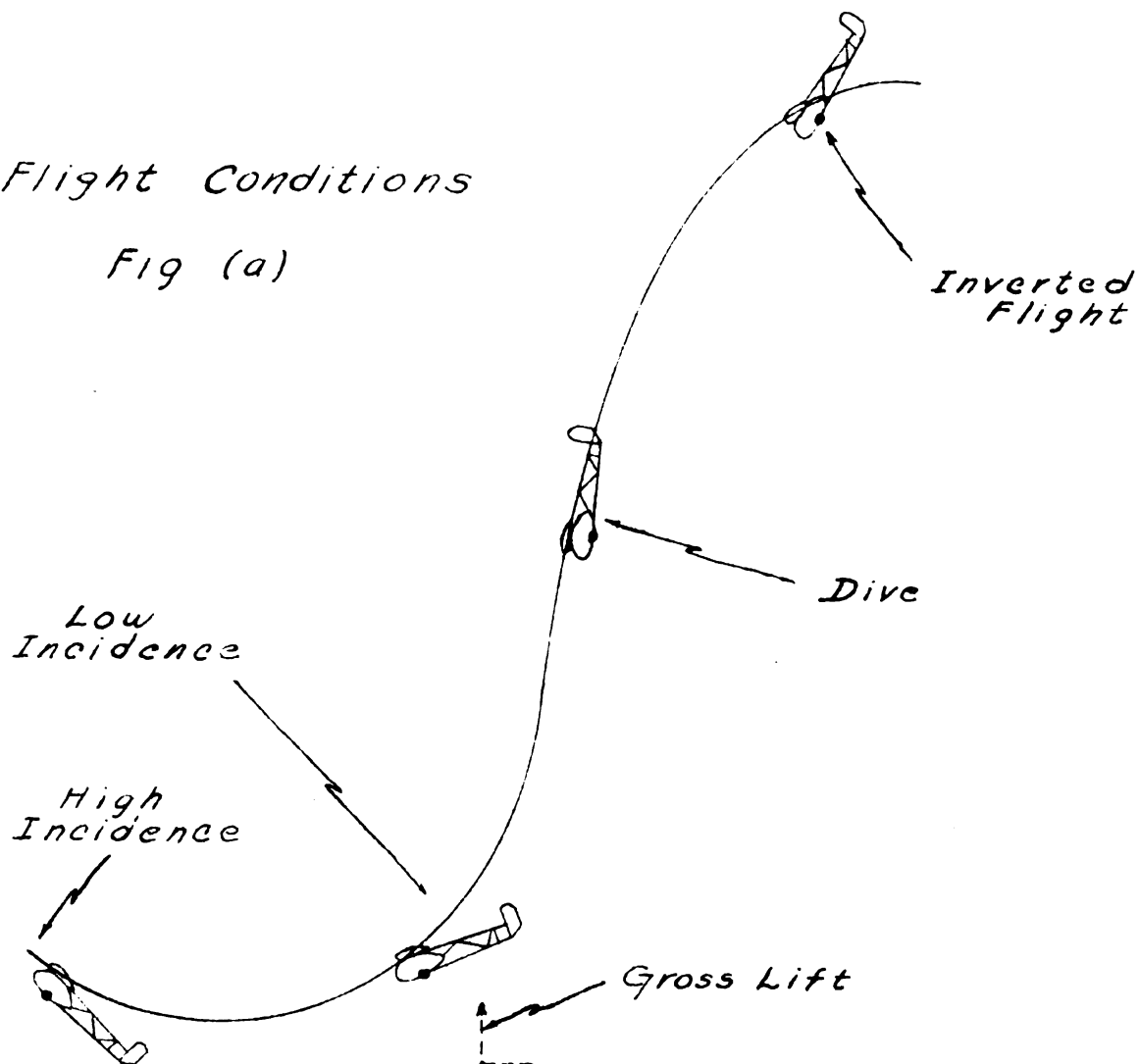
The glider under consideration in this thesis is one that was presented in the "Flying Manual" Vol. 2 - No. 6, 1941 publication. Having given some thought to building a primary glider, I became interested in finding out just how well the wing structure reacted under loading. Since this glider is intended to be of such nature that it could be built by the average laymen, I was curious to know how severe the stresses in the wing structure were as well as become somewhat acquainted with the principles of airplane stress analysis.

In stress analyzing the wings of a heavier than air machine there are four main flight condition that must be investigated. These are high incidence, low incidence, inverted flight, and nose dive condition. These conditions are encountered when a plane is put into a steep dive and then pulled out. This can best be illustrated by sketch (a) p.2.

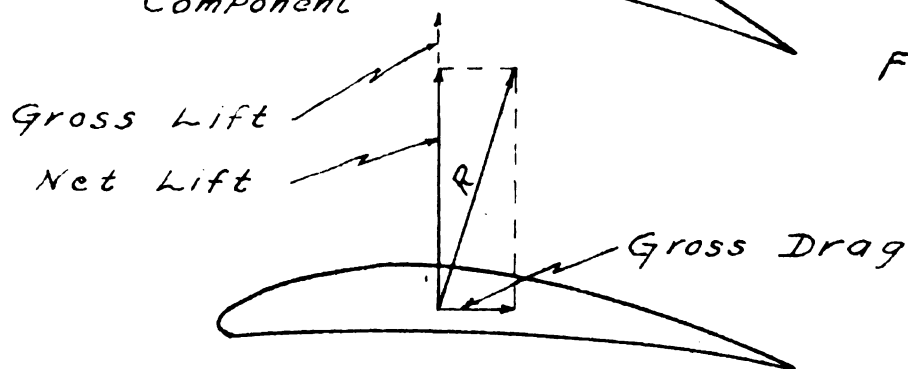
In the high incidence flight condition sketch (b) there is a gross lift component as shown. From this is subtracted the weight of wings giving the net lift component. There is also induced drag making up the drag component. These two forces form the resultant(R). If we now break this resultant up into two components we have

Flight Conditions

Fig (a)



H. I.
Fig. (b)



L. I.
Fig. (c)

the beam lift component and a chord component. The latter acts $\bar{}$ forward and will be considered negative.

In the low incidence sketch (c), the net lift component and drag component are so nearly equal to the beam lift component and chord load respectively that they are considered the same since the wing is almost horizontal in this flight condition. The chord component acts backwards and is considered positive.

In the inverted flight condition the loading is the same as for the L. I. condition except the lift component acts in the opposite direction.

In the dive there is no lift component merely a chord component considered as the weight of ship distributed along the beam.

With these considerations in mind we will proceed with the wing structure analysis.

A - CALCULATING WEIGHTS

(a) WINGS

Length of one wing element = $18'9'' = 225$ in.

Width = 63 in.

Area = $63 \times 225 = 14175$ sq. in.

Tip areas to be subtracted

	$1/3bh = 13 \times 6 \times 9 =$	-----	18.00	sq. in.
	$1/3 \times 53 \times 16.75 =$	-----	295.50	" "
	Space between ailerons $1 \times 98.75 =$	-----	98.75	" "
	" " " end and wing proper			
	$9/32 \times 19 =$	-----	5.30	" "
	Total	-----	416.55	" "

Net wing area $(14175 - 416.6) / 144 = \underline{\underline{95.58}}$ sq. ft.

Total lifting area = $2 \times 95.58 = \underline{\underline{191.16}}$ sq. ft.

*From data on existing wings of this type weight equals 1.10 lb. per sq. ft.

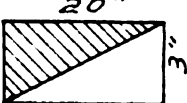
Wing weight = $1.10 \times 191.16 = \underline{\underline{\underline{210.00}}}$ lb.

(b) TAIL SURFACES

(1) Rudder Area	1080.0	sq. in.
$54 \times 20 =$	-----	
$6 \times 24 \times 19/32 =$	-----	147.1 " "
Total	-----	1227.1 " "

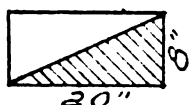
Subtract

	$1/3bh = 1/3 \times 3 \times 6 =$	-----	6.0	sq. in.
---	-----------------------------------	-------	-----	---------

	$(3 \times 20) / 2 =$	-----	30.0	" "
---	-----------------------	-------	------	-----

*Alexander Klemin Airplane Stress Analysis p.116

Rudder Area (Cont.)



$$8 \times 20 / 2 \text{ ----- } 80.0 \text{ sq. in.}$$

$$\text{Total ----- } 116.0 \text{ " "}$$

Net rudder area 1227.1-116.0 -----1111.1 sq. in.

(2) Fin Area

$$1/2(b_1+b_2)h \quad 1/2(27+30)24 \text{ ----- } 684.0 \text{ " "}$$

(3) Stabilizer Area

$$32 \quad 1/2 \times 102 \text{ ----- } 3315.0 \text{ " "}$$

(4) Elevator Area

$$1/2(b_1+b_2)h \quad 1/2(40+54)16 \quad 1/2 \text{ ----- } 775.5 \text{ " "}$$

$$\text{Total tail surface ** ----- } 5880.6 \text{ " "}$$

(5) Tail surface construction is usually taken as weighing 1.0 lb. per sq. ft.

$$\text{Tail surface weight} = 5880.6 / 144 \times 1 = \underline{\underline{40.4 \text{ lb.}}}$$

(c) FUSELAGE WEIGHT

- (*) Spruce -- average wt. = 27 lb. per cu. ft.
- No. 10 gage steel " = 5.63 " " sq. ft.
- " 12 " " = 4.38 " " " "
- " 14 " " = 3.13 " " " "
- " 20 " " = 1.50 " " " "

Bracing wire #30 Music wire = 1.69 lb. per 100 ft.

** 1 3/16" O.D. steel tubing = 1 lb. per ft.

7 strand flexible cable = 2.45 lb. per 100 ft.

(1) Longerons Material 1 1/4" x 1 1/4" x 194.9"

$$1 \frac{1}{4} \times 1 \frac{1}{4} \times 194.9 \times 27 / 1728 \text{ ----- } 4.73 \text{ lb.}$$

*Op. cit. p. 4

**Aviation Mechanics Vol. 1 No.1, p. 80

FUSELAGE WEIGHT (Cont.)

(2) Struts (Rear fuselage)			
(1 1/8x1 1/8x146)	27/1728	-----	2.82 lb.
(3) Struts (Front fuselage)			
(1 1/4x2 3/4x209)	27/ 1728	-----	15.59 "
(4) Skid Truss			
(1 1/4x1 1/4x228)	27/1728	-----	5.56 "
(5) Plywood for Skid 3/32"			
Area = [(14x112)-102]2	= 2932 sq. in.		
Plywood Gusset Plates for			
Fuselage Joints			
Area 2(8x3x4)	-----	192 "	"
		3124 "	"
			Total
* 3/32" Plywood	= 3.2 oz./sq. ft.		
Glue (Casein)	= 8.0 oz./ " "		
Total Wt.---	11.2 "	" "	"
Wt. of The Above Plywood			
(11.2x3124)/16x144	-----	15.20	"
(6) Ash Wood			
Ash Clamp			
2@2x3/4x26x41/1728	-----	1.99	"
Ash Clamp			
2@ 3x3/4x9x41/1728	-----	0.96	"
Ash Skid			
2@3x3/4x44x41/1728	-----	4.91	"
Ash Blocks			
9@3x2x1/2 x3/4x41/1728	-----	4.80	"
Ash Block			
1@6x10x1/2x3/4x41/1728	-----	0.53	"
Ash Rudder Bar			
1@19 1/2x1 1/4x7/8x41/1728	-----	0.71	"
Ash Seat Board			
1@11x11x3/4x41/ 1728	-----	2.15	"

* Niles and Newell, Airplane Structures p. 379

Fuselage Weight (Cont.)

(7) Control Unit

1 3/16" O.D. Steel Tubing (21 11/16)x1/12 -----	1.75 lb.
Torque Tube 1 3/16" Steel Tubing (25 5/8)x1/12 -----	2.09 "
Control Stick Yoke 10 ga. C.R.S. ([(11 1/2 x 3) - 4(4 1/4 x 7/8)] / 144) x 5.63 -----	1.17 "
Torque Tube Bearing Strap 10 ga. C.R.S. [(3 @ 8 1/2 x 3/4) / 144] x 5.63 -----	0.75 "
Rocker Arm 10 ga. C.R.S. [(10 x 1 3/8 x 4) / 144] x 4.38 -----	2.15 "
Elevator Sheave Yoke Pattern 12 ga. C.R.S. [(1 5/8 x 4 1/2) / 144] x 4.38 -----	2.23 "
Six Sheaves 2 in. Dia. 6 x 6/16 -----	2.25 "
Wing Mount Plates 12 ga. C.S. 4 @ [2 1/4 (3 1/4 - 3/4) 1/2 / 144] x 4.38 -----	0.55 "
Longeron Joint Plate 10 ga C.S. 2 @ [(5 1/2 x 1 1/8) / 144] x 5.63 -----	0.54 "
Front Strut Clamps 10 ga. C.S. 2 @ [(2 5/8 x 3 3/16) / 144] x 5.63 -----	0.65 "
Rear Strut Clamps 10 ga. C.S. 2 @ [(3 x 5) / 144] x 5.63 -----	1.17 "
Launching Hook 10 ga. C.S. 2 @ [1 x 8 / 144] x 5.63 -----	0.63 "
Wing Strut 1 1/4" 20 ga. Steel Tube 4 @ 10 2 1/2 ft. x 0.056 -----	2.35 "
Brace Wire Links 14 ga. C.S. 20 @ [(7/8 x 2 7/8) / 144] x 3.13 -----	1.10 "
Wheels Plus Tubing and Axle -----	4.00 "

FUSELAGE WEIGHT (Cont.)

(7) Control Unit (Cont.)

Control Cables (7 strand flexible cable)	
$[(72+68)/100] \times 2.45$	3.43 lb.
Bracing Wire #30 Music Wire	
$[(96)/100] \times 169$	1.62 "
(8) Man Weight	165.00 "
(9) Add for Incidentals	57.00 "
Total wt. of Structure	555.38 "
Call It	555.00 "

B - GENERAL DATA ON GLIDER

(a) Gross Weight of Glider (Man included)----	555.00 lb.
(b) Wt. of Wings (1.10 lb. per sq.ft.) -----	210.00 "
(c) Net Wt. of Glider(Gross Wt.-Wing Wt.)----	345.00 "
(d) Total Span -----	37'- 9"
(e) Length of Sections	
(1) Length of Center Section -----	3 in.
(2) " " Bay -----	116.0 in.
(3) " " Overhang -----	109.0 "
(f) Airfoil ¹ Section Type -----	Modified U.S. 27
(g) Cord Length -----	63 in.
(h) Angle of Incidence -----	0
(i) Dihedral -----	1
(j) Area of Wings (Ailerons included) -----	191.2 sq.ft.
(k) Location of Wing Spars in Percent From Leading Edge of Wing:	
Front Spar 9/63x100 -----	12.8
Rear " 44/63x100 -----	70.0

(l) * Center of Pressure in Per Cent of Chord From Leading edge of wing.

By actual testing of different airfoils in wind tunnel it has been found that the center of pressure for an airfoil of the U.S. 27 type is as follows:

H. I.	-----	30
I. P.	-----	30
L. I.	-----	50

(m) Load Factors Required.

In an airplane structure the load factor at low incidence should not be less than 3 nor should the load factor for inverted flight be less than 2. The low incidence load factor is usually taken as 65 per cent of the ^{high} incidence value and that of inverted flight as 40 per cent of high incidence. Load factor values then become:

H.I.	3/.65	-----	4.6	Use	5.00
Then L. I.	5x.65	-----			3.25
I. P.	5x.40	-----			2.00

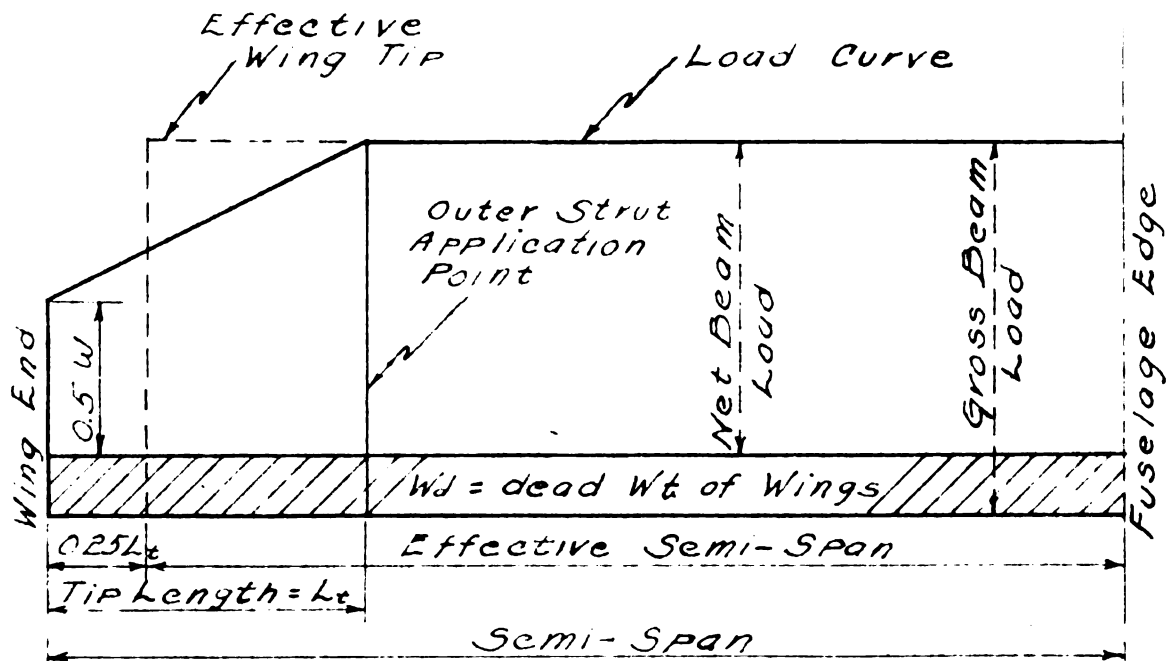
(n) * Ratio of Chord to Beam Lift Component for U.S. 27 Airfoil at:

H.I.	-----	0.175
L.I.	-----	0.15
I. P.	-----	0.0

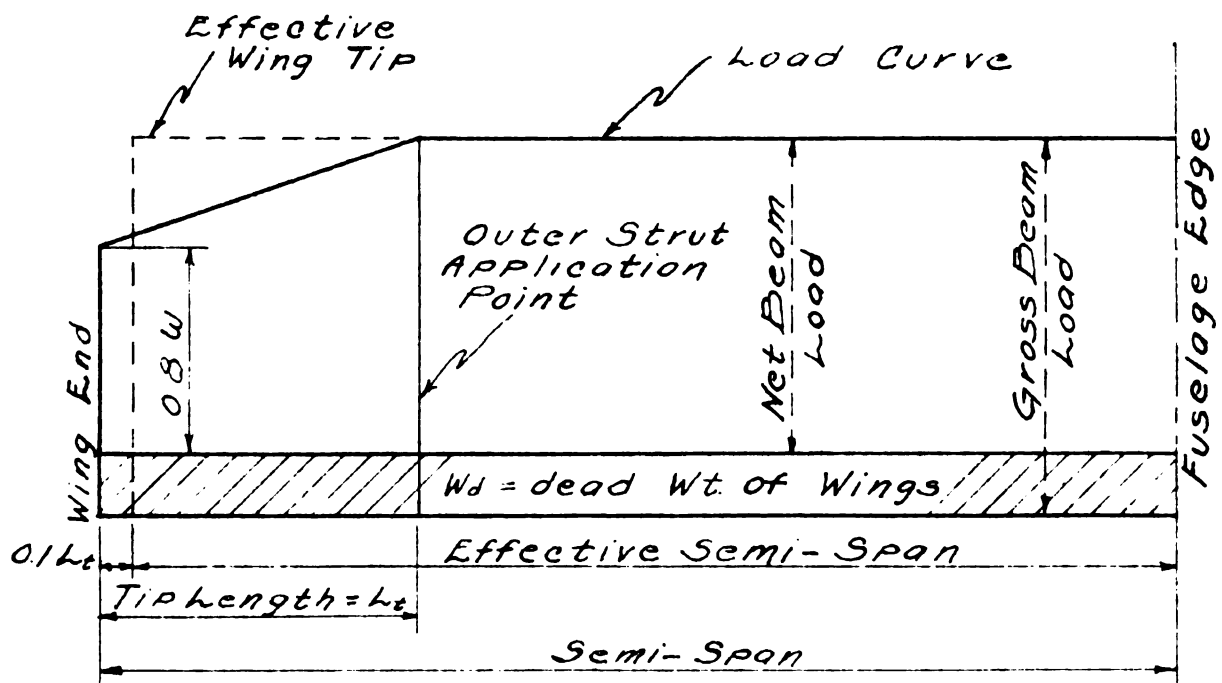
(o) Discussion on Wing Loading.

Since there is no experimental data on the wing tip of ^{the} wing under consideration the usual procedure is to analyze the forces as follows:

* Alexander Klemin, Airplane Stress Analysis p. 118.



*Fig. 1

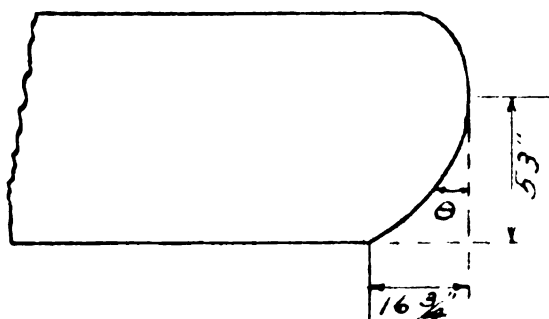


*Fig. 2

If the wing is of a monoplane type varying neither in plan form or thickness and is externally braced so that outer strut point is approximately one chord length from the wing tip use the loading shown in Fig. 1.

Since the glider wing under question has the outer strut point located 109 inches from wing tip and ~~the~~^a chord length equal to 63 inches, it is commonly agreed that more severe stress will be set up due to this fact than the previous loading diagram would give. When the structure is of such nature that strut point is at a distance greater than one chord length from wing tip use the loading shown in Fig. 2 to design the spar sections outboard of the outer inflection point in the bay.

These loading conditions apply to a rectangular wing structure but since in our case we have a negative rake on the wing tip of 17.5°. But according to Warner, Airplane



$$\tan \theta = \frac{16.75}{53} = .316$$

$$\theta = 17.5^\circ$$

Fig. 3

Design unless the changing of tip form doesn't extend inward more than one chord length from tip the lift increase and center of pressure change is so small it can hardly be

given any consideration. Therefore for stress analysis the slight negative rake will be neglected.

In summarizing the above loading conditions it can be said that Fig. 1 loading is used in stress analyzing the members inboard of the outer point of inflection. For stresses outboard of this inflection point use Fig. 2 loading. For future reference call:

Case No./----- Fig. 1 loading
Case NO.2----- " 2 "

In clarifying these loading condition it might be said, what is actually done is to calculate an effective wing which will naturally be smaller than the actual size due to the fact that some air pressure is lost by air slipping off the wing tip. To transform this effective wing to the actual wing we have the slope of load curve from W to $0.5W$ and $0.8W$ in Fig. 1 and Fig. 2 respectively. Also the effective wing is somewhat less in length than the actual since it is decreased $0.25L_c$ and $0.10L_c$ in Fig. 1 and 2 respectively.

Since the overhang from strut point to wing tip is greater than one chord length (wing width) both cases 1 and 2 will have to be used.

C - WING LOADS

(a) Calculate effective Semi-span S_e

For case 1 --- $S_e = 225 - 0.25 \times 109$ ----- 197.62 in.
" " 2 --- $S_e = 225 - 0.10 \times 63$ ----- 218.90 "

(b) Dead Weight of wings per lineal Inch

$$W_d = 210/225 \times 2 \text{ ----- } 0.47 \text{ lb. per lineal in.}$$

(c) Normal Gross Beam Load

$$\begin{aligned} \text{Case 1 : } W_g &= 555/(197.62 \times 2) = 1.40 \text{ lb. per lineal in.} \\ \text{Case 2 : } W_g &= 555/(218.90 \times 2) = 1.27 \text{ " " " "} \end{aligned}$$

(d) Normal Net Load On Beam

$$\begin{aligned} \text{Case 1 : } W_n &= 1.40 - 0.47 \text{ ----- } 0.93 \text{ lb. per lineal in.} \\ \text{Case 2 : } W_n &= 1.27 - 0.47 \text{ ----- } 0.80 \text{ " " " "} \end{aligned}$$

(e) Check These Values

$$\begin{aligned} \text{Case 1 : } 2(0.93 \times 197.62) + 210 &\text{ ----- } 577.58 \text{ lb.} \\ \text{Case 2 : } 2(0.80 \times 218.90) + 210 &\text{ ----- } 560.00 \text{ "} \end{aligned}$$

These check values are somewhat more than the actual gross load of 555 lb. but they produce more severe stresses and are on the safe side. Therefore they will be used.

(f) Calculated Chord Loads

Case - 1					
Flight Condition	Beam Load Per Lin. in.	Chord Beam Ratio	Chord Load Per Lin. in.	Load Factor	Actual Chord Load Per Lin. In.
H.I.	0.93	-0.175	-0.173	5.00	-0.865
L.I.	0.93	0.15	0.140	3.25	0.455
I.F.	0.93	0.00	0.00	2.00	0.00
Dive				1.00	0.77

In a dive the net weight of plane would be distributed along the chord and the load factor would be 1.

$$\begin{aligned} \text{In the above table the dive chord load} \\ (555 - 210)/225 \times 2 \text{ ----- } 0.77 \text{ lb. per lin. in.} \end{aligned}$$

(g) Calculated Distribution of Loads On the Spars

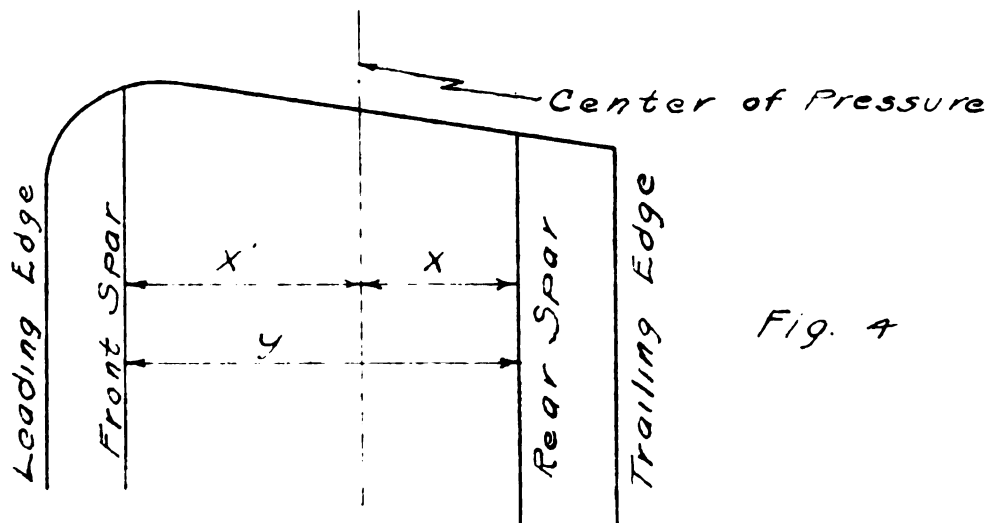


Fig. 4

Front spar takes $x/y \times 100 =$ per cent

" " " $\frac{\text{Rear spar location-center of pressure}}{\text{Rear spar location-front spar location}}$

Rear spar takes $x'/y \times 100 =$ per cent

" " " $\frac{\text{Center pressure-front spar location}}{\text{Rear spar location-front spar location}}$

Then for Case 1 and 2 at H. I. and I. F.

Front spar takes $\frac{70-50}{70-12.8} \times 100 = 69.93 \text{ o/o}$

Rear spar takes $\frac{30-12.8}{70-12.8} \times 100 = 30.07 \text{ o/o}$

For L. I. Condition

Front spar takes $\frac{70-50}{70-12.8} \times 100 = 52.62 \text{ o/o}$

Rear spar takes $\frac{50-12.8}{70-12.8} \times 100 = 65.03 \text{ o/o}$

The above values represents the percentages of the load carried by each spar under the varies conditions. The variation in the load on the spars is due to the travel of the center of pressure toward the leading edge of wing as the angle of incidence increases.

(h) Beam Loads Per Lineal Inch on Spars

Flight Condition	Case	Spar	Net Load Per in. On Wing	Load Factor	o/o Carried by Spar	Net Load per Lin. In.
H.I.	1	Front	0.93	5.00	69.93	3.25
		Rear	0.93	5.00	30.07	1.40
	2	Front	0.80	5.00	69.93	2.80
		Rear	0.80	5.00	30.07	1.20
L.I.	1	Front	0.93	3.25	52.62	1.58
		Rear	0.93	3.25	65.03	1.96
	2	Front	0.80	3.25	52.62	1.37
		Rear	0.80	3.25	65.03	1.69
I.F.	1	Front	0.93	2.00	69.93	-1.29
		Rear	0.93	2.00	30.07	-0.56
	2	Front	0.80	2.00	69.93	-1.01
		Rear	0.80	2.00	30.07	-0.48

Net load per in. = Column (4x5x6)

(1) Compute Moments, Shears and Reactions

Case No. 1 (Unit Loading)

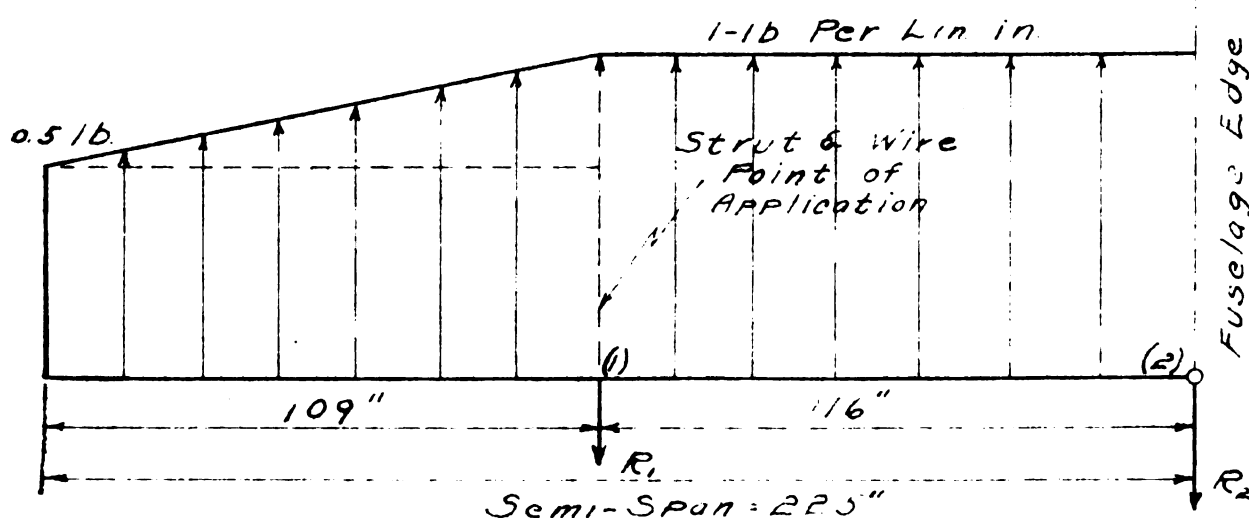


Fig. 5

By placing a unit loading of one pound per lineal inch on spar

$$M_s = (109 \times 0.5 \times 109 / 2) + (109 \times 0.5 / 2 \times 109 / 3) = 3,960.33 \text{ in. lb.}$$

$$M_r = 0$$

Shear right and left of reactions

$$V_L = 109 \times (0.5 + 1) / 2 \text{ ----- } 81.75 \text{ lb.}$$

$$V_R = -3960.33 / 116 - 116 \times \frac{1}{2} \text{ ----- } -92.14 \text{ lb.}$$

$$R_s = \text{Total} \text{ ----- } 173.89 \text{ "}$$

$$V_{L_R} = 116 \times \frac{1}{2} - 3960.33 / 116 \text{ ----- } 23.86 \text{ "}$$

$$V_{R_R} = 0$$

$$(\text{Check}) R_s + R_R = 173.89 + 23.86 = 81.75 + 116 \times 1 = 197.75$$

Case No. 2 (Unit Loading)

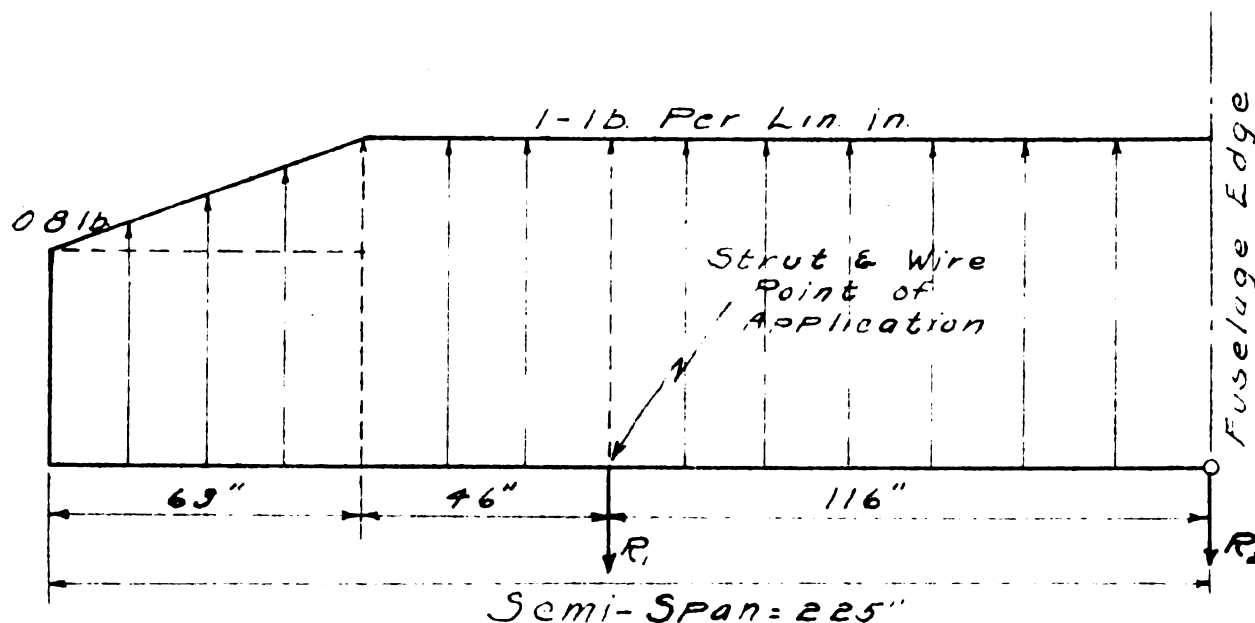


Fig 6

$$M_1 = 63 \times .8(46+63/2) + 0.2 \times 63/2(46-63/3) + 46 \times 1 \times 46/2$$

-----5,452.25 in. lb.

$$M_2 = 0$$

Shear right and left of reactions.

$$V_{L1} = 63 \times (0.8+1)/2 + 46 \times 1 \text{ ----- } 102.70 \text{ lb.}$$

$$V_{R1} = -5,452.25/116 - 116 \times \frac{1}{2} \text{ ----- } \underline{-105.00} \text{ " "}$$

$$R_1 = \text{Total} \text{ ----- } 207.70 \text{ " "}$$

$$V_{L2} = 116 \times \frac{1}{2} - 5452.25/116 \text{ R ----- } 11.00 \text{ "}$$

$$V_{R2} = 0$$

$$\text{Check } R_1 + R_2 = 207.70 + 11.00 = 102.7 + 116 = 218.70 \text{ lb.}$$

To obtain the actual bending moments, shear and reactions multiply the unit values just found by the loading per lineal inch as shown in table under article (h).

TABLE NO. III

Moments

Case	Spar	M, 1 lb./lin. in.	H.I.		L.I.		I.F.	
			Load per Lin. In.	M.	Load per Lin. In.	M.	Load per Lin. In.	M.
1	Front	3,960.73	3.25	12,871.07	1.58	6,257.32	-1.29	-5,108.83
	Rear	3,930.33	1.40	5,544.46	1.96	7,762.25	-0.56	-2,217.78
2	Front	5,452.25	2.80	15,266.30	1.37	7,469.58	-1.01	-5,506.77
	Rear	5,452.25	1.20	6,542.70	1.69	9,214.30	-0.48	-2,617.08

Shear

Case	Spar	V _{R1} (Max.) 1 lb./lin. in.	H.I.		L.I.		I.F.	
			Load per Lin. In.	V _{R1}	Load per Lin. In.	V _{R1}	Load per Lin. In.	V _{R1}
1	Front	-98.14	3.25	239.46	1.58	145.58	-1.29	118.86
	Rear	98.14	1.40	129.00	1.96	180.59	-0.56	51.60
2	Front	-105.70	2.80	295.96	1.37	144.81	-1.01	106.76
	Rear	105.70	1.20	126.84	1.69	173.63	-0.48	50.74

Reactions

Case	Spar	1 lb./lin. in.		H.I.		L.I.		I.F.	
		R ₁	R ₂	Load per Lin. In.	R ₁	R ₂	Load per Lin. In.	R ₁	R ₂
1	Front	173.89	23.86	3.25	565.14	77.55	1.58	274.75	37.70
	Rear	173.89	23.86	1.40	243.45	33.40	1.96	340.82	46.77
2	Front	207.70	11.00	2.80	581.56	30.80	1.37	284.55	15.07
	Rear	207.70	11.00	1.20	249.24	13.20	1.69	351.01	13.53
								-224.32	30.78
								-97.38	13.36
								-209.78	11.11
								-99.62	5.28

(j) Nose Dive Condition

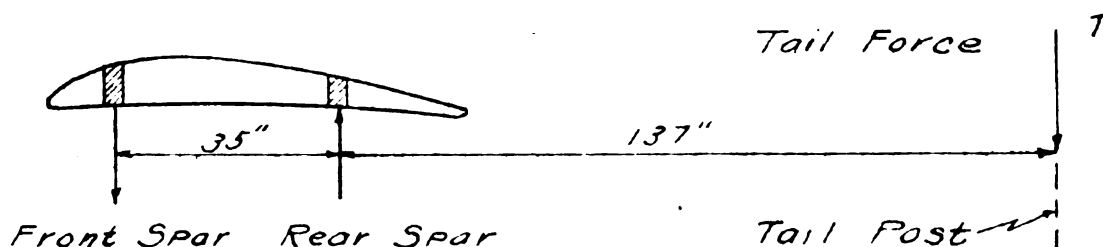


Fig. 7

In the nose dive condition the three forces shown in Fig. 1 keeps the structure in equilibrium. If we assume the loading on the front spar being the same as for the I.F. condition then:

$$\sum M = 0$$

$$137(R.S.) = 172 (F.S.)$$

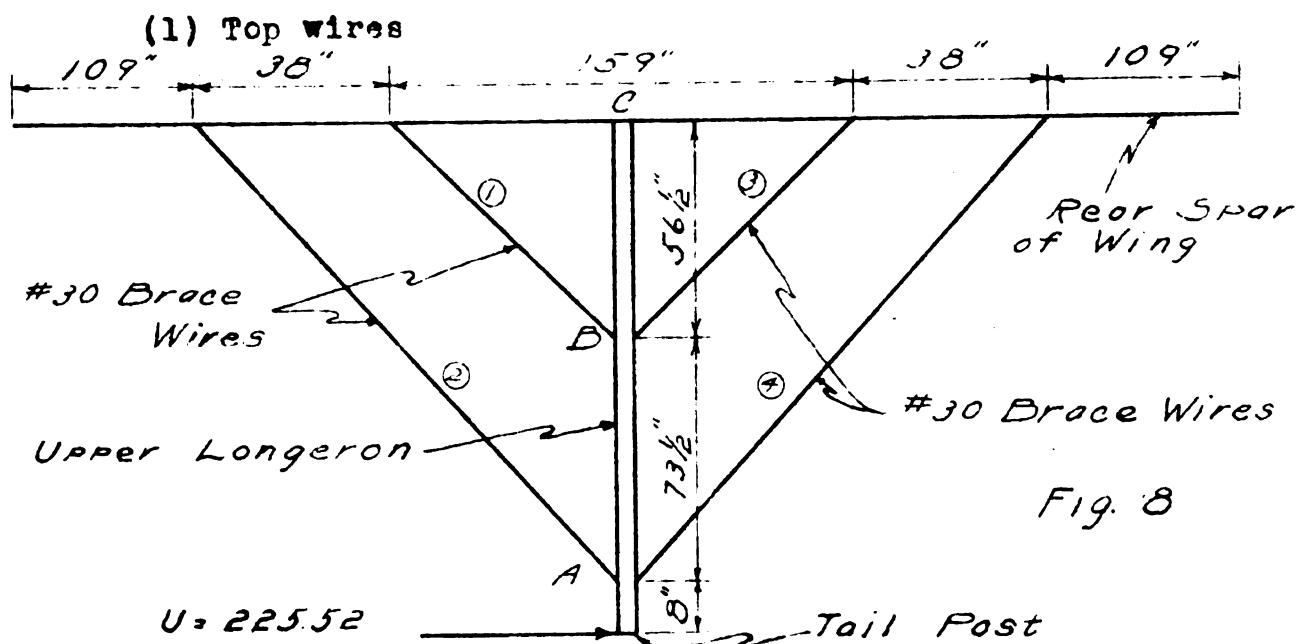
$$R.S. = 1.255 (F.S.)$$

By using the worst loading on the front spar in I.F. condition (Table II) the following results are obtained.

TABLE NO. IV

		I.F. and Nose Dive (Front Spar)	Nose Dive Rear Spar -1.255 F.S.
Case No. 1	Load /Lin. in.	-1.22	1.58
	M ₁	-5108.83	6,257.32
	R ₁	-224.32	274.75
	R ₂	30.72	37.69
Case No. 2	Load/Lin. In.	-1.01	1.24
	M ₁	-2,217.78	6,760.79
	R ₁	209.78	257.55
	R ₂	11.11	13.64

(k) Stresses In Tail Guide Wires.



Calculate the force P on fin and rudder

From-----Sect. A(Art.b,1 and2)

Rudder area	-----	1111.1	sq.in.
Fin	"	684.0	" "
Total vertical tail area	-----	1795.1	" "

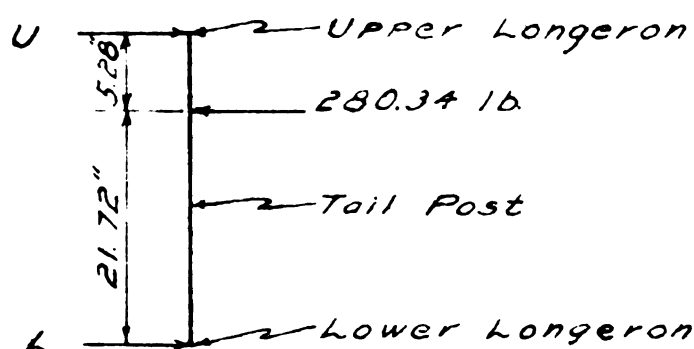
*Force on vertical surfaces 22.5 lb./sq. ft.

$P = 1795.1 \times 22.5 / 144$ ----- 280.34 lb.

Area above upper longeron			
3x26/2	39	sq.	in.
21x26	546	"	"
2x20	40	"	"
Total	-----	665	" "

$44X = 232.55$

$X = 5.28$ inches down from upper longeron is a point where there is as much fin and rudder area above as below. This is the point of application for the force P on tail post.



$$\begin{aligned}
 27U &= 280.34 \times 21.72 \\
 U &= 280.34 \times 21.72 / 27 \quad \underline{\underline{225.52 \text{ lb.}}} \\
 L &= \underline{\underline{54.82 \text{ lb.}}}
 \end{aligned}$$

Since the longeron is a flexible beam the slight bending resistance offered by the butt joint at trailing edge of wing is neglected. The structure in Fig. 8 is, however, indeterminate; therefore, it will be investigated in the following manner.

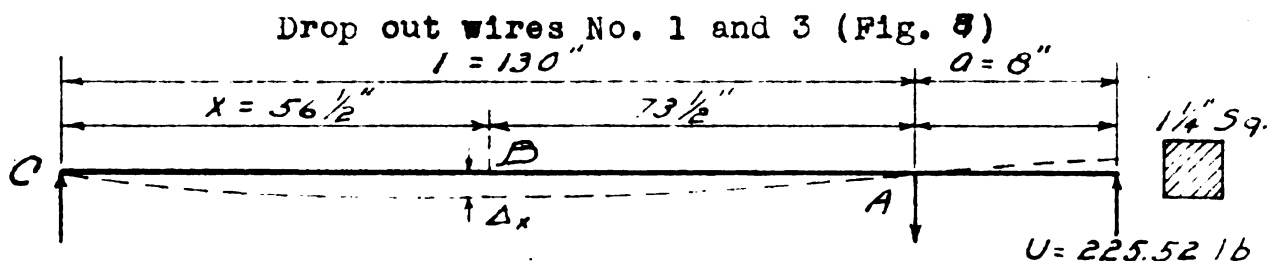


Fig. 9

$$\begin{aligned}
 I &= d^4/12 \quad (1/4)^4/12 \\
 &= 0.2034 \text{ in}^4
 \end{aligned}$$

$$\Delta_x = \frac{Uax}{6EI} (1^2 - x^2) = \frac{225.52 \times 8 \times 56 \frac{1}{2}}{6 \times 1.2 \times 10^4 \times 0.2034 \times 130} (130^2 - 56 \frac{1}{2}^2) = 7.366 \text{ in.}$$

Then solve for the horizontal component at B_H by substituting into

$$\Delta_x = \frac{B_H a^2 b^2}{3EI}$$

$$B_H = \frac{3EI \Delta_x}{a^2 b^2} = \frac{7.366 \times 3 \times 1.2 \times 10^4 \times 0.2034 \times 130}{(56 \frac{1}{2})^2 - (73 \frac{1}{2})^2} = 41.31 \text{ lb.}$$

(See Fig. 9')

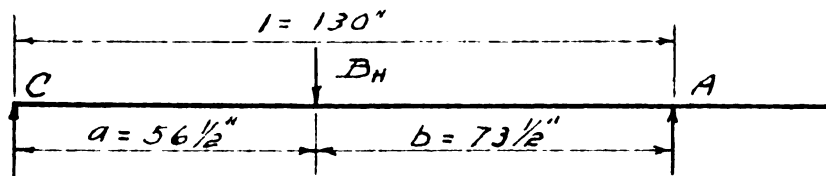


Fig. 9'

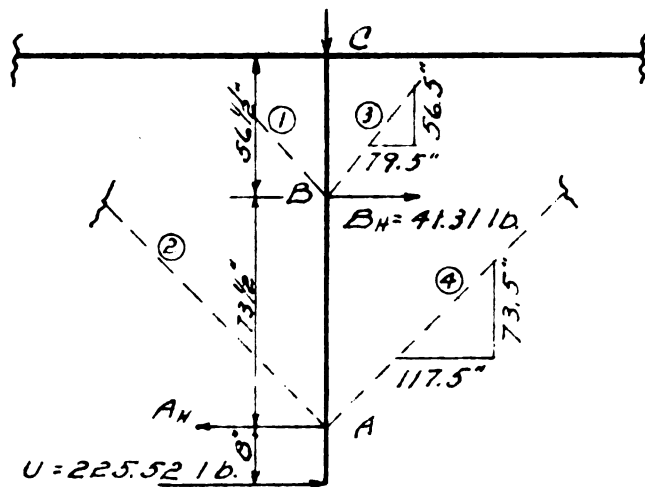


Fig. 10

$$\sum M_C = 0$$

$$130A_H - 41.31 \times 56\frac{1}{2} - 225.52 \times 138 = 0$$

A_H	-----	257.37 lb.
$A_H(73.5/117.50)$	-----	161.11 "

$$\sum M_A = 0$$

$$41.31 \times 73.5 - 130C_H - 8 \times 225.52 = 0$$

C_H	-----	9.46 "
(Check) $41.31 + 225.52$	-----	266.83 "

Force in wire No. 1 and 2; #30 music wire

$$F_1 = (97.5/79.5)41.31 = 50.59 \text{ lb.}$$

$$F_2 = (134.9/117.5)257.37 = 295.10 \text{ lb.}$$

Stresses

$$S_1 = 50.59/.00503 \text{ ----- } 1,005.7 \text{ lb./sq. in.}$$

$$S_2 = 295.1/.00503 \text{ ----- } 58,703.0 \text{ " " "}$$

$$\text{Maximum allowable stress } 165,000 \text{ " " "}$$

(1) Stresses in wing struts

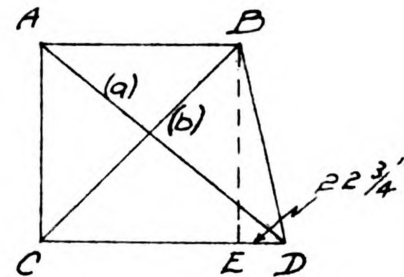
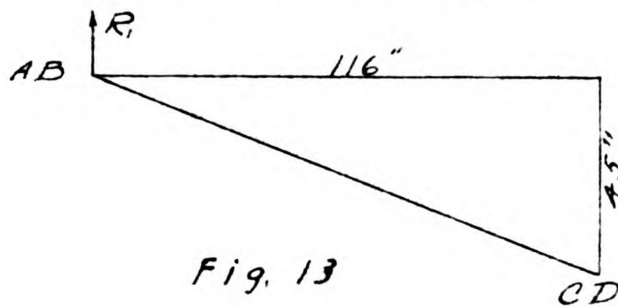


TABLE NO. V

Length and Components of Wing Struts

Member	V	H	D	V	H	D	L	V - H	- D	- L
Front---(AC)	45	116.0	0.0	2025	13595.56		15620.56	124.98		
Rear---(BD)	45	116.0	22.75	"	"	517.56	16138.12	127.04		

Front L/V 124.98/45 2.777

Strut H/L 116.0 /124.98 0.933

D/L 0.0 /124.98 0.0

Rear L/V 127.04/45 2.822

Strut H/L 116.0 /127.04 0.917

D/L 22.75/ 127.04 0.179

Loads in Wing Struts - Spar and Drag Loads

Case	Load- ing Con- dition	Reac- tion R	Load In Strut R(L/V)	Axial Spar Loads (H/L)(Col.5)	Drag Loads (D/L)(Col.5)
1	Front	H.I.	565.14	1569.39T	1464.24C
		L.I.	274.75	762.98T	711.77C
		I.F.	-224.32	-623.61C	581.83T
		N.D.			0.000
	Rear	H.I.	243.45	687.02T	630.00C
		L.I.	340.82	961.79T	881.96C
		I.F.	-97.38	274.81C	251.99T
		N.D.	274.75	775.34T	710.99C
2	Front	H.I.	581.56	1614.99T	1506.89C
		L.I.	284.55	790.20T	737.26C
		I.F.	-209.78	581.96C	542.96T
		N.D.			
	Rear	H.I.	249.24	703.36T	644.98C
		L.I.	351.01	990.94T	908.69C
		I.F.	-99.69	-281.32C	257.97T
		N.D.	257.55	726.80T	665.29C

In the above table the front strut is in the same vertical plane as the front spar therefore it causes no drag load due to lift.

(1) From TABLE NO.V (Front Strut)

Greatest strut tension force ----- 1614.99 lb.

$S = 1614.99 / 0.1336$ ----- 12,052.10 lb./sq. in.

Allowable stress ----- 55,000 " " "

Factor of Safety $55,000 / 12,081.1$ ---- 4.57

Since this is a long slender tube Euler's Formula will apply in the case of compression strength.

Greatest compression force ----- 623.61 lb.

$$P = C \pi^2 EI / L^2$$

$$C = 1$$

$$E = 28 \times 10^6$$

$$I = 0.02467$$

$$L = 124.98$$

$$P = 1 \times 28 \times 10^6 \times 0.02467 / 124.98^2$$

4,364.6 lb.

Factor of Safety $4,364.6 / 623.61$ ---- 7.02

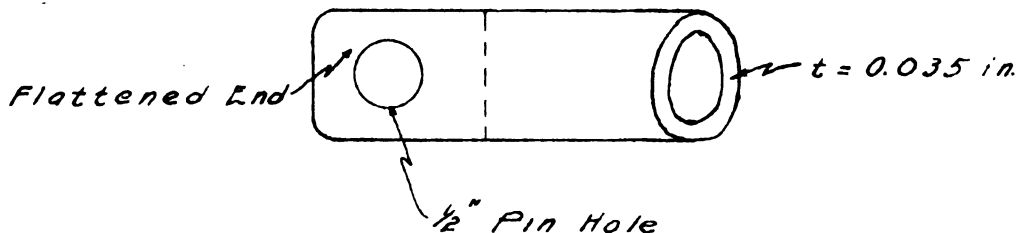
(2) Check $\frac{1}{2}$ " strut pin for shear

$$v = (1614.99) / \left(\frac{0.5}{4} \right) \times \text{sq. in.} \text{ -- } 8,210 \text{ lb./sq. in.}$$

Allowable = 35,000 " " "

Factor of Safety $35,000 / 8,210 = 4.26$

(3) Check pin End of Tube for Shear and Bearing



Area of a 20 gage $1\frac{1}{4}$ " O.D. tube ----- 0.1336 sq. in.

Subtract $\frac{1}{2}$ " pin hole

Shear area = $0.1336 - (\frac{1}{2} \times 0.035 \times 2)$ --- 0.0986 " "

$v = 1614.99 / 0.0986 = 16,390$ lb./sq. in.

Allowable shear for mild carbon steel
equals ----- 45,000 " " "

Factor of safety = $45,000 / 16,390$ ----- 2.75

Bearing force = $1614.99 / \frac{1}{2} \times 2 \times 0.035$ -- 46,200 lb./sq. in.

Allowable bearing stress = 90,000 " " "

Factor of safety = $90,000 / 46,200$ ----- 1.95

(4) Rear Strut

In discussing the rear strut let us consider the left wing. Referring to Fig. 5 there might be a time when left rudder is used which would put wires No. 2 and the lower tail brace wires in tension as calculated. However, to find the worst tension force on the rear strut these forces are considered zero because there might be a condition where right rudder was used putting the right hand wires in tension.

From TABLE NO V the greatest tensile force (rear strut)
equals ----- 990.94 lb.

Since this force is less than the force on front strut and from the fact that both struts are of the same material and crossection the latter has a greater safety factor in tension than the front strut.

In testing for compression strength, however, the tension in the lower tail brace wire produces a downward reaction.

$$\begin{array}{rcl}
 Q & = & 16.48 \text{ lb. } \left[\text{see Sect.-C-Art. k(2)} \right] \\
 P & = & 16.48 \times L/V = 16.48 \times 2.822 \text{ ----- } 46.51 \text{ lb.} \\
 \text{Load in strut} & \text{---} & \text{----- } 281.32 \text{ " } \\
 \text{Total load in strut} & \text{---} & \text{----- } 327.83 \text{ " }
 \end{array}$$

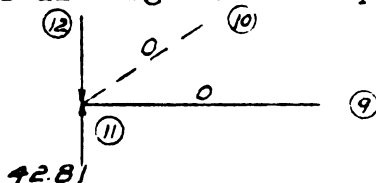
This force is also less than the compressive force solved for in the case of the front strut. Therefore, no further investigation will be needed. A strut of less crossection could have been used here. In calculating the forces in the struts no consideration was made for the stresses taken by the two wires (a) and (b) Fig. 13. These wires no doubt take some stress but in the event that they would become slack or broken the struts should be so designed to carry the load. The wing structure is considered a rigid structure; therefore, it is difficult to predict what deflection if any takes place in the wing thus putting stress in these wires.

D - INVESTIGATION OF INTERNAL WING STRUCTURES

(a) Truss Solution By The Method of Joints (Case No. 2)

Referring to Fig. 14 and starting at joint No. (11)

the following solution applies:

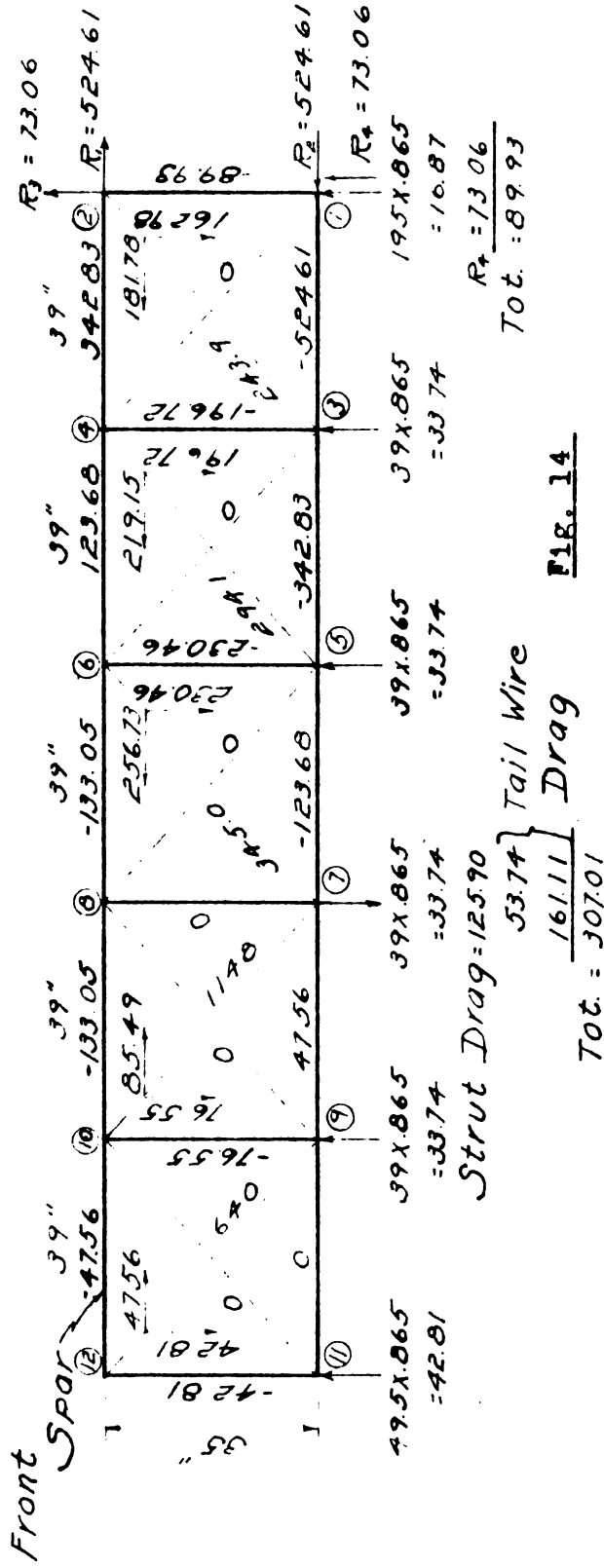


$$\begin{array}{l}
 \sum V = 0 \\
 42.81 - (11-12) = 0 \\
 (11-12) \text{----- } 42.81 \text{ lb. c.}
 \end{array}$$

$$\begin{array}{l}
 \sum H = 0 \\
 (11-9) - 39/35 \times \text{Vert. (11-10)} = 0 \\
 (11-9) \text{----- } 0.00
 \end{array}$$

LOADS ON WING TRUSS DURING HIGH INCIDENCE FLIGHT CONDITION

WITH TAIL WIRE FORCES APPLIED



Panel point loads = panel width multiplied by beam chord component (0.865 lb. per 11n. in.) taken from Table in Sect. C-Art. (g).

$$\sum M = 0$$

$$R_1 = \frac{[(195 \times 42.81) + 33.74(156 + 78 + 39) - (307.01 \times 117)]}{35} = 524.61 \text{ lb.}$$

$$\sum H = 0$$

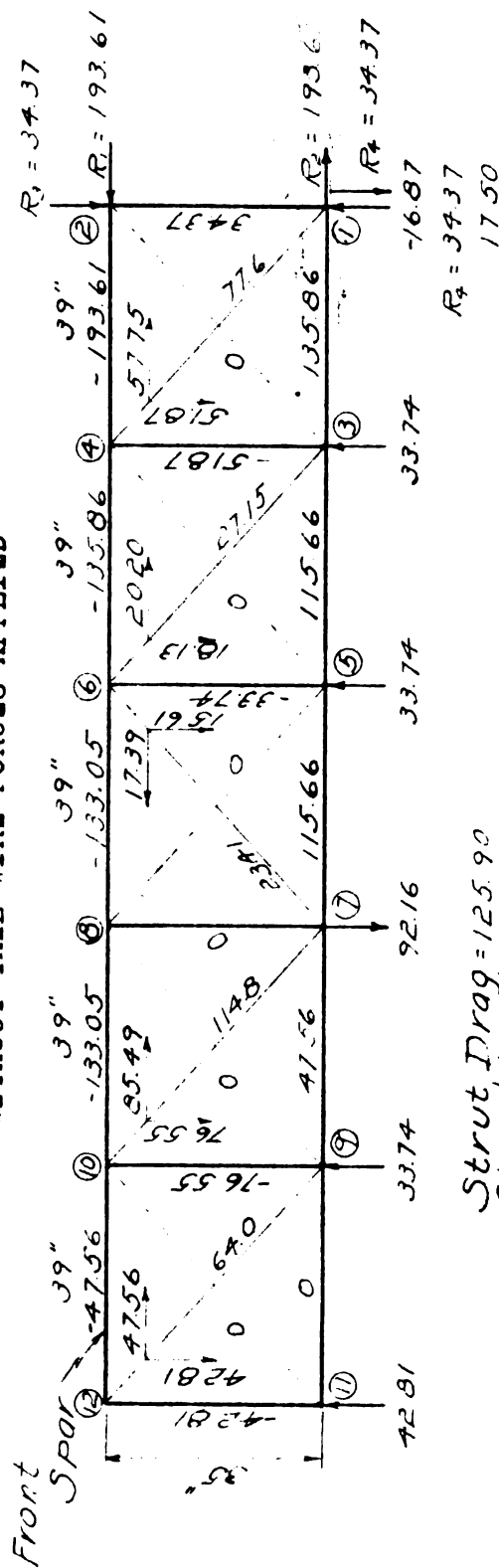
$$R_1 = R_2 = 524.61 \text{ lb.}$$

$$\sum V = 0$$

$$R_3 = R_4 = \frac{[42.81 + (3 \times 33.74) + 16.87 - 307.01]}{2} = 73.06 \text{ lb.}$$

LOADS ON WING TRUSS DURING HIGH INCIDENCE FLIGHT CONDITION

WITHOUT TAIL WIRE FORCES APPLIED



Similar to the solution of Fig. 14

$$R_1 = 193.61 \text{ lb.}$$

$$R_2 = 193.61 \text{ "}$$

$$R_3 = 34.37 \text{ "}$$

$$R_4 = 34.37 \text{ "}$$

LOADS ON WING TRUSS DURING LOW INCIDENCE FLIGHT CONDITION

WITH TAIL WIRE FORCES APPLIED

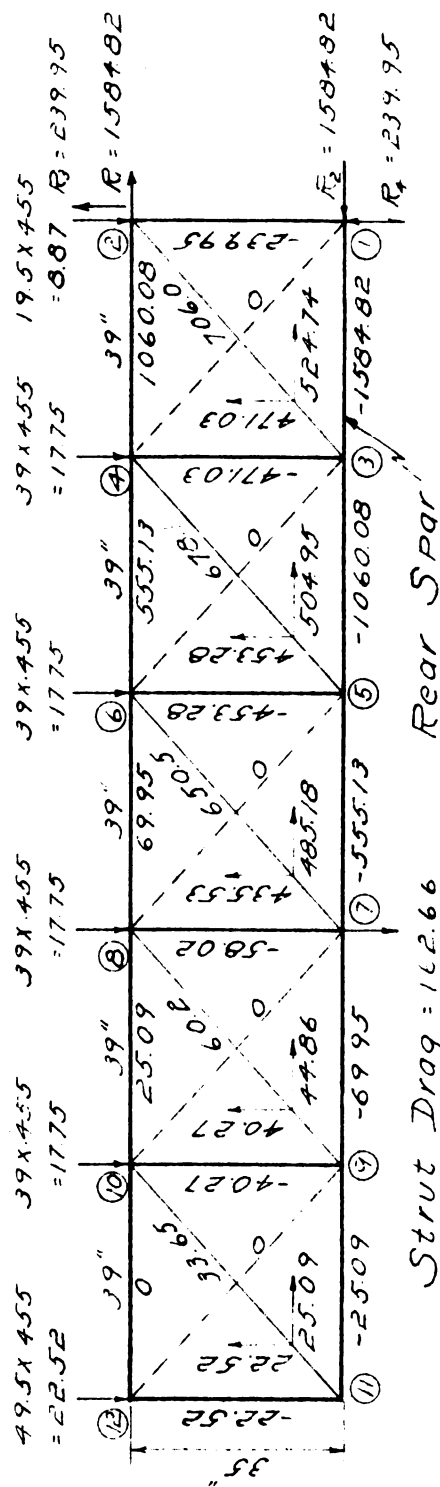


FIG. 16

Similar to the solution of FIG. 14

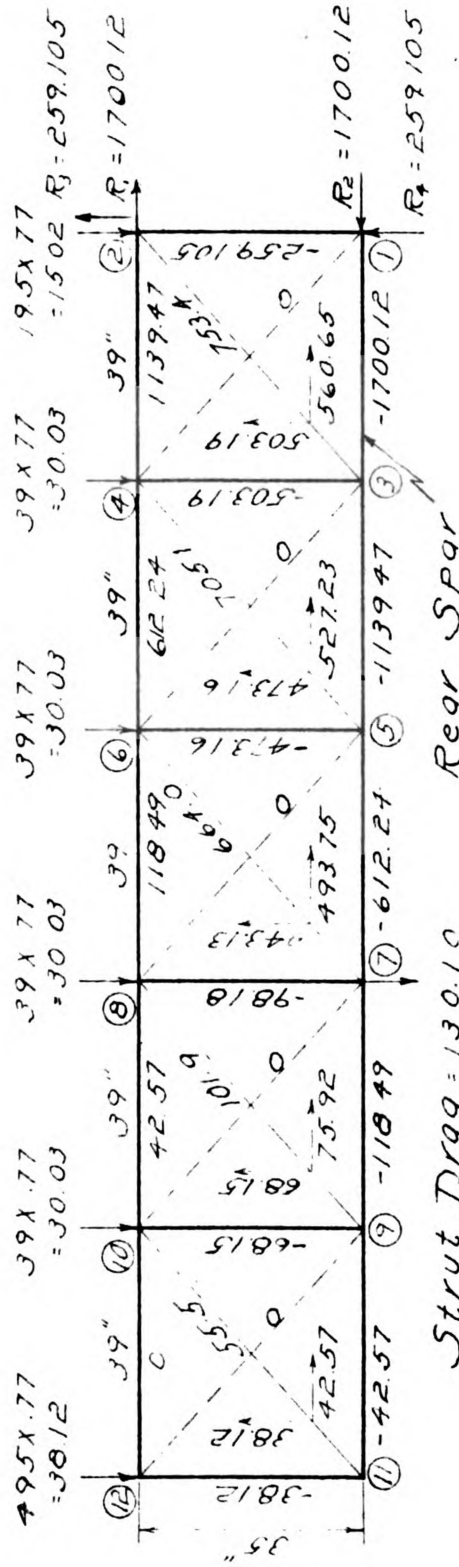
$$R_1 = 1584.82 \text{ lb.}$$

$$R_2 = 1584.82 \text{ "}$$

$$R_3 = 239.95 \text{ "}$$

$$R_4 = 239.95 \text{ "}$$

LOADS ON WING TRUSS DURING DIVE WITH TAIL WIRE FORCES APPLIED



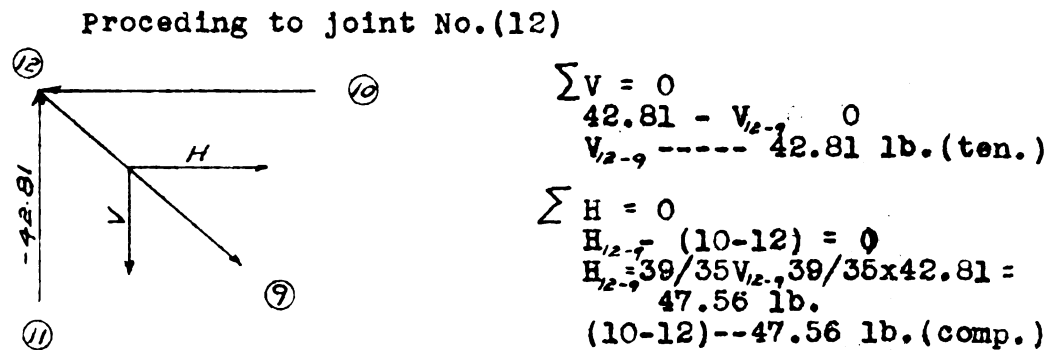
Similar to the solution of Fig. 14

$$R_1 = 1700.12 \text{ lb.}$$

$$R_2 = 1700.12 \text{ "}$$

$$R_3 = 259.105 \text{ "}$$

$$R_4 = 259.105 \text{ "}$$



The remainder of joints were solved in a similar manner.

(b) Truss Solution For Case No. 1

If we solve the wing truss for case No. 1 loading in a similar manner to the solution of truss for case No.2 loading (Fig. 14, 15, 16, 17) we will obtain the necessary information to solve for the maximum bending moment in-board of the inflection point. The solution of truss for case No. 1 yields the following values. Only those values used are listed.

H.I.			
Front	(2 - 4)	335.85 lb.	
Spar	(4 - 6)	119.87 "	
	(6 - 8)	-133.63 "	
H.I.			
Front	(2 - 4)	-203.37 lb.	Rear (1 - 3) 142.24 lb.
Spar	(4 - 6)	-142.24 "	Spar (3 - 5) 118.83 "
	(6 - 8)	-133.05 "	(5 - 7) 118.83 "
L.I.			
Front	(2 - 4)	1081.26 lb.	Rear (1 - 3) -1616.27 lb.
Spar	(4 - 6)	565.71 "	Spar (3 - 5) -1081.26 "
	(6 - 8)	69.95 "	(5 - 7) -565.71 "

The truss for case No.1 loading was not solved as completely as for case No.2 because in checking the stresses inboard of the inflection of the spars, we are interested in the condition that produces the greatest bending moment under the case No. 1 loading as stated in Sect.-B Art.(o). In case of the rear spar if we inspect TABLE NO. VII the condition that will produce the greatest bending moment occurs under H.I. . Therefore to be sure we will investigate as shown in TABLE NO. VII.

(c) Check Stresses In Truss Wires

(1) Drag wires (#30 music wire)

From Fig.(14,15,16,17) the greatest load occurs in wire 2 - 3 (Fig. 17).

$$\begin{aligned} F &= 753.4 \text{ lb.} \\ \text{Area}(\#30 \text{ wire}) &= (\overline{0.08^2}/4) \times \pi \text{ ----- } 0.00503 \text{ sq. in.} \\ S = P/A &= 753.4/0.00503 \text{ ----- } 149,780 \text{ lb./sq. in.} \\ \text{Allowable} &\text{ ----- } 165,000 \text{ " " "} \\ \text{Factor of safety} &= 165,000/149,780 \text{ ----- } 1.01 \end{aligned}$$

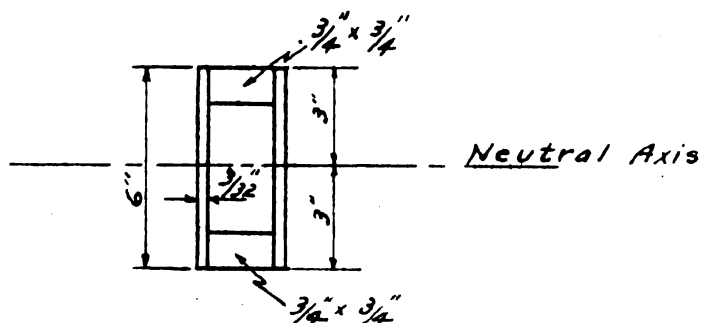
(2) Anti-drag wires (#26 music wire)

$$\begin{aligned} F &= 114.8 \text{ lb.} \\ \text{Area}(\#26 \text{ wire}) &= (\overline{0.63^2}/4) \times \pi \text{ ----- } 0.0031 \text{ sq. in.} \\ S = P/A &= 114.8/0.0031 \text{ ----- } 37,032 \text{ lb./sq. in.} \\ \text{Factor of safety} &= 165,000/37,032 \text{ ----- } 4.45 \end{aligned}$$

B - CHECK THE DESIGN OF THE WING SPARS

(a) Spar Constants(Front Spar)

(1) Position of neutral axis:



Since the top and bottom of flanges are identical the neutral axis occurs at center as shown.

(2) Moment of inertia:

$$\begin{aligned} I(\text{flanges}) &= 2 \left[\left(\frac{1}{12} \times \frac{3}{4} \times 6^3 \right) + \left(\frac{3}{4} \right) \times (2.625)^2 \right] = 7.961 \text{ in.}^4 \\ I(\text{webs}) &= 2 \left[\frac{1}{12} \times \frac{3}{16} \times 3 \times 6^3 \right] = 3.375 \text{ in.}^4 \\ I(\text{total}) &= 11.336 \text{ in.}^4 \end{aligned}$$

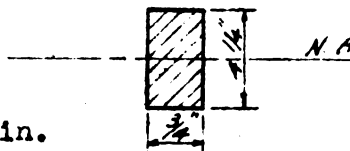
$$\begin{aligned} y/I &= 3/11.336 = 0.265 \\ \text{Area} &= \frac{3}{4} \times \frac{3}{4} + 6 \times \frac{3}{32} = 1.688 \text{ sq. in.} \end{aligned}$$

At strut point and fuselage (spar solid)

$$\begin{aligned} I &= \frac{1}{12} \times \frac{3}{4} \times 6^3 = 13.5 \text{ in.}^4 \\ y/I &= 3/13.5 = 0.222 \\ \text{Area} &= \frac{3}{4} \times 6 = 4.5 \text{ sq. in.} \end{aligned}$$

(b) Spar Constants (rear spar)

$$\begin{aligned} I &= \frac{1}{12} \times \frac{3}{4} \times (4.5)^3 = 5.63 \\ y/I &= (2 \frac{1}{8})/5.63 = 0.377 \\ \text{Area} &= \frac{3}{4} \times 4 \frac{1}{2} = 3.185 \text{ sq. in.} \end{aligned}$$



(c) Axial Load On Spars (In bay)

Case No. 1 TABLE NO. VII					
Flight Condition	Spar	Average Due to Drag	Due to Lift	Due to Tail Wires	Total
H.I.	Front	107.36	-1464.24		-1624.79
	Rear	-323.52	-630.00	-122.98	-626.35
H.I.	Front	-159.55	-1464.24		-1624.79
	Rear	126.63	-630.00	-122.98	-626.35
L.I.	Front	571.64	-711.77		-140.13
	Rear	-1087.72	-881.96	-172.16	-2141.84

Case No. 2
TABLE NO. VI11

Flight Condition	Spar	Average Due to Drag	Due to Lift	Due to Tail wires	Total
H.I. (Fig. 14)	Front	111.15	-1506.89		-1395.74
	Rear	-330.34	-644.98	-326.52	-1301.84
* H.I. (Fig. 15)	Front	-154.17	-1506.89		-1661.06
	Rear	122.39	-644.98	-41.31	-563.90
L.I.	Front	561.72	-737.26		-165.54
	Rear	-1066.67	-908.69	-326.52	-2301.88
N.D.	Front	623.40	542.96		1166.36
	Rear	-1150.61	-665.29	-326.52	-2142.42

Average axial spar load due to drag in TABLE NO.VI1
(See Sect.-D, Art. (b))

Average axial spar load due to drag in TABLE NO.VI11
(See Fig. 14,15,16,and 17)

Axial load due to lift (See TABLE NO. V)

Axial load due to tail wires (See Sect.-C, Art. k)

* In this particular flight condition the tail brace wire No. (1) Fig. 10 produces a_n axial spar load as shown at the second panel point.

(d) Determining the Outer Point of Inflection

By referring to Sect.-C, Art.(i) it becomes obvious that the outer point of inflection is determined by case No. 1 loading because M₁ is less than in case No. 2. Also the loads are larger per inch in case No. 1 thus reducing the moment to zero sooner in the bay.

Point of inflection equals:

$$M_x = M_1 + V_{R1} x + wx^2/2 = 0$$

From Sect.-C, Art. (1)

$$\begin{array}{ll} M, & \text{----- } 3,960.33 \text{ in. lb.} \\ V_R, & \text{----- } -92.14 \text{ lb.} \\ W & \text{----- } 1.00 \text{ "} \end{array}$$

$$3,960.33 - (-92.14X) - X/2 = 0$$

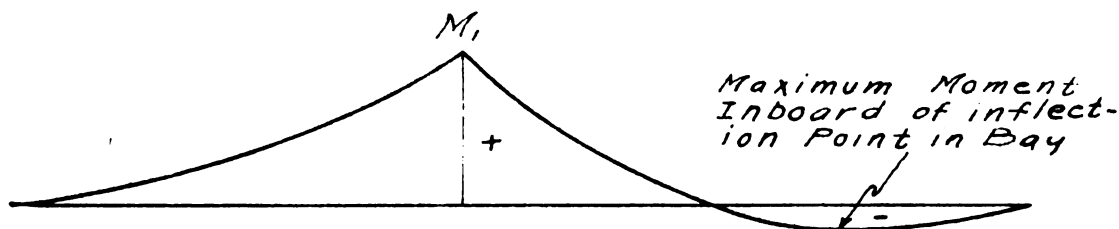
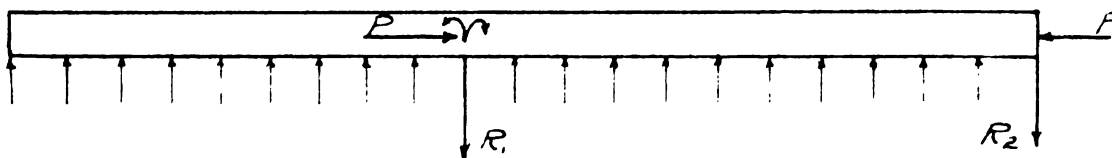
$$X = \frac{184.28 - \sqrt{(184.28)^2 - 4(7920.33)}}{2}$$

$$X = \underline{68.28} \text{ or } 116.00$$

This formula neglects the effect of axial loading on the spar but for the purpose of determining the inflection point the above values are accurate enough.

(e) Determining the Maximum Moments in Bay

Beam Loading



Bending Moment Diagram

By inspection we can readily see that the wing beams are not simple supported, but have a bending moment at the strut point and also a uniform plus axial loading; therefore, to determine the maximum bending moment in the bay of beam the precise moment equation will be made use of. Calling strut point No. 1 and the point where wing is fastened to fuselage No. 2 then the moment between these two points can be found from equations:

$$(1) * M_1 = \frac{D_2 - D_1 \cos L/j}{\sin L/j} \sin x/j + D_1 \cos x/j + wj^2$$

$$\begin{aligned} D &= M - wj^2 \\ D &= M - wj^2 \\ j &= (EI/P)^{1/2} \end{aligned}$$

$$(2) * \tan x/j = \frac{D_2 - D_1 \cos L/j}{D_1 \sin L/j}$$

$$(3) * \text{Maximum moment between points 1 and 2}$$

$$M_{1-2} = D_1 \sec x/j + wj^2$$

With these formulae in mind we will proceed to solve for the maximum moments in the following tables.

* Niles and Newell, Airplane Structures p. 201.

TABLE NO. IX

DATA FOR SOLVING EQUATIONS (1, 2, 3)

Case No. 2

Spar	Flight Condition	Total Axial Load P	I	E.I.	$J^2 = \frac{EI}{D}$	J	Span L	L/j Radians	51.3L/j Degrees	Load per in. w	M.
Front	H.I.	-1,395.74	11.336	13,603,200	9,316.34	36.5	116.0	1.302	68.37	2.80	15,266.30
	H.I. (2)	-1,681.06	11.336	13,603,200	7,938.25	39.4	116.0	1.318	75.52	2.80	13,266.30
	L.I.	-165.54	11.336	13,603,200	3,174.70	286.6	116.0	0.405	23.20	1.37	7,469.58
Rear	H.I.	-1,301.84	5.63	6,756,000	5,182.58	71.03	116.0	1.610	92.25	1.20	6,542.70
	H.I. (2)	-563.90	5.63	6,756,000	11,930.84	109.3	116.0	1.061	60.80	1.20	6,542.70
	L.I.	-2,301.98	5.63	6,756,000	2,924.99	54.1	116.0	2.144	122.85	1.69	9,214.30
	N.I.	-2,143.42	5.63	6,756,000	5,153.44	56.1	116.0	2.068	118.50	1.24	6,760.79

Case No. 1

Front	H.I.	-1,336.88	11.336	13,603,200	10,024.61	100.12	116.0	1.158	66.35	3.25	12,521.07
	H.I. (2)	-1,624.79	11.336	13,603,200	8,372.28	91.5	116.0	1.263	72.65	3.25	12,521.07
	L.I.	-140.13	11.336	13,603,200	97,075.00	311.5	116.0	0.372	21.32	1.58	6,257.32
Rear	H.I.		5.63				116.0			1.40	5,544.46
	H.I. (2)	-626.35	5.63	6,756,000	10,786.30	108.9	116.0	1.118	64.06	1.40	5,544.46
	L.I.	-2141.84	5.63	6,756,000	3,154.30	56.2	116.0	2.064	113.26	1.96	7,762.25
	N.I.		5.63							1.58	6,257.32

E (Spruce) = 1.2×10^6

Axial Load (P) - See Tables VII & VIII Sect. E - Art. (c)

Load per in. = w - See Table II Sect. C - Art. (h)

M₁ - See Sect. C - Art. (i & j)

TABLE NO. X
MAXIMUM MOMENTS IN BAY
Case No. 1

	Front Spar			Rear Spar		
	H.I.	H.I.(2)	L.I.	H.I.	H.I.(2)	N.F.
Load = (w)	3.25	3.25	1.53	1.40	1.40	1.96
$J^2 = EI/P$	10,024.61	9,374.23	97,075.00		10,726.30	3,154.30
wj^2	32,573.33	27,106.91	153,378.50		15,100.32	6,182.43
M_1	12,871.07	12,871.07	6,257.32		5,514.46	7,762.25
$D_1 = M_1 - wj^2$	-19,708.91	-14,333.84	-147,121.18		-9,556.36	1,579.32
M_2	0	0	0		0	0
$D_2 = M_2 - wj^2$	-32,573.33	-27,209.91	-153,378.50		-15,100.32	-6,182.43
L/j (degrees)	66.35	72.66	21.32		64.06	118.26
$\cos L/j$	0.40115	0.23804	0.2350		0.43759	-0.47350
$D_1 \cos L/j$	-7,906.23	-4,273.65	-137,553.30		-4,161.76	-743.17
$D_2 - D_1 \cos L/j$	-24,673.75	-22,936.26	-13,820.20		-10,919.06	-5,434.26
$\sin L/j$	0.91601	0.95435	0.36360		0.89918	0.88075
$D_1 \sin L/j$	-18,053.56	-13,697.14	-55,493.26		-8,592.89	1,391.43
$\tan x/j =$						
$D_1 \sin L/j$	1.8046	1.5760	.2957		1.2707	-3.9052
$\tan^2 x/j + 1$	4.2566	3.8090	1.0974		2.61478	16.250587
$\sec x/j =$						
$(\tan^2 x/j + 1)^{1/2}$	2.063	1.951	1.004		1.617	4.031
$D_1 \sec x/j$	-40,659.48	-27,965.09	-147,737.66		-15,452.63	6,368.25
$M_1 - 2 =$						
$D_1 \sec x/j + wj^2$	-8,079.50	-755.17	5,670.84		-352.81	12,550.68
x (in.)	100.59	94.43			93.84	102.01

TABLE NO. XI

MAXIMUM MOMENTS IN BAY

Case No. 2

	Front Spar			Rear Spar		
	H.I.	L.I.	H.I.	H.I.	L.I.	N.D.
Load - w	2.80	1.37	1.30	1.20	1.59	1.24
$j = EI/P$	9,316.34	82,174.70	5,189.58	11,990.84	2,934.29	3,153.44
wj^2	26,085.75	112,579.34	6,327.50	-14,396.08	4,960.13	3,910.26
M_1	15,266.30	7,469.58	6,541.70	6,541.70	9,214.30	6,760.79
$D_1 = M_1 - wj^2$	-10,819.45	-105,109.76	315.20	-7,847.39	4,254.17	7,250.53
M_2	0	0	0	0	0	0
$D_2 = M_2 - wj^2$	-26,085.75	-112,579.34	-6,327.50	-14,396.08	-4,960.13	-3,910.26
L/j (degrees)	68.87	23.20	92.55	60.3	122.85	113.50
$\cos L/j$	0.36050	0.91882	-0.03955	0.42786	0.54244	-0.47716
$D_1 \cos L/j$	-3,900.41	-96,576.35	-12.466	-4,875.60	-2,507.70	-1,274.64
$D_2 - D_1 \cos L/j$	-22,185.34	-16,002.39	-6,215.05	-2,507.48	-7,652.43	-2,675.62
$\sin L/j$	0.93276	0.39458	0.99922	0.97292	0.34009	0.87282
$D_1 \sin L/j$	-10,091.95	-41,079.63	314.35	-6,846.54	3,573.99	2,505.10
$\tan x/j = \frac{D_2 - D_1 \cos L/j}{D_1 \sin L/j}$	2.19832	0.38954	-19.7333	1.3886	-0.7421	-1.0521
$\tan^2 x/j + 1$	5.832611	1.15174	390.4031	2.2282	1.5507	2.1069
$\sec x/j = (\tan x/j + 1)^{1/2}$	2.415	1.073	19.76	1.711	1.246	1.452
$D_1 \sec x/j$	-26,128.97	-112,782.77	6,228.35	-13,420.02	5,297.62	4,139.97
$M_1 - 2 =$	43.22	-203.43	12,555.85	966.06	10,257.75	8,049.23
$D_1 \sec x/j + wj$			116.6		274.4	
x						

TABLE NO. XII

COMPUTATIONS OF STRESSES

Case No. 2 - Loading

Spar	Loading Condition	Moment	Magnitude of Moment	Y/I	Bending Stress $f_b = \frac{MY}{I}$	Axial Load	Area	Compression $f_c = P/A$	Total Stress $f_b + f_c$	Allowable Bending Stress	Allowable Combined Stress
Front At	H.I.	M ₁₋₂	-8,079.50	.265	-2,141.07	-1,128.94	1.69	-668.01	-2,809.08	6,100	-5,150
Fuselage	"	"	-8,079.50	.222	-3,639.41	-1,128.94	4.5	-250.86	-3,890.27	9,400	-9,050
	H.I.(2)	M ₁₋₂	-755.17	.265	-200.12	-1,663.16	1.69	-987.07	-1,197.19	6,100	-3,190
Fuselage At	"	"		.222	-167.64	-1,663.16	4.5	-370.70	-538.34	9,400	-6,400
Rear	H.I.(2)	M ₁₋₂	-352.81	.377	-133.00	-616.14	3.19	-193.15	-326.15	9,400	-7,200

Case No. 1 - Loading

Front	H.I.	M ₁	15,286.30	.265	-4,045.57	-1,639.94	1.69	-970.38	-4,115.95	6,100	-6,100
	L.I.	M ₁	7,469.53	.265	-1,979.44	-667.31	1.69	-394.86	-2,374.30	6,100	-5,400
Rear	H.I.	M ₁	6,542.70	.377		-1,005.19	3.19			9,400	
	H.I.(2)	M ₁	6,542.70	.377		-529.32	3.19			9,400	
	L.I.	M ₁	3,214.30	.377	-3,473.79	-1,720.34	3.19	-5264.13		9,400	-7,500
	N.I.	M ₁	6,730.79	.377		-1,601.05	3.19			9,400	

The allowable stress column was determined from a set of curves prepared by the Forest Products Laboratory at Madison, Wisconsin, presented by Niles and Newell, Airplane Structures - page 249.

In the TABLES X and XI when the moments were plus this indicates that there is no inflection point. If we solve for (x), the distance from strut point to point of maximum moment, and this is found to be a value greater than (L), the length from strut point to fuselage, then this indicates that M, at the strut point is the worst moment. Thus:

In TABLE NO. XI under rear spar H.I.

$$\tan x/j = -19.733$$

$$x/j = 92.9 = 92.9/57.3 = 1.621(\text{radians})$$

$$x = 1.621 \times 72.03 = 116.6 \text{ in.}$$

which is greater than L (116.0 in.)

If we were to choose some point outboard of the strut point it can be very easily seen that the axial force as well as bending moment on spar would become less and enable one to design the spar with a smaller crosssection as the section becomes nearer to the wing tip. However, since the spars are of the same crosssection throughout in this case there is no need to check the forces outboard of the strut point.

CONCLUSION

In working through this problem in stress analysis the forces in the ribs at panel points were solved for in the solution of truss but no mention was made of as to what the allowable stresses are. The strength of the ribs in a wing structure are not computed mathematically but are tested in a mechanical jig. However, in our case the forces in ribs are not severe as can be seen in Fig. 17. The worst compressive stress being 503.19 pounds.

An inspection of the results found in this thesis reveals that none of the members of the wing structure are overstressed. On the contrary most of the members have a high factor of safety. Some of the safety factors are as high as 4 or 5. The design could perhaps be made more conservative but this would not be recommended since this glider is intended to be built by the average air-minded person who in most instances would have no means of testing the structure or material, an operation which would be necessary if one ~~was~~^{were} to adhere to the 1.5 safety factor which is used in some aircraft design. Furthermore in a glider of this nature we are not as much interested in attaining high speeds as we are in reliability under rough usage.

This thesis does not go into any great detail in the application of aerodynamic principles, but I have merely applied some of the fundamentals to one of the simpler type of aircraft design.

BIBLIOGRAPHY

- Niles and Newell, Airplane Structures, New York, John Wiley and Sons, Inc., 1929.
- Warner and Johnston, Aviation Handbook, New York and London, McGraw - Hill Book Co., Inc., 1931.
- Klemin, Alexander, Airplane Stress Analysis, New York, The Ronald Press Co., 1929.
- Warner, Edward P., Airplane Design, New York, McGraw-Hill Book Co., Inc., 1927.
- Klemins, Alexander, Simplified Aerodynamics, Chicago, The Goodheart Wilcox Co., Inc., 1930.
- Trayer, George W., Wood In Aircraft Construction, Washington D.C., The National Lumber Manufacturers Ass., 1930.
- Rathbun, John B., Aeroplane Construction and Operations, Chicago, Stanton and VanVliet Co., 1918.

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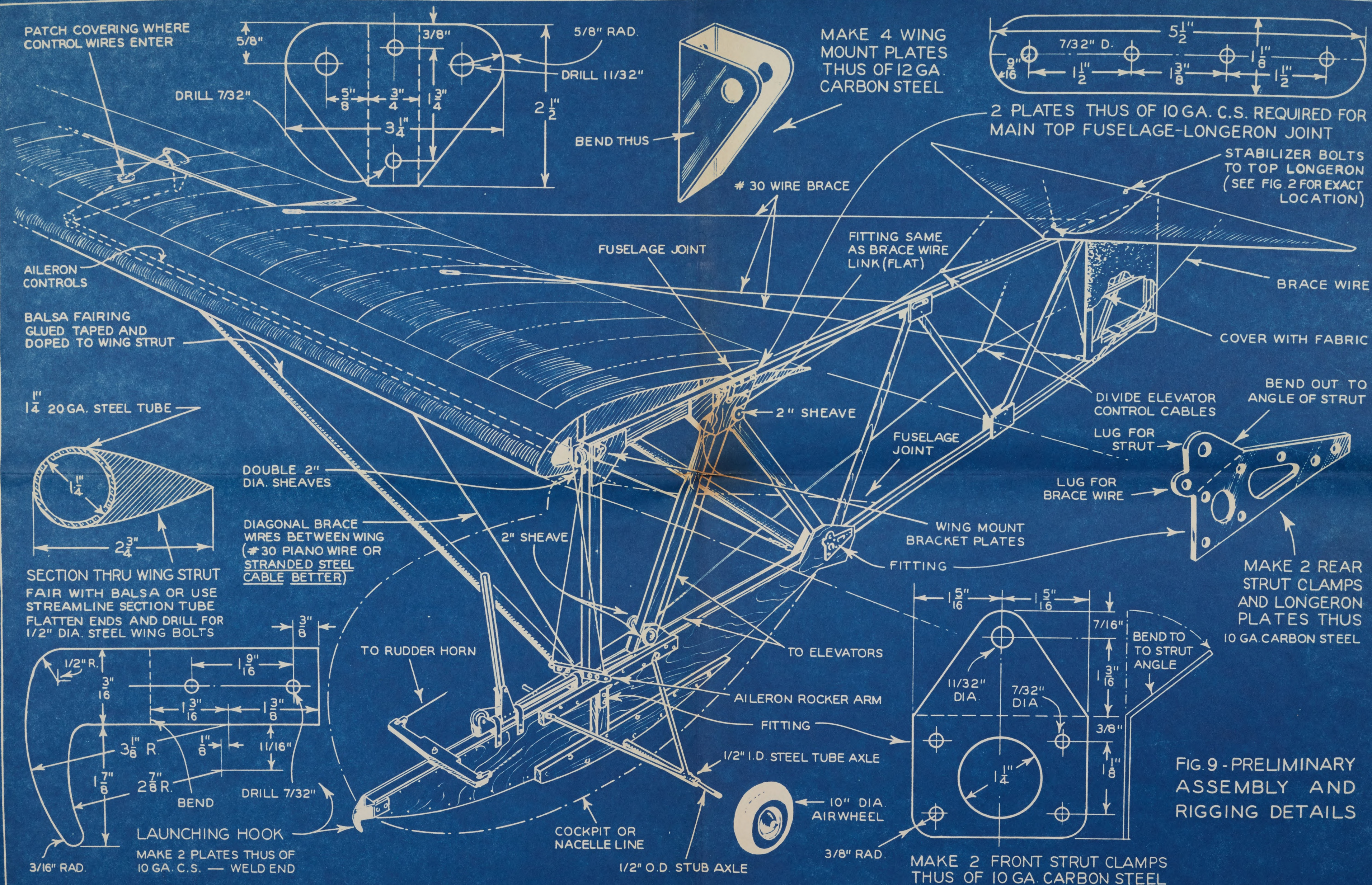
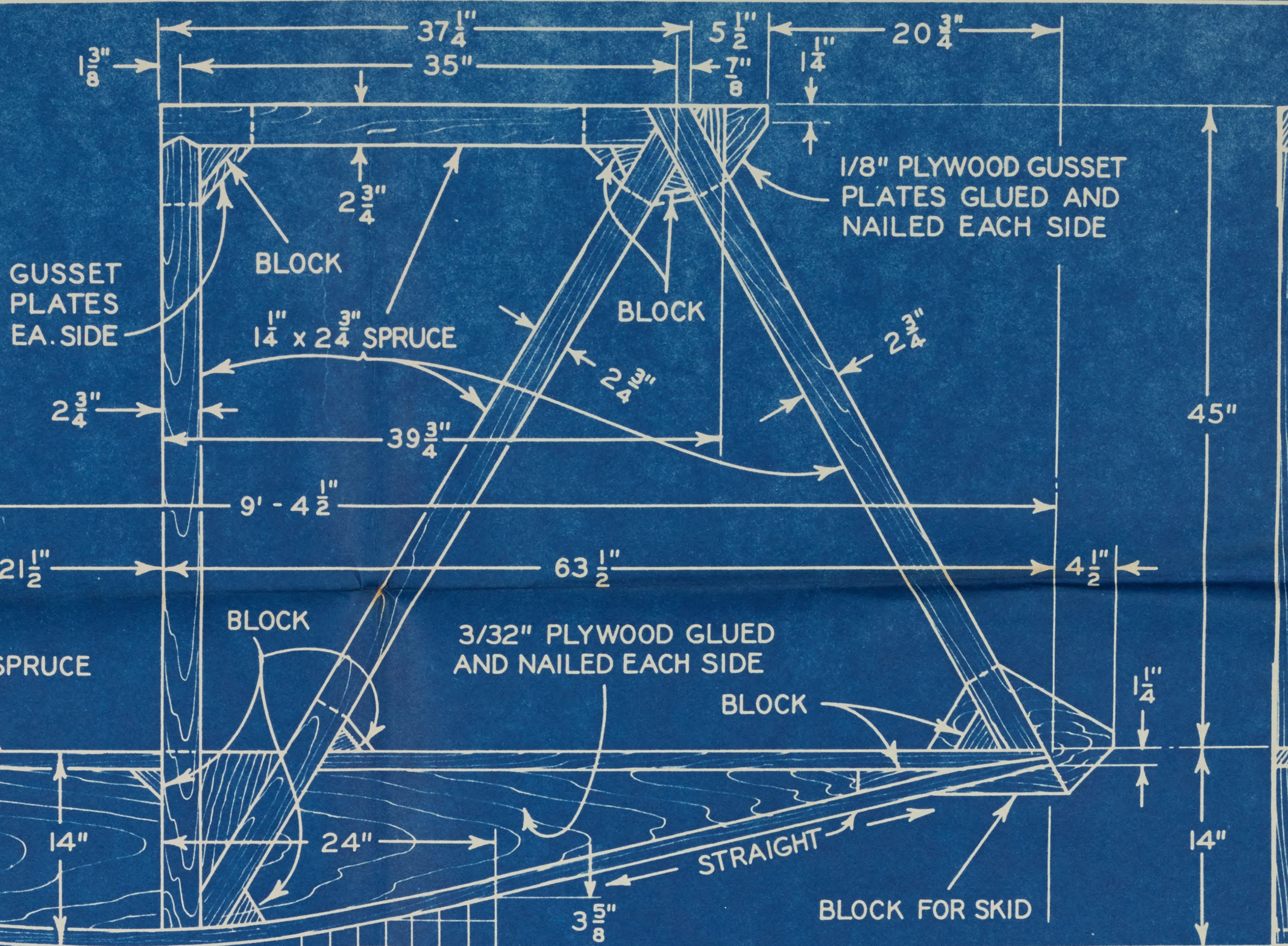
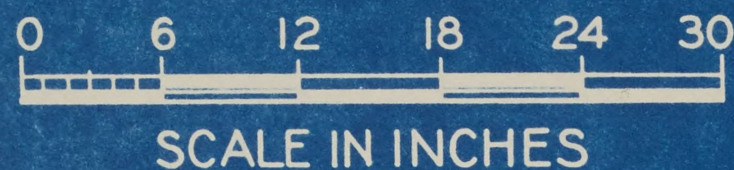
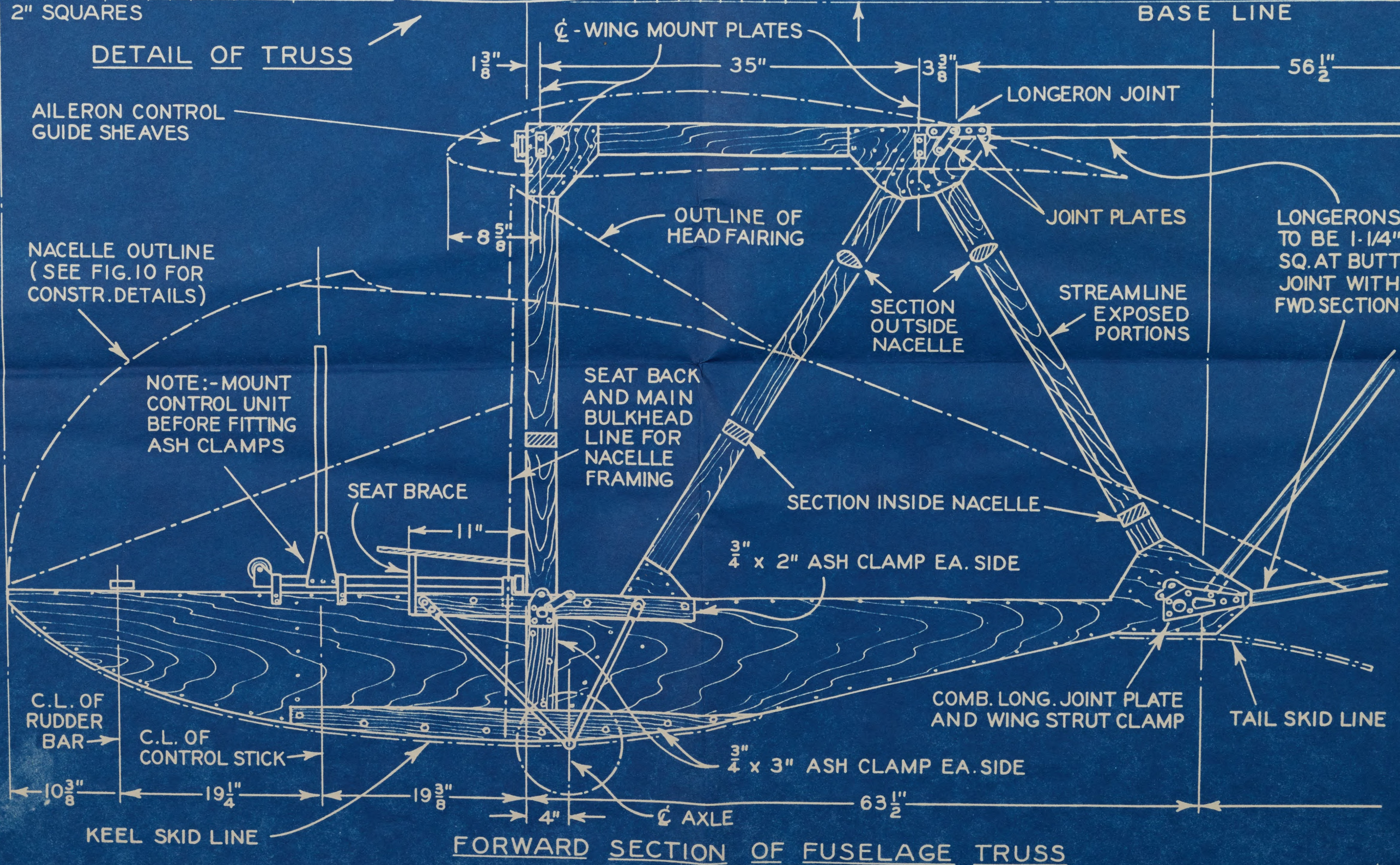


FIG. 2 — CONSTRUCTION
PROFILE OF FUSELAGE
& RUDDER — PLAN VIEW
OF STABILIZER & FLAPS

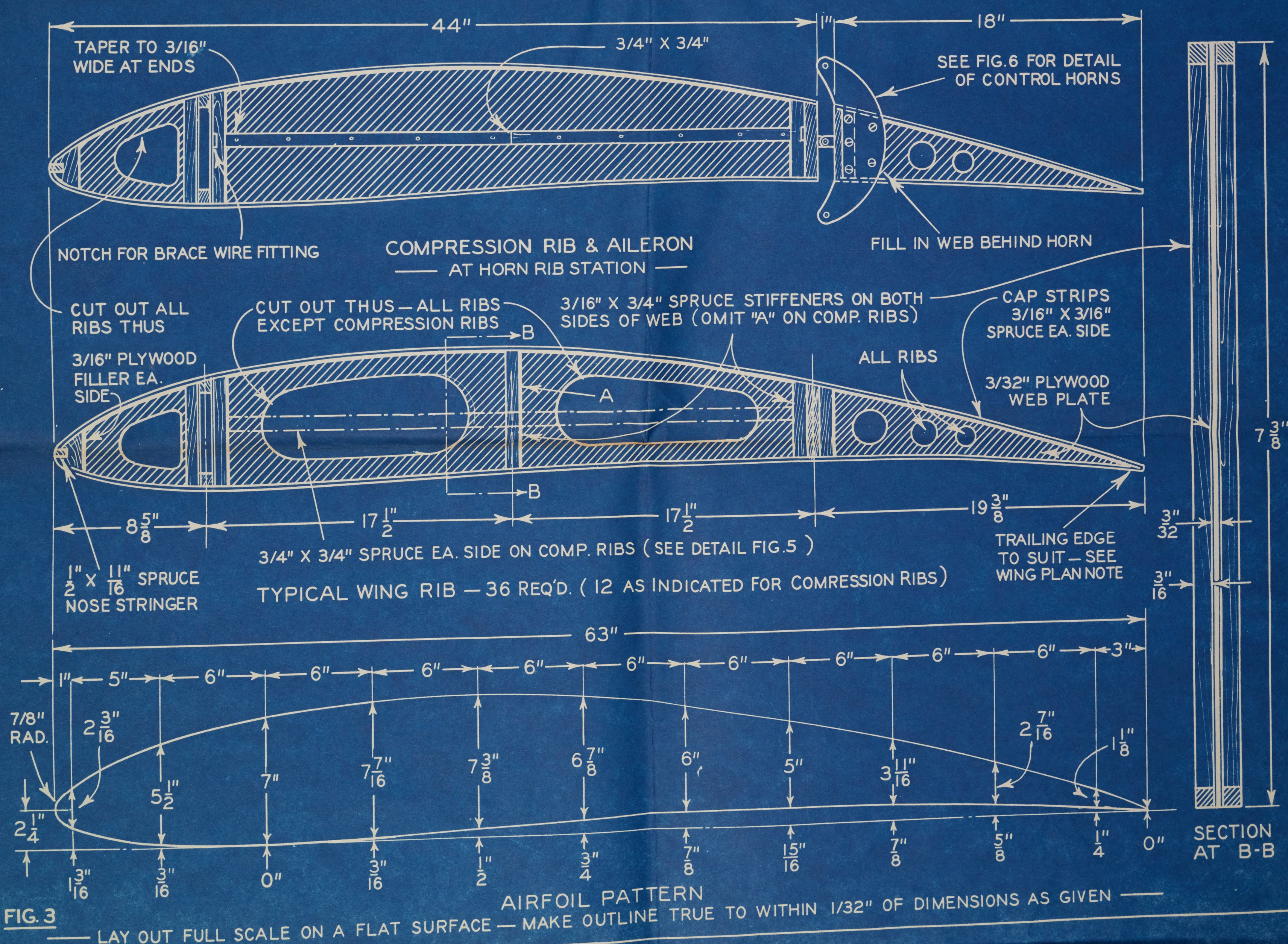
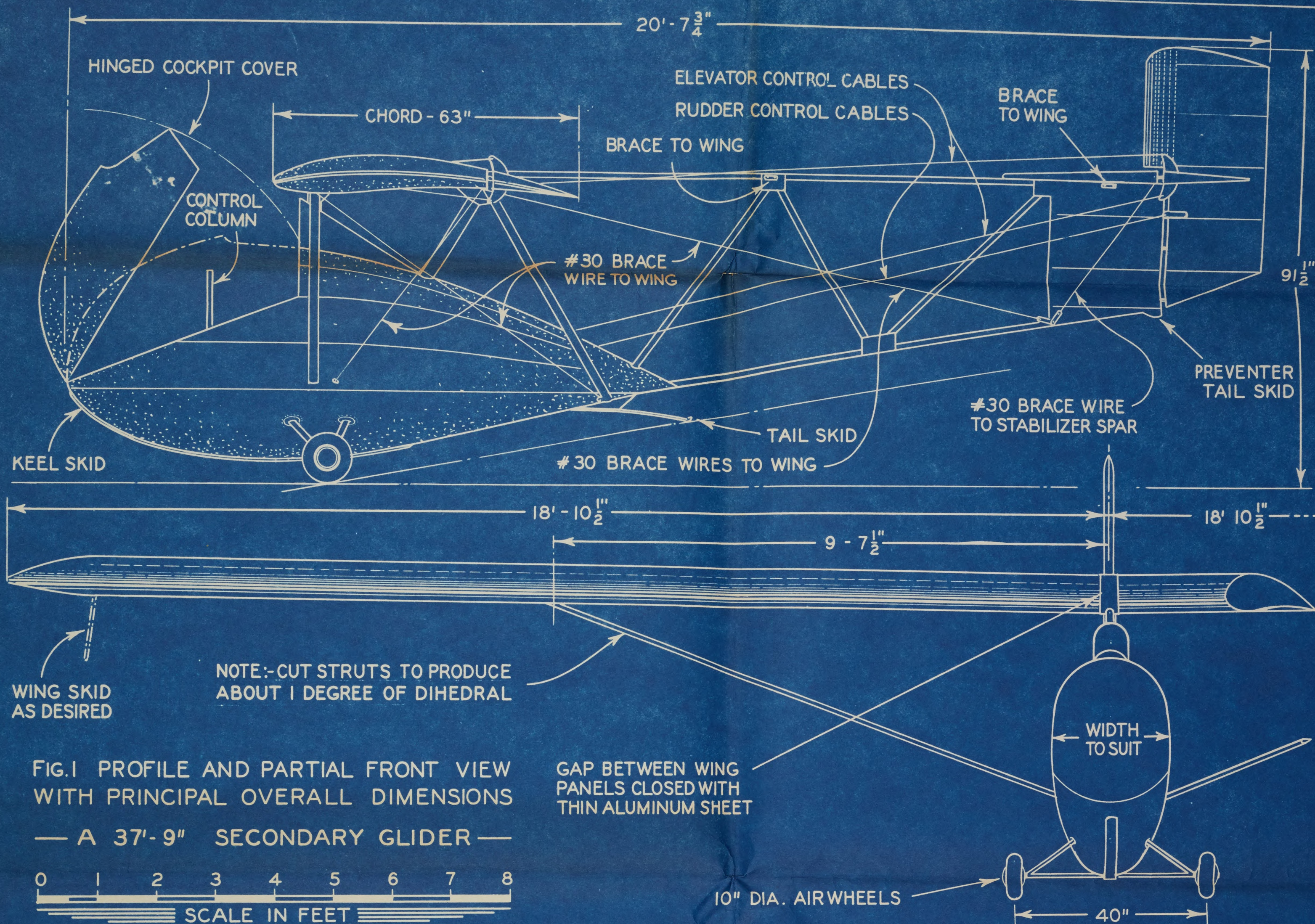


DETAIL OF TRUSS



COVER TIPS WITH 3/32" PLYWOOD





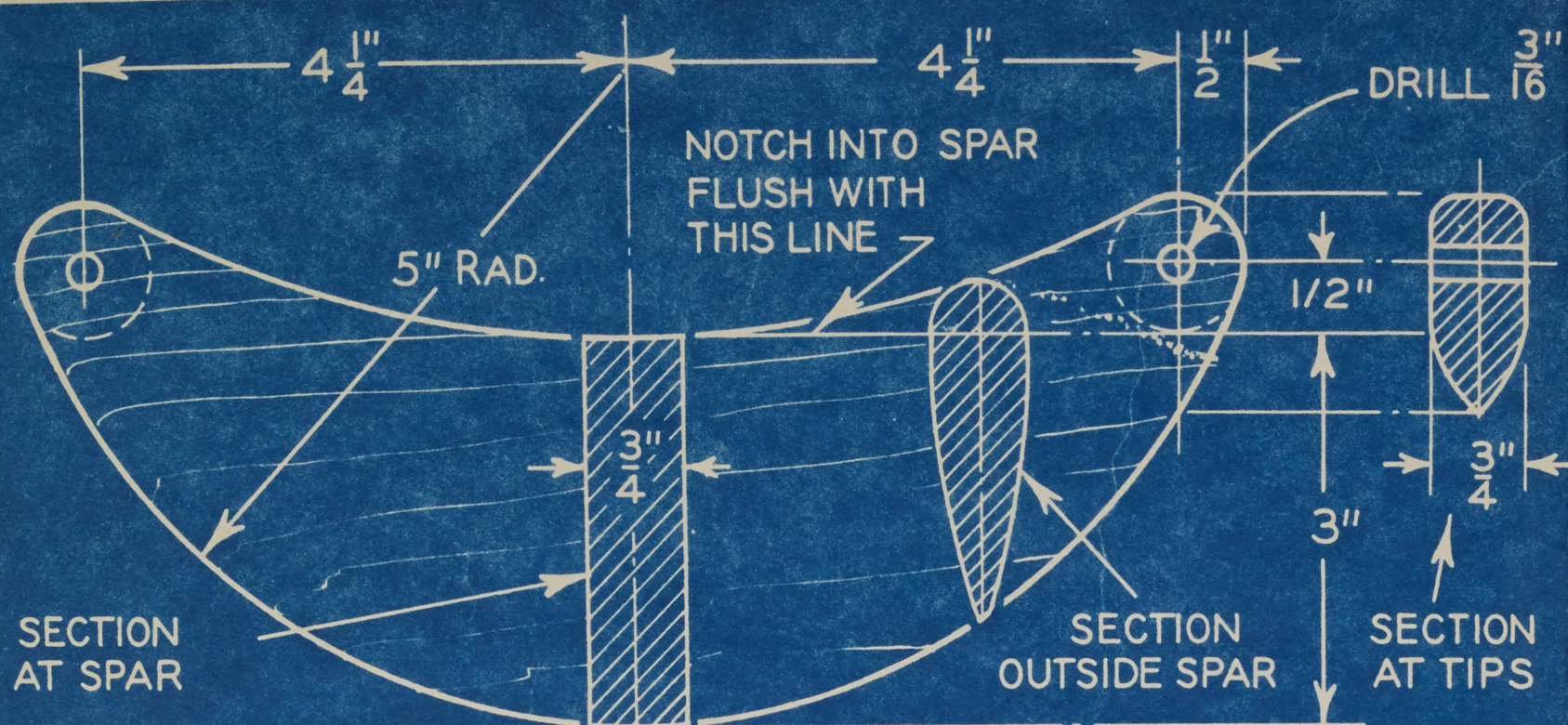


FIG. 5 - WING CONSTRUCTION DETAILS
SHOWING COMPRESSION RIB AT AILERON HORN STATION
AND LAYOUTS FOR AILERON CONTROL & DRAG WIRE FITTINGS

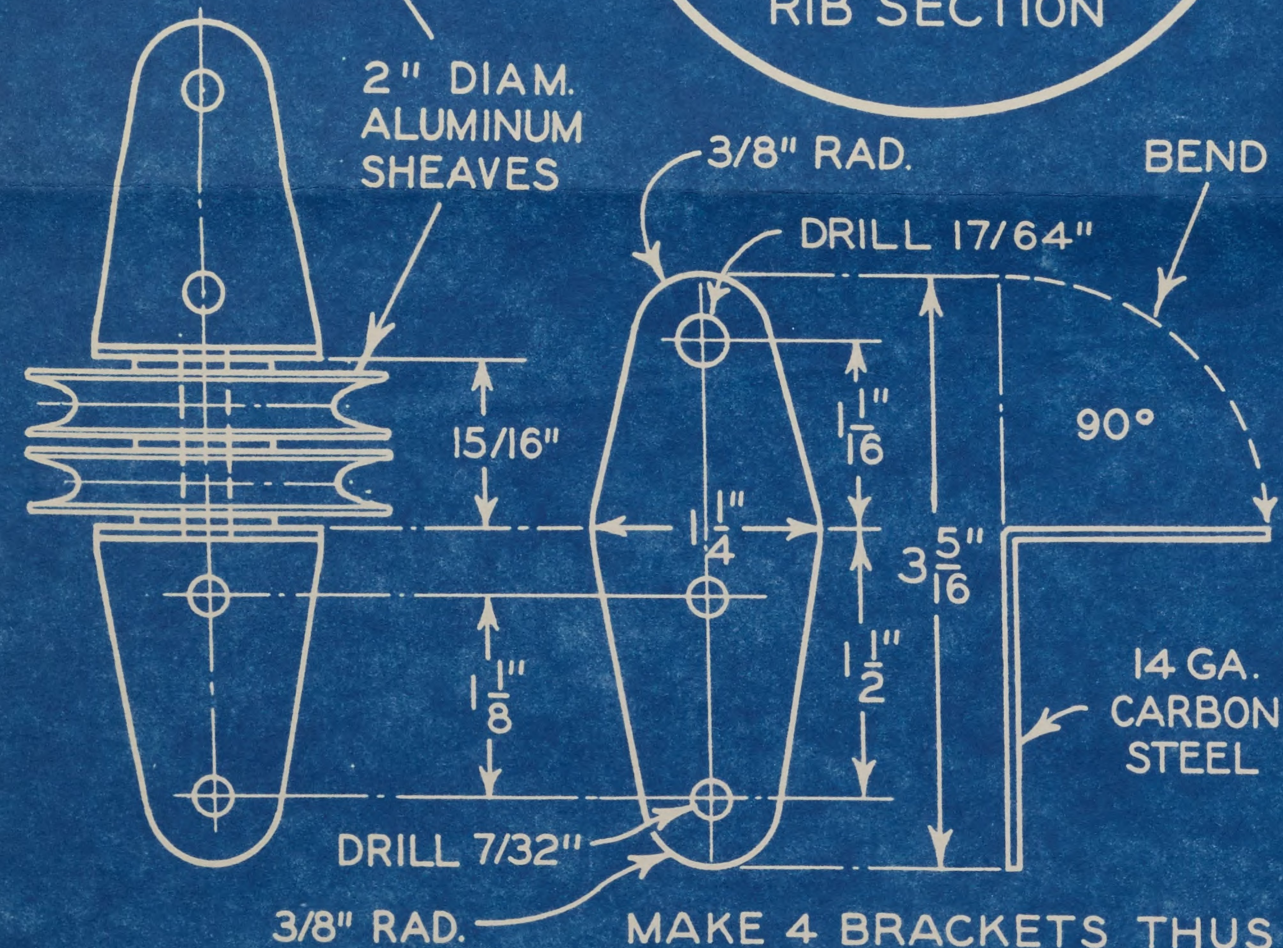
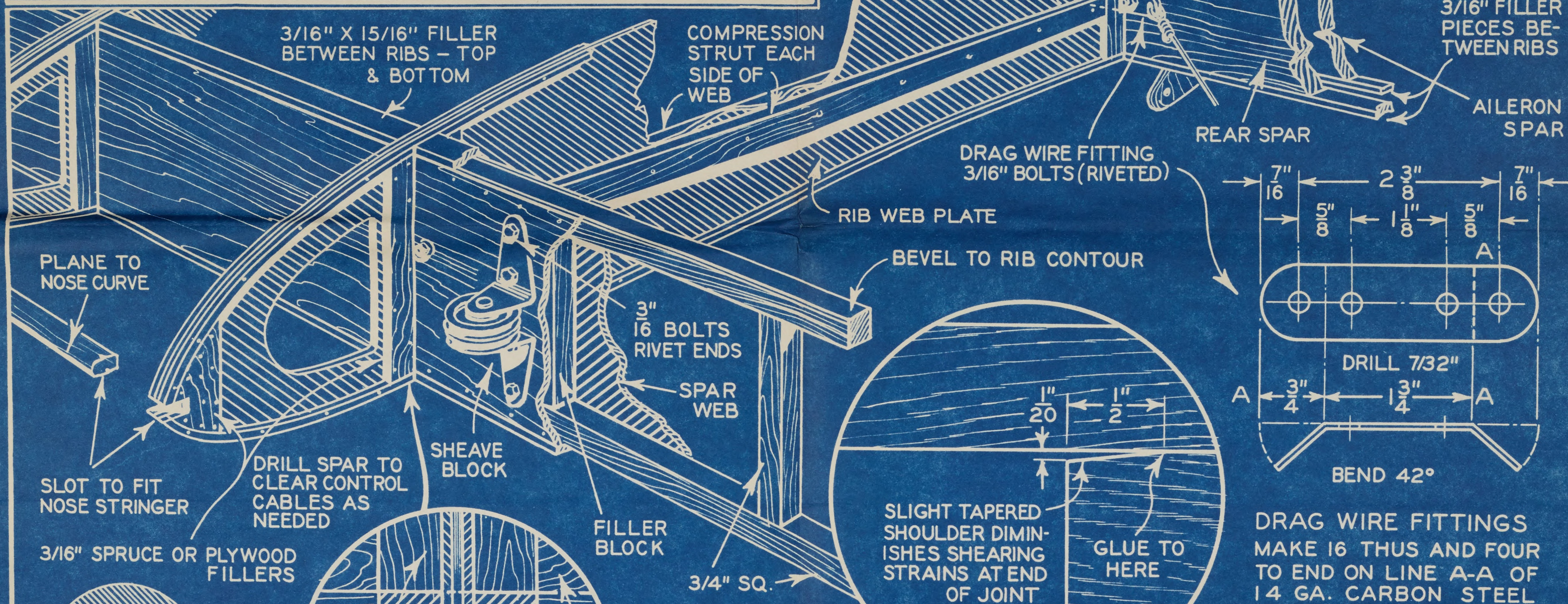
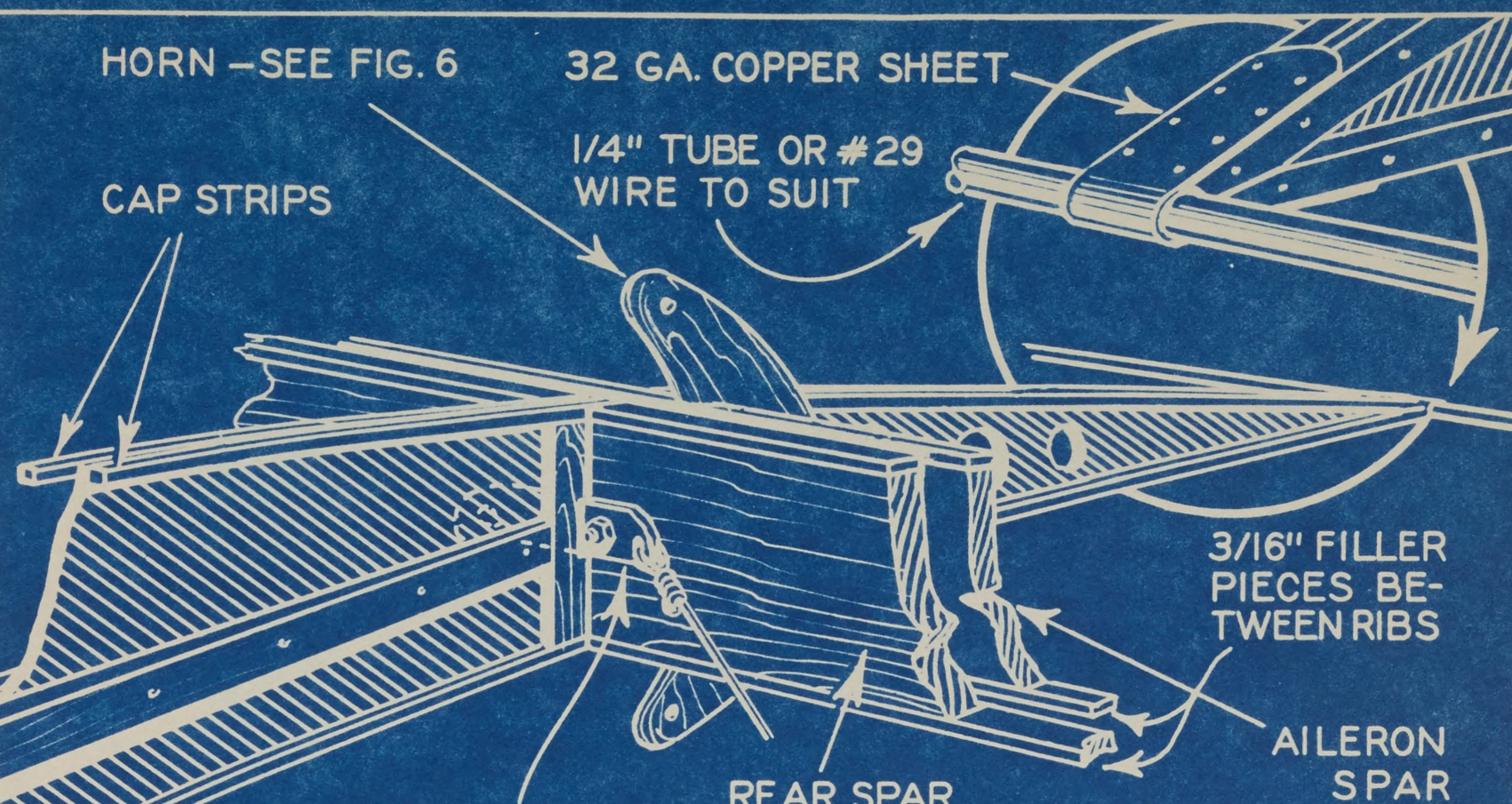
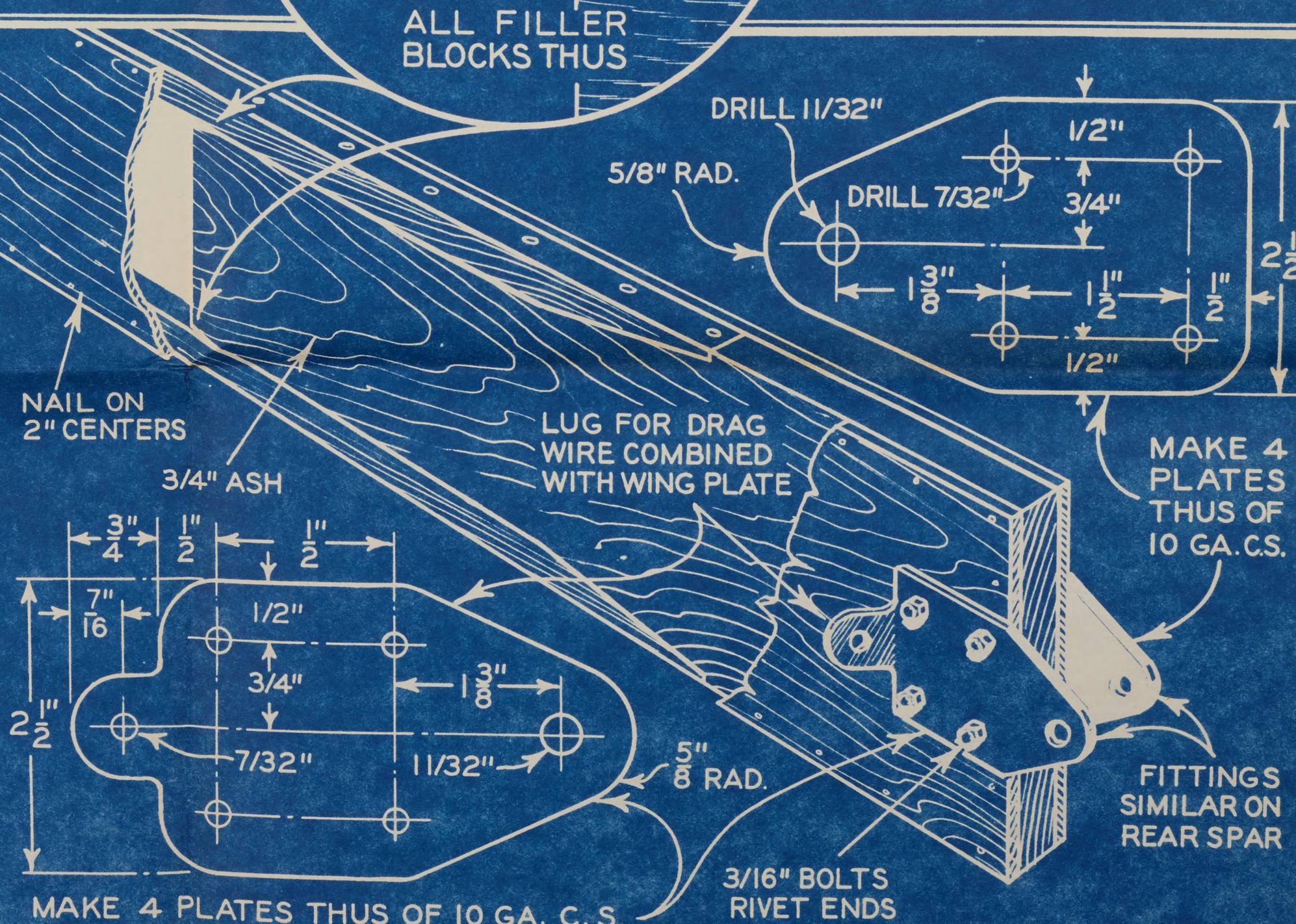
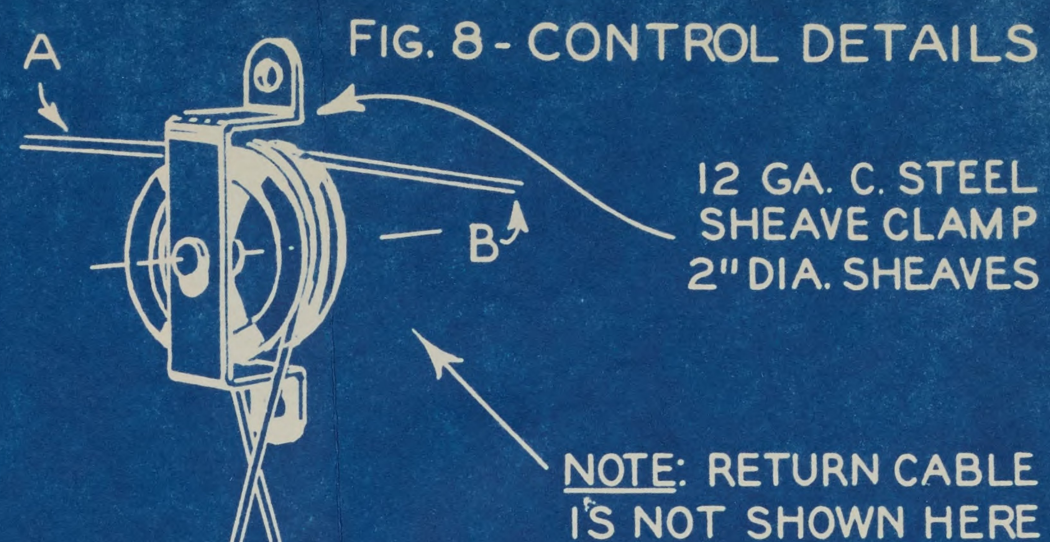


FIG. 7 - DETAIL OF FRONT SPAR SHOWING WING MOUNT FITTINGS



[illegible]

CONTROL STICK — 1 $\frac{3}{16}$ " O.D. STEEL TUBING
FLATTEN END AND DRILL AS REQUIRED —

SPACER TUBES EA. SIDE TO RUNNING FIT



ELEVATOR SHEAVE
YOKE PATTERN
MAKE 1 OF 12 GA.
— C.R. STEEL —

2" DIA. —
SHEAVE

CUT SLOT
IN TUBE
TO CLEAR
SHEAVE

RIVET YOKE TO TUBE

TORQUE TUBE
25 $\frac{5}{8}$ " OVERALL

**ROCKER ARM
RIVET OR BOLT
THRU TUBE**

BOLT

3"
1 1/16 O.D. STEEL
TUBING

BEARING STRAPS

BEARING BLOCKS (3) OF OILED MAPLE

CONTROL STICK YOKE—RIVET
OR BOLT TO TORQUE TUBE ➔

MAKE 3 OF
10 GA. C.R.S.

FWD.EDGE 

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