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TECHNICAL REPORT

NO. TR-AC-17

(RESTRICTED)

WING - STRESS ANALYSIS
MIG-15 - SERIAL NO. 120147

PROJECT NO. 10115

2 MARCH 1953



AIR TECHNICAL INTELLIGENCE CENTER

WRIGHT-PATTERSON AIR FORCE BASE
DAYTON, OHIO

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AUTH: CG, *July H*
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DATE: 2 March 1953

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SUMMARY

Purpose

The purpose of this stress analysis is to determine the maximum probable load factor for which the wing of the MIG-15 Airplane was designed.

Factual Data

This report was prepared by Cornell Aeronautical Laboratory, Inc., Buffalo, N.Y., under Contract No. AF 33(038)-16993. Because of manpower limitations, the work was not refined to the extent to which it was originally planned. This applies particularly to final selection of the proper amount of effective skin in the section properties.

The work is divided into three sections: the determination of section properties, calculation of unit bending moments, and the computing of permissible design load factors, as indicated by the strength of separate elements. The procedure as outlined in Reference 3 is followed essentially, and is described in detail below.

Discussion

In order to determine the bending strength of a panel, sections at three locations are analyzed. The section at the fuselage center line is a build up I-beam, whereas those at stations 117.5 and 179.4, approximately normal to front and rear spars, consist of conventional built-up beams, angles and skin. Section properties are determined for normal flight, using allowable column buckling stresses and effective skin, according to the method indicated in reference 1. From these allowable and section moduli, allowable bending moments are calculated. The amount of moment relief afforded by the ring bulkhead to Station 0 is assumed to be negligible.

For a 1-g condition at a gross weight of 10,200 pounds airload normal moments are then found, assuming a constant airload distribution over the panel. Drag load moments and the effects of angle of attack are ignored. Similarly dead-weight wing moments are computed. At the time this work was done, dead weight distribution data were unavailable, therefore, a distribution for another airplane was adapted. Finally, at contact points of the sections being investigated, "allowable" bending moments already determined are divided by net moments representing a 1-g loading, thus giving a permissible load factor.

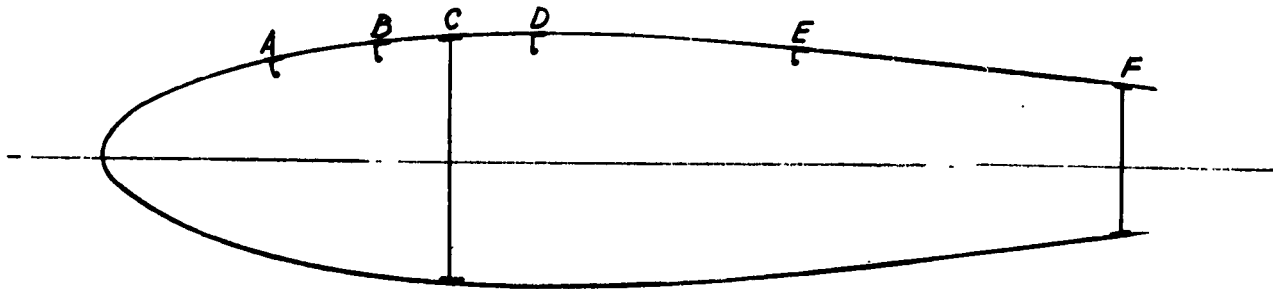
The next step would be to increase the amount of effective skin when the crippling stress has not been attained, and to decrease the effective area of components like the rear spar at Sta. 179.4, which are much less effective than the first determination of section properties indicate. This step was not accomplished.

In the determination of allowable stresses, the material of the skin and stringers was assumed to have the same properties as 24ST aluminum alloy, an approximation substantiated by reference 4. The ultimate load factors, as computed from allowable stresses of critical items at the selected wing stations (Ref. P. 40) are recorded and the locations identified below.

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ULTIMATE LOAD FACTORS

Item	Sta. 117.5	Sta. 179.4	Sta. 0
A	12.9	-	-
B	12.6	17.3	-
C	13.0	15.2	-
D	-	14.6	-
E	14.4	13.7	-
F	24.6	9.15*	-
Main Beam	-	-	10.75

*Effective area for this item should be reduced.

Conclusions

It appears that the ultimate load factor for the wing would be approximately 12, except for the fact that the factor for the carry-through steel beam is only 10.75.

However, it may be pointed out that this discrepancy does not exist if limit load factors were the criterion of strength. Using the data from reference 4, for the steel beam,

$$N \text{ (Limit)} = \frac{160000}{180000} \times 10.75 = 9.6$$

and for the panel

$$N \text{ (Limit)} = \frac{49000}{63000} \times 12 = 9.4$$

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SECTION I

WING SECTION PROPERTIES

SYMBOLS

A	-	Area of cross section, square inches
b	-	Width of sections; subscript "bending"
C	-	Fixity sufficient for columns
d	-	Distances as noted
E	-	Modulus of Elasticity in tension
F _c	-	Allowable compressive stress of column
F _{cr}	-	Allowable critical buckling stress
F _{cg}	-	Crippling stress for complete section
H.T.	-	Heat Treat
I _{xx}	-	Moment of Inertia about xx Axis
I _{xp}	-	Moment of Inertia about x centroidal principal axis
I _{xy}	-	Product moment of inertia about centroidal axes
I _{yy}	-	Moment of Inertia about yy axis
I _{yp}	-	Moment of Inertia about y centroidal principal axis
L	-	Length of column
M _A	-	Bending Moment due to Air Load
M _t	-	Total Bending Moment (1G condition)
M _w	-	Bending Moment due to dead weight
N	-	Load Factor
q	-	Dynamic Pressure (psi)
t	-	Thickness
w	-	Sheet Effective Width
W	-	Weight
x'	-	Chord Reference axis
x	-	Chord Centroidal axis
x _p	-	Perpendicular distance from x principal axis to given fiber
y'	-	Wing vertical reference axis
y	-	Wing vertical centroidal axis
y _p	-	Perpendicular distance from y principal axis to given fiber
y	-	Distance from neutral axis to given fiber
φ	-	Angle between centroidal x and y axes and centroidal principal x and y axes
ρ	-	Radius of Gyration
Σ	-	Summation

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MAIN BEAM CROSS SECTION - FUSELAGE CENTERLINE

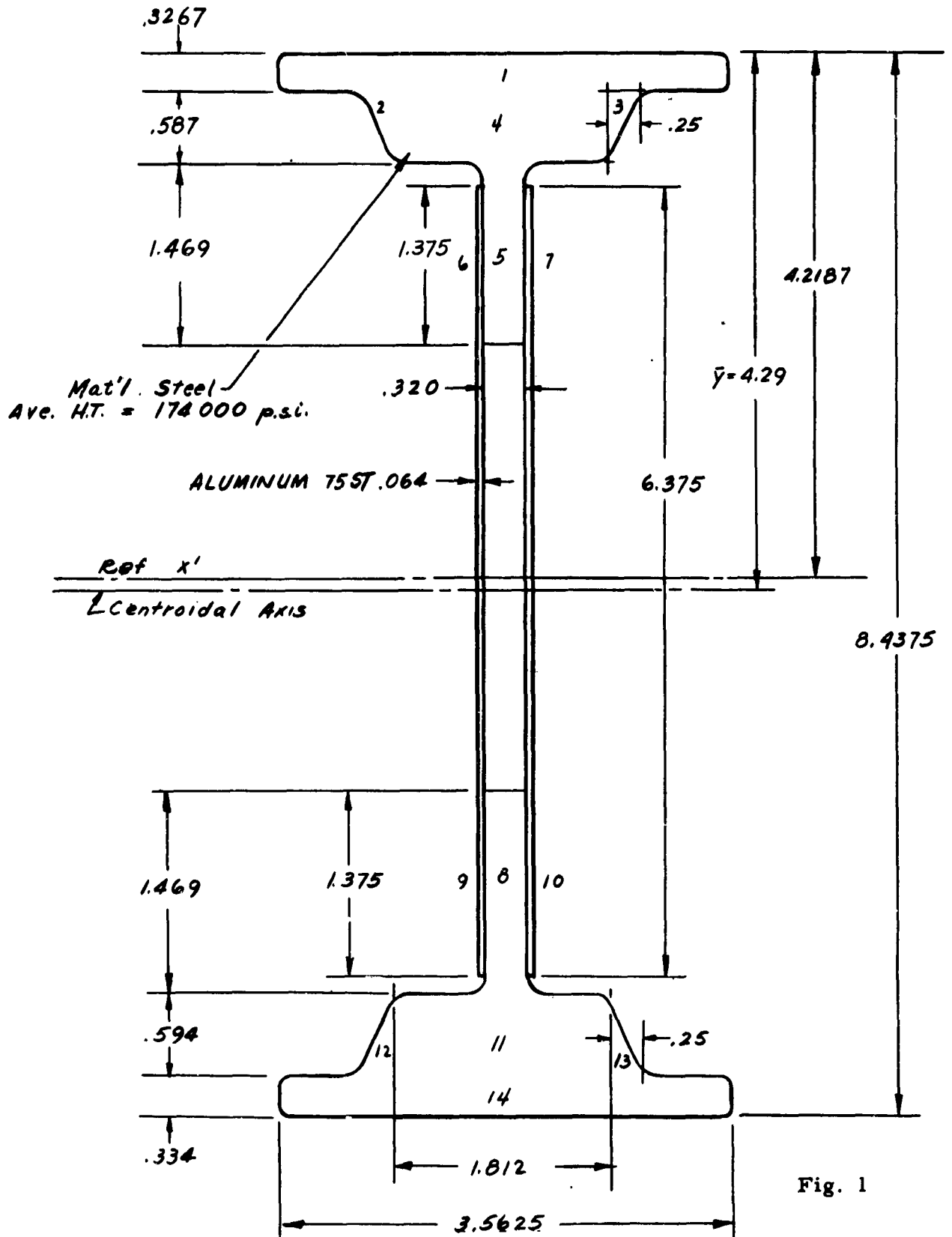


Fig. 1

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Section Properties - Fuselage Centerline

Main Beam (Ref. P. 5)

(1) Item	(2) A	(3) y'	(4) Ay' (2)x(3)	(5) Ay' ² (3)x(4)	(6) I _{ox}
1 3.52 x .327	1.164	4.05	4.72	19.10	.010
2 .25 x .587/2	.073	3.86	.28	1.09	-
3 .25 x .587/2	.073	3.86	.28	1.09	-
4 .587 x 1.812	1.062	3.31	3.50	11.53	.030
5 .320 x 1.47	.470	2.57	1.21	3.10	.085
*6 1.375 x .064/2.8	.031	2.52	.07	.20	-
7 1.375 x .064/2.8	.031	2.52	.07	.20	-
8 1.47 x .32	.470	-2.56	-1.20	3.06	.085
*9 1.375 x .064/2.8	.031	-2.51	-.07	.20	-
*10 1.375 x .064/2.8	.031	-2.51	-.07	.20	-
11 1.812 x .594	1.075	-3.59	-3.86	13.80	.032
12 .594 x .25/2	.074	-3.69	-.27	1.01	-
*13 1.375 x .064/2.8	.031	-3.69	-.11	.42	-
14 3.56 x .334	1.190	-4.05	-4.82	19.50	.013
Σ	5.806		-.279	74.50	.255

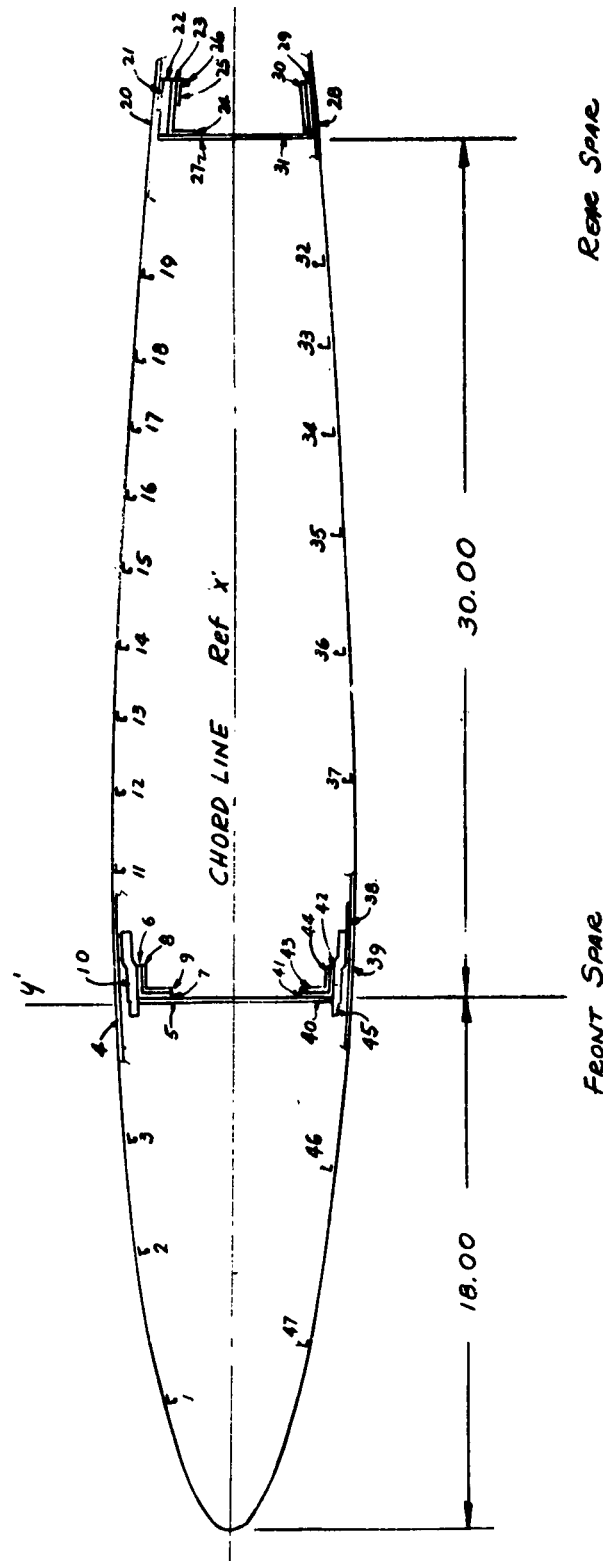
*Adjusted for steel by modulus ratio.

$$\bar{y} = \frac{\sum Ay'}{\sum A} = \frac{-.279}{5.806} = -.048 \text{ in.}$$

$$\begin{aligned} I_{xx} &= \sum Ay'^2 + \sum I_{ox} - \sum Ay^2 \\ &= 74.5 + .255 - 5.806 \times (-.048)^2 \\ &= 74.755 - .013 \\ &= 74.74 \text{ in.}^4 \end{aligned}$$

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SECTION - WING STA 117.5 (Ref. 2)

Fig. 2

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SECTION PROPERTIES - WING STATION 117.5

(Ref. P. 7)

1 Item	2 A	3 y'	4 Ay'	5 Ay' ²	6 x'	7 Ax'	8 Ax' ²	9 Ax'y'
1	.298	3.156	.94	2.95	-11.01	-3.28	36.1	-10.3
2	.298	3.484	1.04	3.60	- 5.84	-1.74	10.1	- 6.0
3	.298	4.281	1.28	5.48	- 2.25	- .67	1.5	- 2.8
4	.130	4.578	.59	2.74	1.78	.23	.4	1.0
5	.115	3.312	.38	1.26	-	-	-	-
6	.156	3.812	.59	2.26	.70	.10	.0	.4
7	.195	3.312	.64	2.14	.12	.02	.0	.0
8	.058	3.234	.18	.60	.23	.01	.0	.0
9	.067	3.703	.24	.91	.73	.05	.0	.1
10	.981	4.190	4.11	17.25	1.18	1.17	1.3	4.9
11	.425	4.547	1.93	8.80	3.60	1.53	5.5	6.9
12	.425	4.625	1.96	9.10	7.12	3.02	21.6	13.9
13	.425	4.656	1.98	9.20	10.17	4.33	44.1	20.2
14	.425	4.656	1.98	9.20	13.04	5.55	72.3	25.8
15	.425	4.578	1.94	8.72	15.96	6.80	109.0	31.2
16	.425	4.453	1.89	8.43	18.89	8.04	152.0	35.8
17	.425	4.281	1.82	7.83	21.76	9.22	200.0	39.5
18	.425	4.015	1.70	6.84	24.70	10.50	259.0	42.1
19	.425	3.718	1.58	5.86	27.28	11.60	316.0	43.2
20	.138	3.390	.46	1.59	31.25	4.31	134.0	14.6
21	.039	3.219	.12	.40	31.90	1.24	39.8	4.0
22	.570	3.156	1.79	5.66	31.25	17.80	554.0	56.0
23	.214	2.969	.63	1.88	31.25	6.68	208.0	19.8
24	1.000	2.500	2.50	6.25	30.07	30.07	906.0	75.2
25	.139	2.797	.39	1.08	31.78	4.42	140.4	12.4
26	.065	2.469	.16	.39	32.18	2.09	67.4	5.1
27	.113	2.703	.30	.82	30.03	3.38	101.8	9.2
28	.228	-3.094	- .70	2.18	30.78	7.03	216.2	-21.7
29	.375	-2.922	-1.09	3.18	30.78	11.55	356.0	-33.8
30	.111	-2.750	- .30	.84	30.78	3.42	105.4	- 9.4
31	.156	-1.547	- .24	.37	30.03	4.73	140.4	- 7.2
32	.390	-3.281	-1.28	4.20	26.81	10.45	280.0	-34.3
33	.393	-3.484	-1.37	4.78	23.18	9.12	212.0	-31.8
34	.396	-3.578	-1.42	5.07	19.62	7.76	152.5	-27.8
35	.419	-3.703	-1.55	5.70	15.53	6.50	101.0	-24.0
36	.401	-3.734	-1.49	5.57	11.57	4.65	53.7	-17.2
37	.417	-3.703	-1.54	5.70	7.90	3.30	26.2	-12.1
38	.168	-3.641	- .61	2.22	2.57	.43	1.1	- 1.5
39	.670	-3.672	-2.46	9.00	1.68	1.13	1.1	- 4.1
40	.184	-1.969	- .36	.71	-	-	-	-
41	.188	-2.133	- .40	.85	.12	.02	.0	- .0
42	.203	-2.633	- .53	1.40	.62	.12	.0	- .3
43	.062	-2.031	- .12	.25	.23	.01	.0	- .0
44	.058	-2.500	- .14	.36	.73	.04	.0	- .1
45	1.210	-3.106	-3.76	11.60	.89	1.08	.9	- 3.3
46	.554	-3.156	-1.75	5.60	- 6.12	-3.39	20.7	+10.7
47	.554	-2.469	-1.36	3.38	-11.93	-6.61	78.6	+16.3
Σ	15.836		+10.67	+204.26		+188.46	5126.7	+244.1

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$$\bar{y} = \frac{\sum Ay'}{\sum A} = \frac{10.67}{15.836} = .673'' \quad \begin{array}{l} \text{(Ref. I - P. A3.11)} \\ \text{(Ref. P. 8)} \end{array}$$

$$\begin{aligned} I_{xx} &= \sum Ay'^2 - \sum A (\bar{y})^2 \\ &= 204.26 - 15.836 (.673)^2 \\ &= 204.36 - 7.2 \\ &= 197.16 \text{ in.}^4 \end{aligned} \quad \begin{array}{l} \text{(Ref. I - P. A3.11)} \\ \text{(Ref. P. 8)} \end{array}$$

$$\bar{x} = \frac{\sum Ax'}{\sum A} = \frac{188.46}{15.836} = 11.9'' \quad \begin{array}{l} \text{(Ref. I - P. A3.11)} \\ \text{(Ref. P. 8)} \end{array}$$

$$\begin{aligned} I_{yy} &= \sum Ax'^2 - \sum A (\bar{x})^2 \\ &= 5126.7 - 15.836 (11.9)^2 \\ &= 5126.7 - 2250 \\ &= 2876.7 \text{ in.}^4 \end{aligned} \quad \begin{array}{l} \text{(Ref. I - P. A3.11)} \\ \text{(Ref. P. 8)} \end{array}$$

$$\begin{aligned} I_{xy} &= \sum Ax'y' - \sum A \bar{x}\bar{y} \\ &= 244.13 - (15.836 \times .673 \times 11.9) \\ &= 244.13 - 127 \\ &= 117.13 \text{ in.}^4 \end{aligned} \quad \begin{array}{l} \text{(Ref. I - P. A3.11)} \\ \text{(Ref. P. 8)} \end{array}$$

$$\tan 2\phi = \frac{2 I_{xy}}{I_y - I_x} \quad \text{(Ref. I - P. A3.11)}$$

$$\tan 2\phi = \frac{2 \times 117.13}{2876.7 - 197.16}$$

$$\tan 2\phi = \frac{234.3}{2679.5} = .0877$$

$$2\phi = 5^\circ$$

$$\phi = 2.5^\circ$$

$$I_{xp} = I_x \cos^2 \phi + I_y \sin^2 \phi - 2 I_{xy} \sin \phi \times \cos \phi \quad \begin{array}{l} \text{(Ref. I -} \\ \text{P. A3.11)} \end{array}$$

$$I_{xp} = 197.16 \times (.999)^2 + 2876.7 \times (.0436)^2 - [234.3 \times .0436 \times .999]$$

$$I_{xp} = 197.16 + 5.45 - 10.2$$

$$I_{xp} = 192.41 \text{ in.}^4$$

$$I_{yp} = I_x \sin^2 \phi + I_y \cos^2 \phi + 2 I_{xy} \sin \phi \times \cos \phi \quad \begin{array}{l} \text{(Ref. I -} \\ \text{P. A3.12)} \end{array}$$

$$I_{yp} = 197.16 \times (.0436)^2 + 2876.7 \times (.999)^2 + [234.3 \times .0436 \times .999]$$

$$I_{yp} = .374 + 2876.7 + 10.2$$

$$I_{yp} = 2887.3 \text{ in.}^4$$

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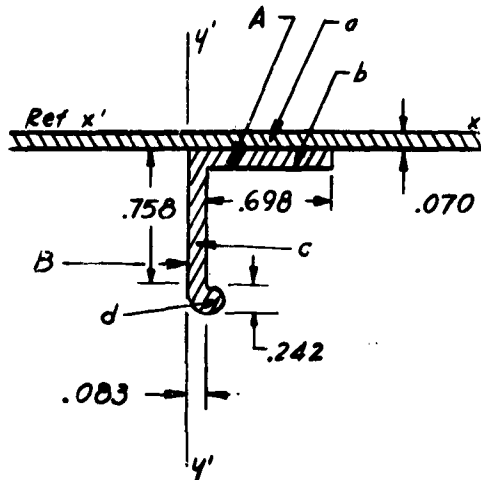
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Section Properties - Wing Station 117.5

Bulb Angle Section - Items 1, 2, 3.

(Ref. P. 7)



Buckling Stress

Element A

$$\frac{b_1}{t_1} = \frac{.698}{.083} = 8.4 \quad F_{cr1} = 38,000 \text{ psi}$$

Element B

(Ref. I - Fig. B5.13)

$$\frac{b_2}{t_2} = \frac{.917}{.083} = 11 \quad F_{cr2} = 48,750 \text{ psi (Ref. I - Fig. B5.13)}$$

$$F_{cs} = \frac{(F_{cr1} b_1 + F_{cr2} b_2)}{b_1 + b_2} \quad (\text{Ref. I - P. B5.7(6)})$$

$$F_{cs} = \frac{(38,000 \times .698) + (48,750 \times .917)}{1.615}$$

$$F_{cs} = \frac{71,100}{1.615} = 44,000 \text{ psi for Bulb Angle}$$

Effective Skin

$$w = 1.7 t \sqrt{\frac{E}{F_{cs}}} \quad (\text{Ref. I - P. B4.2(6)})$$

$$= 1.7 (.070) \sqrt{\frac{10.3 \times 10^6}{4.4 \times 10^4}} = 1.82 \text{ in.}$$

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Section Properties - Wing Station 117.5

Section Properties (Ref. P. 10)

Item	A	y'	A _{y'}	A _{y'} ²	x'	A _{x'}
a 1.8 (.070)	.127	.035	.004	.0001	.432	.055
b .698 (.083)	.058	.107	.006	.0007	.432	.025
c .758 (.083)	.063	.445	.028	.0125	.041	.003
d 3.14 (.141)	.046	.945	.044	.041	.121	.005
Σ	.294	ȳ=.28	.082	.054	ȳ=.30	.088

$$\bar{y} = \frac{\sum A_{y'}}{\sum A} = \frac{.082}{.294} = .28'' \quad (\text{Ref. I - P. A3.11})$$

$$\bar{x} = \frac{\sum A_{x'}}{\sum A} = \frac{.088}{.294} = .30''$$

$$I_{xx} = \sum A_{y'}^2 - \sum A_{\bar{y}}^2 \quad (\text{Ref. I - P. A3.11})$$

$$I_{xx} = .054 - .294 (.28)^2$$

$$= .031 \text{ in.}^4$$

$$\rho = \sqrt{\frac{I_{xx}}{A}} = \frac{.031}{.294} = .326 \text{ in.} \quad (\text{Ref. I - P. A3.5})$$

Column Allowable Buckling Stress

Using the Johnson Column Formula

$$F_c = F_{cs} - \frac{(F_{cs})^2 \times \left(\frac{L}{\rho}\right)^2}{4 C \pi^2 E} \quad (\text{Ref. I - P. B6.4(3)})$$

$$F_c = 44,000 - \frac{(44,000)^2 \left(\frac{10.75}{.326}\right)^2}{4 (2) (3.14)^2 (10.3 \times 10^6)}$$

$$= 44,000 - 2620 = 41,380 \text{ psi}$$

Revised Effective Skin

$$w = 1.7 t \sqrt{\frac{E}{F_c}} \quad (\text{Ref. I - P. B6.4})$$

$$= 1.7 (.07) \sqrt{\frac{10.3 \times 10^6}{41,380}} = 1.88 \text{ in.}$$

$$\text{Total area } A = .167 + (1.88 \times .07) = .298 \text{ in.}^2$$

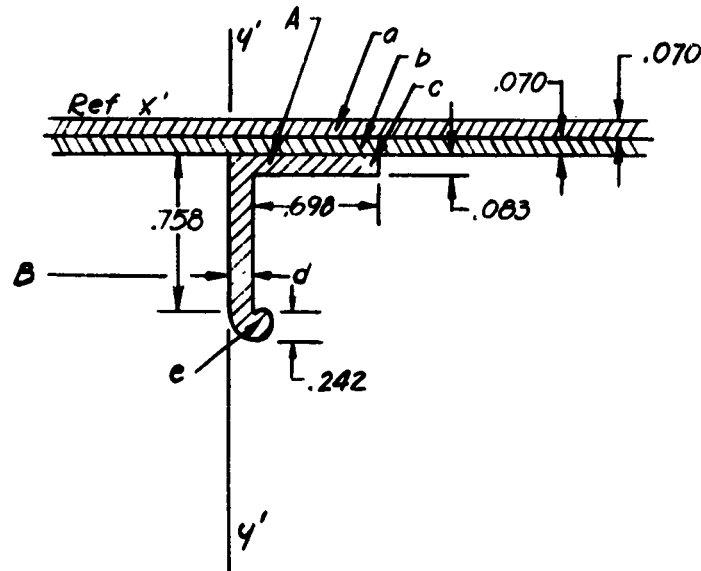
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Section Properties - Wing Station 117.5

Bulb Angle Section - Items 11, 12, 13, 14, 15, 16, 17, 18, 19. (Ref. P. 7)



Buckling Stress

Element A

$$\frac{b_1}{t_1} = \frac{.698}{.083 + .035} = 5.9$$

$$F_{cr1} = 43,000 \text{ psi}$$

Element B

(Ref. I - Fig. B5.13)

$$\frac{b_2}{t_2} = \frac{.917}{.083} = 11$$

$$F_{cr2} = 48,750 \text{ psi}$$

(Ref. I - Fig. B5.13)

$$F_{cs} = \frac{(43,000 \times .698 \times .118) + (48,750 \times .083 \times .917)}{(.698 \times .118) + (.917 \times .083)}$$

$$F_{cs} = \frac{3540 + 3710}{.0824 + .076} = \frac{7250}{.1584} = 45,900 \text{ psi} \quad (\text{Ref. I - P. B5.7(5)})$$

Effective Skin

$$W = 1.7 \times .070 \sqrt{\frac{10.3 \times 10^6}{4.59 \times 10^4}}$$

(Ref. I - P. B4.2(6))

$$W = 1.78 \text{ in.}$$

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Section Properties - Wing Station 117.5

Section Properties (Ref. P. 12)

Item	A	y'	A _{y'}	A _{y'} ²	x'	A _{x'}
a 1.78 x .070	.125	.035	.0044	.00015	.432	.0540
b 1.78 x .070	.125	.105	.0131	.00138	.432	.0540
c .698 x .083	.058	.177	.0010	.00182	.432	.0250
d .758 x .083	.063	.514	.0324	.01660	.041	.0026
e 3.14 x (.121) ²	.046	1.015	.0467	.04740	.121	.0055
Σ	.417	$\bar{y} = .234$	.0976	.06735	$\bar{x} = .339$	.1411

$$\bar{y} = \frac{.0976}{.417} = .234 \text{ in.} \quad (\text{Ref. I - P. A3.11})$$

$$\bar{x} = \frac{.1411}{.417} = .339 \text{ in.}$$

$$I_{xx} = \Sigma A y'^2 - \Sigma A (\bar{y})^2 \quad (\text{Ref. I - P. A3.11})$$

$$I_{xx} = .067 - (.417 \times (.234)^2)$$

$$I_{xx} = .067 - .023$$

$$I_{xx} = .044 \text{ in.}^4$$

$$\rho = \sqrt{\frac{.044}{.417}} = .324 \text{ in.} \quad (\text{Ref. I - P. A3.5})$$

Column Allowable Buckling Stress

(Ref. I - P. B6.4(3))

$$F_c = 45,900 - \frac{(45,900)^2 \times \left(\frac{10.75}{.324}\right)^2}{4 \times 2 \times (3.14)^2 \times 10.3 \times 10^6} \quad (\text{Ref. P. 12})$$

$$F_c = 45,900 - \frac{23.2 \times 10^{11}}{8.1 \times 10^8}$$

$$F_c = 45,900 - 2860 = 43,040 \text{ psi}$$

Revised Effective Skin

$$w = 1.7 \times .070 \sqrt{\frac{10.3 \times 10^6}{4.304 \times 10^4}} \quad (\text{Ref. I - P. B6.4})$$

$$w = 1.84 \text{ in.}$$

Area of Skin and Doubler

$$A = 1.84 \times .140 = .258 \text{ in.}^2$$

$$\text{Total Area} = .167 + .258 = .425 \text{ in.}^2$$

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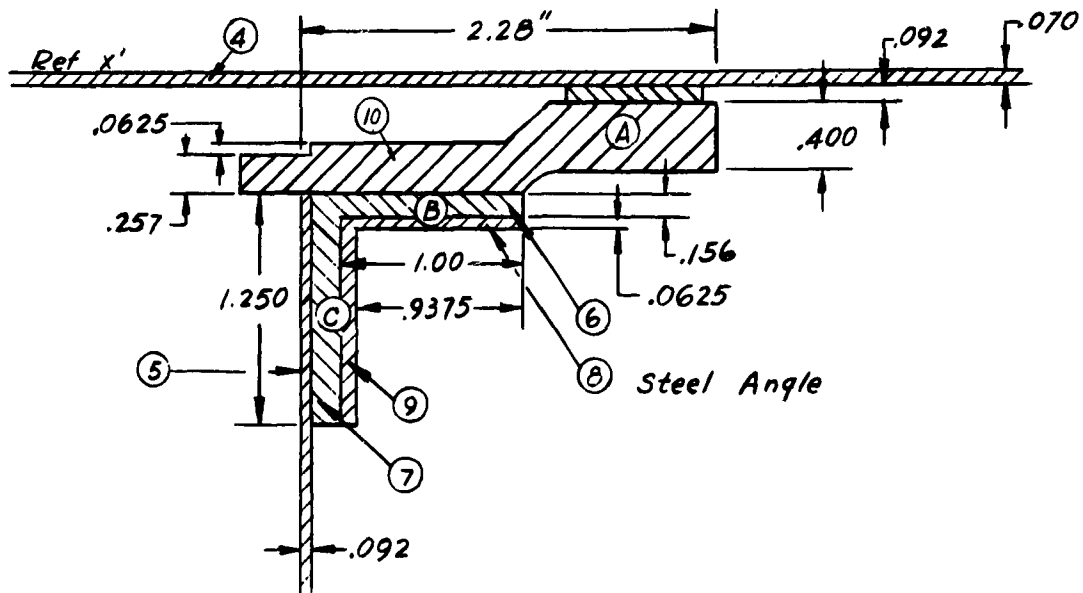
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Section Properties - Wing Station 117.5

Front Spar Section - Items 4, 5, 6, 7, 8, 9, 10.

(Ref. P. 7)



Buckling Stress

Element A

$$\frac{b_1}{t_1} = \frac{2.28}{.359} = 6.3$$

$$F_{cr1} = 42,500$$

(Ref. I - Fig. B5.13)

Element B

$$\frac{b_2}{t_2} = \frac{.9375}{.218} = 4.3$$

$$F_{cr2} = 46,500 \text{ psi}$$

(Ref. I - Fig. B5.13)

Element C

$$\frac{b_3}{t_3} = \frac{.938}{.264} = 3.55$$

$$F_{cr3} = 49,000 \text{ psi}$$

(Ref. I - Fig. B5.13)

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Section Properties - Wing Station 117.5

$$F_{cs} = \frac{(42,500 \times 2.28 \times .359) + (46,500 \times .937 \times .218) + (49,000 \times .938 \times .264)}{(2.28 \times .359) + (.937 \times .218) + (.938 \times .264)}$$

$$F_{cs} = \frac{56,450}{1.272} = 44,500 \text{ psi} \quad (\text{Ref. P. 14})$$

Effective Skin

$$w = 1.7 \times .070 \sqrt{\frac{10.3 \times 10^6}{4.45 \times 10^4}} \quad (\text{Ref. I - P. B4.2(6)})$$

$$w = 1.81 \text{ in.}$$

Section Properties

Item	A	y'	A _{y'}	A _{y'} ²
4 1.81 x .070	.126	.035	.004	0
5 1.250 x .092	.115	1.30	.149	.194
6 1.00 x .156	.156	.800	.125	.100
7 1.250 x .156	.195	1.30	.254	.254
8 .9375 x .062	.058	.950	.055	.055
9 1.09 x .062	.067	1.42	.095	.095
10	.981	.400	.392	.392
Σ	1.698	ȳ = .633	1.074	.968

(Ref. P. 14)

$$\bar{y} = \frac{1.074}{1.698} = .633$$

$$I_{xx} = \Sigma A_{y'}^2 - \Sigma A \bar{y}^2 \quad (\text{Ref. I - P. A3.11})$$

$$I_{xx} = .968 - (1.698 \times (.633)^2)$$

$$I_{xx} = .968 - .68 = .288 \text{ in.}^4$$

$$\rho = \sqrt{\frac{.288}{1.698}} = .412 \text{ in.}$$

(Ref. I - P. A3.11)

Column Allowable Buckling Stress

$$F_c = F_{cs} - \frac{F_{cs} \times \left(\frac{L}{\rho}\right)^2}{4 c \pi^2 E} \quad (\text{Ref. I - P. B6.4(3)})$$

$$F_c = 44,500 - \frac{(44,500) \times \left(\frac{10.75}{.412}\right)^2}{8.1 \times 10^6}$$

$$F_c = 44,500 - 1670 = 42,830 \text{ psi}$$

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Section Properties - Wing Station 117.5

Revised Effective Skin

$$w = 1.7 t \sqrt{\frac{E}{F_c}} \quad (\text{Ref. I - P. B6.4})$$

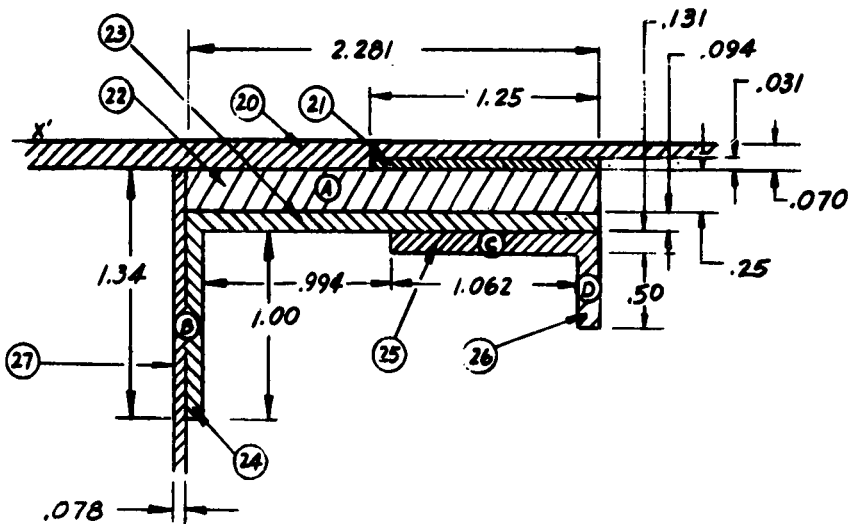
$$w = 1.7 \times .07 \sqrt{\frac{10.3 \times 10^6}{4.28 \times 10^4}} \quad (\text{Ref. P. 15})$$

$$w = 1.85 \text{ in.}$$

Area of Skin

$$A = 1.85 \times .070 = .130 \text{ in.}^2$$

Rear Spar Section - Items 20, 21, 22, 23, 24, 25, 26, 27. (Ref. P. 7)



Buckling Stress

Element A

$$\frac{b_1}{t_1} = \frac{2.187}{.297} = 7.4$$

$$F_{cr1} = 40,000 \text{ psi}$$

(Ref. I - Fig. B5.13)

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Section Properties - Wing Station 117.5

Element B

$$\frac{b_2}{t_2} = \frac{1.00}{.133} = 7.5$$

$$F_{cr2} = 40,000 \text{ psi} \quad (\text{Ref. I - Fig. B5.13})$$

Element C

$$\frac{b_2}{t_2} = \frac{1.062}{.131} = 8.1 \quad (\text{Ref. I - Fig. B5.13})$$

$$F_{cr3} = 39,500 \text{ psi}$$

Element D

$$\frac{b_4}{t_4} = \frac{.500}{.131} = 3.8 \quad (\text{Ref. I - Fig. B5.13})$$

$$F_{cr4} = 48,500 \text{ psi}$$

$$F_{cs} = \frac{(40,000 \times 2.187 \times .297) + (48,500 \times .5 \times .131) + (40,000 \times 1. \times .133) + 2.187 \times .297) + (.5 \times .131) + (1.00 \times .133) +$$

$$\frac{(39,500 \times 1.062 \times .131)}{1.062 \times .131} \quad (\text{Ref. I - P. B5.7 (5) })$$

$$F_{cs} = \frac{39,990}{.987} = 40,500 \text{ psi}$$

Effective Skin

$$w = 1.7 \times .070$$

$$\sqrt{\frac{10.3 \times 10^6}{4.05 \times 10^4}}$$

(Ref. I - P. B4.2(6))

$$w = 1.90 \text{ in.}$$

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Section Properties - Wing Station 117.5

Section Properties (Ref. P. 16)

Item	A	y'	Ay'	Ay' ²
20 1.90 x .070	.133	.035	.0046	.00016
21 1.25 x .031	.039	.081	.0031	.00026
22 2.281 x .25	.570	.222	.1265	.0281
23 2.281 x .094	.214	.394	.0843	.0332
24 1.00 x .094	.094	.941	.0885	.0890
25 1.062 x .131	.139	.506	.0704	.0356
26 .631 x .131	.082	.756	.0620	.0468
27 1.34 x .078	.104	.767	.0797	.0614
Σ	1.374	$\bar{y} = .378$	.519	.2945

$$\bar{y} = \frac{\Sigma Ay'}{\Sigma A} \quad (\text{Ref. I - P. A3.11})$$

$$\bar{y} = \frac{.519}{1.374} = .378 \text{ in.}$$

$$I_{xx} = \Sigma Ay'^2 - \Sigma A \bar{y}^2 \quad (\text{Ref. I - P. A3.11})$$

$$I_{xx} = .2945 - 1.374 \times (.378)^2$$

$$I_{xx} = .2945 - .195 = .0995 \text{ in.}^4$$

$$\rho = \sqrt{\frac{I_{xx}}{A}} \quad (\text{Ref. I - P. A3.5})$$

$$= \sqrt{\frac{.0995}{1.374}} = .269 \text{ in.}$$

Column Allowable Buckling Stress (Ref. I - P. B6.4(3))

$$F_c = 40,500 - \frac{(40,500) \times \left(\frac{10.75}{.269}\right)^2}{8.1 \times 10^8} \quad (\text{Ref. P. 17})$$

$$F_c = 40,500 - \frac{26.2 \times 10^{11}}{8.1 \times 10^8}$$

$$F_c = 40,500 - 3240 = 37,260 \text{ psi}$$

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Section Properties - Wing Station 117.5

Revised Effective Skin

(Ref. I - P. B6.4)

$$w = 1.7 \times .070 \sqrt{\frac{10.3 \times 10^6}{3.726 \times 10^4}}$$

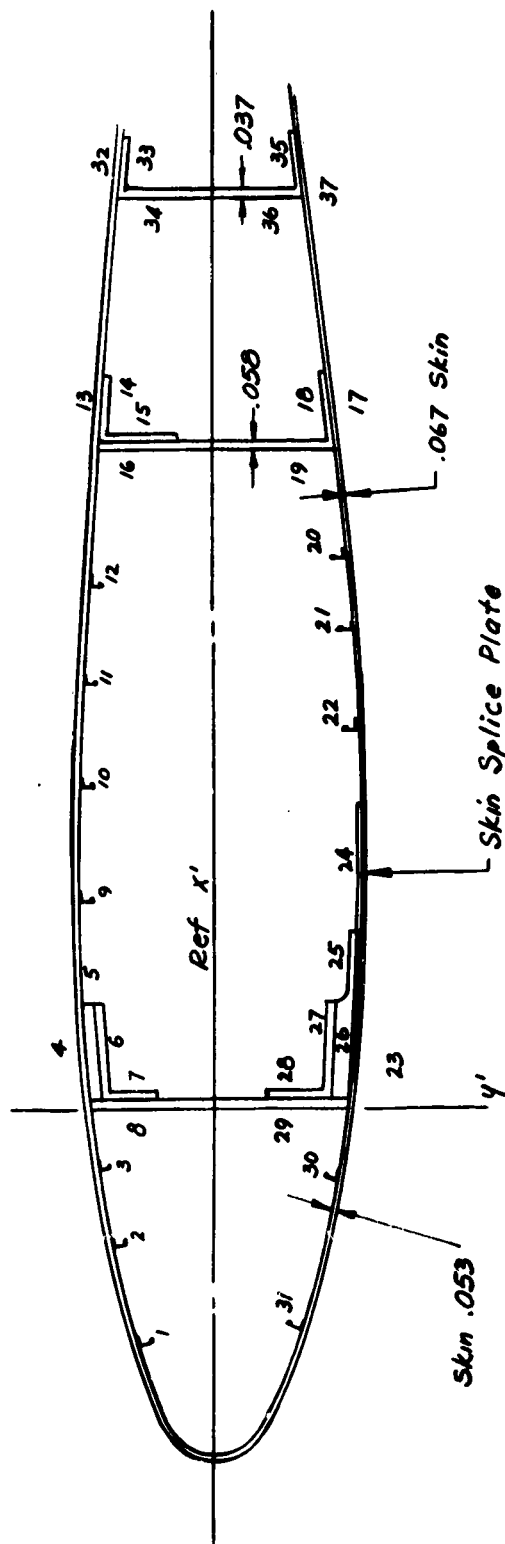
(Ref. P. 18)

$$w = 1.98 \text{ in.}$$

$$\text{Area of skin } A = 1.98 \times .070 = .138 \text{ in.}^2$$

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SECTION - WING STA 179.4 (Ref. 2)

Fig. 3

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SECTION PROPERTIES - WING STATION 179.4

1 Item	2 A	3 y'	4 Ay'	5 Ay' ²	6 x'	7 Ax'	8 Ax' ²	9 Ax'y'
1	.243	2.250	.547	1.235	-11.187	-2.72	30.4	- 6.12
2	.243	3.156	.765	2.405	- 6.156	-1.49	9.2	- 4.72
3	.243	3.453	.840	2.890	- 2.984	- .72	2.1	- 2.50
4	.077	3.890	.300	1.170	.641	.04	.0	.19
5	.187	3.781	.706	2.675	.656	.12	.0	.46
6	.150	3.640	.546	1.980	.656	.09	.0	.35
7	.109	3.125	.340	1.060	.094	.01	-	.03
8	.067	3.250	.218	.710	-	-	-	-
9	.243	3.875	.940	3.640	6.844	1.66	11.3	6.42
10	.243	3.969	.965	3.840	11.812	2.77	33.8	11.00
11	.243	3.922	.953	3.740	16.719	4.06	68.0	15.90
12	.243	3.609	.876	3.160	21.596	5.25	113.0	18.90
13	.097	3.453	.335	1.160	26.000	2.52	65.8	8.70
14	.051	3.359	.171	.575	26.031	1.33	34.4	4.46
15	.058	2.937	.170	.502	25.560	1.48	37.9	4.35
16	.058	2.937	.170	.502	25.500	1.48	37.6	4.34
17	.355	-2.800	-.995	2.770	25.946	9.22	238.0	-25.80
18	.058	-2.750	-.159	.437	26.033	1.51	39.2	- 4.15
19	.116	-1.812	-.210	.380	25.500	2.96	75.5	- 5.36
20	.586	-3.000	-1.760	5.280	20.096	11.78	237.5	-35.40
21	.586	-3.156	-1.850	5.800	14.000	8.20	115.0	-25.90
22	.586	-3.125	-1.830	5.700	7.625	4.47	34.0	-13.94
23	.490	-2.953	-1.445	4.260	.562	.27	.1	-.81
24	.178	-2.953	-.525	1.550	2.500	.44	1.1	- 1.31
25	.148	-2.844	-.420	1.190	1.687	.25	.4	-.71
26	.292	-2.828	-.826	2.340	.625	.18	.1	-.51
27	.146	-2.625	-.384	1.010	.625	.09	.0	-.24
28	.132	-2.000	-.264	.528	.094	.01	.0	-.02
29	.116	-1.937	-.225	.436	-	-	-	-
30	.165	-2.297	-.379	.870	- 6.281	-1.03	6.5	+ 2.38
31	.165	-1.719	-.284	.487	-11.250	-1.86	20.8	+ 3.20
32	.203	2.10	.426	.894	34.810	7.06	246.0	+14.80
33	.041	2.03	.083	.168	34.820	1.43	49.6	2.90
34	.037	1.66	.061	.102	34.240	1.27	43.3	2.10
35	.041	-1.77	-.072	.128	34.830	1.43	49.6	- 2.51
36	.074	-.86	-.063	.055	34.240	2.53	86.6	- 2.16
37	.455	-2.18	-.990	2.160	31.940	14.50	464.0	-31.60
Σ	7.488		-3.210	67.830		79.61	2151.3	-63.26

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$$\bar{y} = \frac{-3.21}{7.488} = -0.428 \text{ in.}$$

(Ref. I - P. A3.11)
(Ref. P. 21)

$$\begin{aligned} I_{xx} &= \sum A y'^2 - \sum A x (\bar{y})^2 \\ &= 67.8 - 7.5 \times (-0.428)^2 \\ &= 67.8 - 1.31 \\ &= 66.5 \text{ in.}^4 \end{aligned}$$

(Ref. I - P. A3.11)
(Ref. P. 21)

$$\bar{x} = \frac{79.61}{7.5} = 10.62 \text{ in.}$$

$$\begin{aligned} I_{yy} &= \sum A x^2 - \sum A x (\bar{x})^2 \\ &= 2151.3 - 7.5 \times (10.62)^2 \\ &= 2151.3 - 847 \\ &= 1304 \text{ in.}^4 \end{aligned}$$

(Ref. I - P. A3.11)
(Ref. P. 21)

$$\begin{aligned} I_{xy} &= \sum A x' y' - \sum A x \bar{x} \times \bar{y} \\ &= -63.26 - 7.5 \times 10.62 \times -0.428 \\ &= -63.26 + 34.1 \\ &= -29.16 \text{ in.}^4 \end{aligned}$$

(Ref. I - P. A3.11)
(Ref. P. 21)

$$\tan 2\phi = \frac{2 I_{xy}}{I_y - I_x}$$

(Ref. I - P. A3.11)

$$\tan 2\phi = \frac{-58.32}{1237.5} = -0.0472$$

$$\phi = -1^\circ 20'$$

$$I_{xp} = I_x \cos^2 \phi + I_y \sin^2 \phi - 2 I_{xy} \sin \phi \times \cos \phi \quad (\text{Ref. I - P. A3.11})$$

$$I_{xp} = 66.5 \times (.999)^2 + 1304 \times (.0236)^2 - [2 \times -29.16 \times -.0236 \times .999]$$

$$I_{xp} = 66.5 + .73 - 1.37$$

$$I_{xp} = 66 \text{ in.}^4$$

$$I_{yp} = I_x \sin^2 \phi + I_y \cos^2 \phi + 2 I_{xy} \sin \phi \times \cos \phi \quad (\text{Ref. I - P. A3.12})$$

$$I_{yp} = 66.5 \times (.0236)^2 + 1304 \times (.999)^2 + [2 \times -29.16 \times -.0236 \times .999]$$

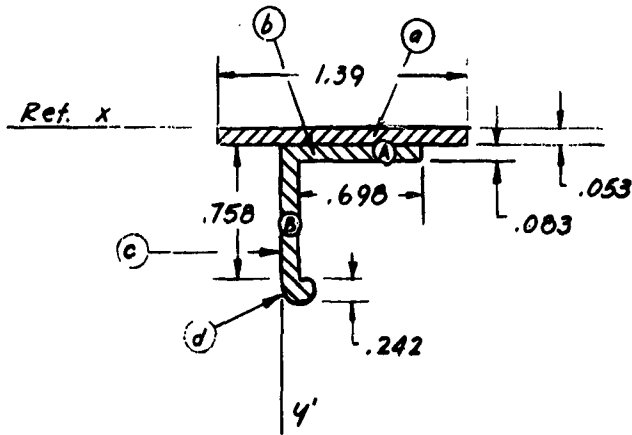
$$= .037 + 1304 + 1.37$$

$$= 1305 \text{ in.}^4$$

Section Properties - Wing Station 179.4

Bulb Angle Section - Items 1, 2, 3, 9, 10, 11, 12.

(Ref. P. 20)



Buckling Stress

Element A

$$\frac{b_1}{t_1} = \frac{.698}{.083} = 8.4$$

$$F_{cr1} = 38,000 \text{ psi}$$

(Ref. I - Fig. B5.13)

Element B

$$\frac{b_2}{t_2} = \frac{.917}{.083} = 11.0$$

$$F_{cr2} = 48,750 \text{ psi}$$

(Ref. I - Fig. B5.13)

$$F_{cs} = \frac{(F_1 b_1 + F_2 b_2)}{b_1 + b_2}$$

(Ref. I - P. B5.7(6))

$$F_{cs} = \frac{(38,000 \times .698) + (48,750 \times .917)}{.698 + .917}$$

$$F_{cs} = \frac{71,100}{1.615} = 44,000 \text{ psi}$$

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Section Properties - Wing Station 179.4

Effective Skin

(Ref. I - P. B4.2(6))

$$w = 1.70 \times .053 \sqrt{\frac{10.3 \times 10^6}{4.4 \times 10^4}}$$

(Ref. P. 23)

$$w = .09 \times 15.3$$

$$w = 1.39 \text{ in.}$$

Section Properties

(Ref. P. 23)

Item	Area	y'	A _{y'}	A _{y'} ²	x'	A _{x'}
a	.074	.026	.002	.0	.432	.032
b	.058	.094	.005	.0	.432	.025
c	.063	.432	.027	.013	.041	.003
d	.046	.932	.043	.040	.121	.005
Σ	.241	$\bar{y} = .321$	.077	.053	$\bar{x} = .270$	.065

$$\bar{y} = \frac{\Sigma A y'}{\Sigma A} = \frac{.077}{.241} = .321 \text{ in. (Ref. I - P. A3.11)}$$

$$\bar{x} = \frac{\Sigma A x'}{\Sigma A} = \frac{.065}{.241} = .270 \text{ in.}$$

$$I_{xx} = \Sigma A y'^2 - \Sigma A \bar{y}^2 \quad (\text{Ref. I - P. A3.11})$$

$$I_{xx} = .053 - .241 \times (.321)^2$$

$$I_{xx} = .053 - .025$$

$$I_{xx} = .0283 \text{ in.}^4$$

$$\rho = \sqrt{\frac{I_{xx}}{A}} \quad (\text{Ref. I - P. A3.5})$$

$$\rho = \frac{.0283}{.2406} = .344 \text{ in.}$$

Johnson's Parabolic Equation

$$F_c = F_{cs} - \frac{F_{cs}^2}{4 c \pi} \times \left(\frac{L}{\rho}\right)^2 \quad (\text{Ref. I - P. B6.4(3) })$$

$$F_c = 44,000 - \frac{(44,000)^2}{4 \times 2 \times (3.14)^2} \times \left(\frac{11.75}{.344}\right)^2 \quad (\text{Ref. P. 23})$$

$$F_c = 44,000 - 2800$$

$$F_c = 41,200 \text{ psi}$$

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Section Properties - Wing Station 179.4

Revised Effective Skin

$$w = 1.7 t \sqrt{\frac{E}{F_c}} \quad (\text{Ref. I - P. B6.4})$$

$$w = 1.7 \times .053 \sqrt{\frac{10.3 \times 10^6}{4.12 \times 10^4}} = 1.43 \text{ in. (Ref. P. 24)}$$

$$A = 1.43 \times .053$$

$$A = .076 \text{ in.}^2$$

$$A = .167 \text{ in.}^2 \quad (\text{Ref. P. 24 b, c, d})$$

$$A = .243 \text{ in.}^2 \quad \text{Total Area}$$

Buckling Stress of Skin Between Rivets

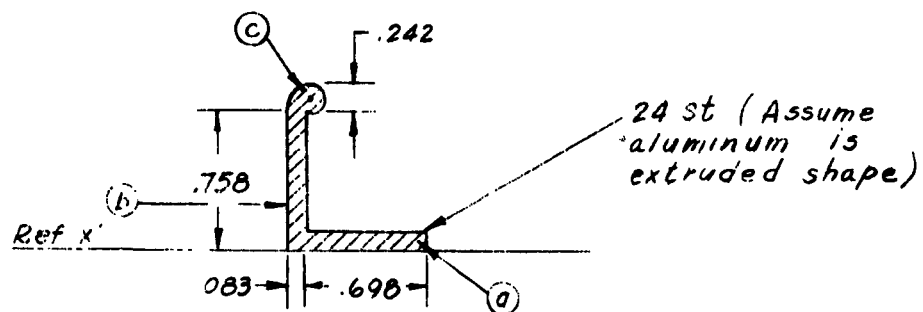
$$\text{Rivet Spacing} = 1.00 \text{ in.}$$

$$F_c = \text{Inter-rivet Buckling Stress of Skin}$$

$$F_c = 43,000 \text{ psi} \quad (\text{Ref. I - Fig. B4.4})$$

Typical Bulb Angle

Column Allowable Buckling Stress



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Section Properties - Wing Station 179.4

Section Properties (Ref. P. 25)

Item	A	y'	Ay'	Ay' ²
a .698 x .083	.058	.041	.0024	.0001
b .758 x .083	.063	.378	.0238	.009
c $\pi \times (.121)^2$	.046	.879	.0404	.0354
Σ	.167	y = .398	.0666	.0445

$$\bar{y} = \frac{\Sigma Ay'}{\Sigma A} \quad (\text{Ref. I - P. A3.11})$$

$$\bar{y} = \frac{.0666}{.167} = .398 \text{ in.}$$

$$I_{xx} = \Sigma Ay'^2 - \Sigma Ay^2 \quad (\text{Ref. I - P. A3.11})$$

$$I_{xx} = .0445 - .167 \times (.398)^2$$

$$I_{xx} = .0445 - .0264$$

$$I_{xx} = .0181 \text{ in.}^4$$

$$\rho = \sqrt{\frac{I_{xx}}{A}} \quad (\text{Ref. I - P. A3.5})$$

$$= \frac{.018}{.167}$$

$$= .328''$$

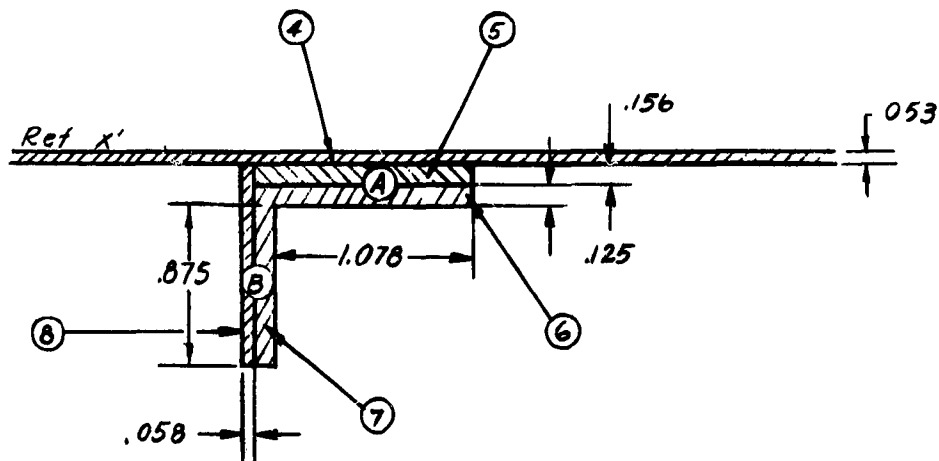
$$\frac{I}{\rho} = \frac{11.75}{.328} = 35.8$$

$$F_c = 37,500 \text{ psi} \quad (\text{Ref. I - Fig. B1.10})$$

Section Properties - Wing Station 179.4

Front Spar Section - Items 4, 5, 6, 7, 8.

(Ref. P. 20)



Buckling Stress

Element A

$$\frac{b_1}{t_1} = \frac{1.078}{.125 + \frac{.156}{2}} = 5.3$$

$$F_{cr1} = 44,500 \text{ psi}$$

(Ref. I - Fig. B5.13)

Element B

$$\frac{b_2}{t_2} = \frac{.875}{.125 + \frac{.058}{2}} = 5.7$$

$$F_{cr2} = 44,000 \text{ psi}$$

(Ref. I - Fig. B5.13)

$$F_{cs} = \frac{(44,500 \times 1.078 \times .203) + (44,000 \times .875 \times .154)}{(1.078 \times .203) + (.875 \times .154)} \quad (\text{Ref. I - P. B5.7 (5)})$$

$$F_{cs} = \frac{9750 + 5940}{.354} = \frac{15690}{.354} = 44,300 \text{ psi}$$

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Section Properties - Wing Station 179.4

Effective Skin - Item #4

$$w = 1.7 t \sqrt{\frac{E}{F_{cs}}} \quad (\text{Ref. I - P. B4.2(6)})$$

$$w = 1.7 \times .053 \sqrt{\frac{10.3 \times 10^6}{4.43 \times 10^4}} \quad (\text{Ref. P. 27})$$

$$w = .09 \times 15.3$$

$$w = 1.37 \text{ in.}$$

Section Properties

(Ref. P. 27)

Item	A	y'	A _{y'}	A _{y'} ²
4 1.37 x .053	.073	.026	.0019	0
5 1.203 x .156	.187	.131	.0245	.0032
6 1.078 x .125	.135	.271	.0366	.0099
7 1.00 x .125	.125	.709	.0886	.0628
8 1.156 x .058	.067	.631	.0423	.0268
Σ	.587	$\bar{y} = .326$	.1939	.1027

$$\bar{y} = \frac{\Sigma A y'}{\Sigma A} = \frac{.194}{.595} = .326 \text{ in.} \quad (\text{Ref. I - P. A3.11})$$

$$I_{xx} = \Sigma A y'^2 - \Sigma A \bar{y}^2 \quad (\text{Ref. I - P. A3.11})$$

$$I_{xx} = .1027 - .595 \times (.326)^2$$

$$I_{xx} = .040 \text{ in.}^4$$

$$\rho = \sqrt{\frac{I_{xx}}{A}} = \sqrt{\frac{.040}{.595}} = .26 \text{ in.} \quad (\text{Ref. I - P. A3.5})$$

Johnson's Parabolic Column Equation

$$F_c = F_{cs} - \frac{F_{cs}^2 \times \left(\frac{L}{\rho}\right)^2}{4 C \pi^2 E} \quad (\text{Ref. I - P. B6.4(3)})$$

$$F_c = 44,300 - \frac{(44,300)^2 \times \left(\frac{11.75}{.26}\right)^2}{8.12 \times 10^8}$$

$$F_c = 44,300 - 4950 = 39,350 \text{ psi}$$

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Section Properties - King Station 179.4

Revised Effective Skin

$$w = 1.70 \times .053 \quad \boxed{\frac{10.3 \times 10^6}{3.935 \times 10^4}}$$

(Ref. I - P. B6.4)

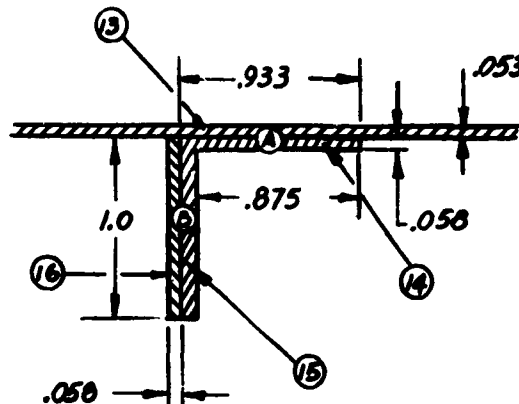
W = 1.46 in.

$$\begin{aligned} A &= 1.46 \times .053 \\ &= .077 \text{ in.}^2 \end{aligned}$$

(Ref. P. 28)

Rear Spar Section - Items 13, 14, 15, 16.

(Ref. P. 20)



Buckling Stress

Klement A

$$\frac{b_1}{t_1} = \frac{.875}{.058} = 15.05$$

$$F_{cr1} = 19,500 \text{ psi}$$

(Ref. I - Fig. B5.12)

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Section Properties - Wing Station 179.4

Element B

$$\frac{b_2}{t_2} = \frac{.942}{.087} = 10.8 \quad (\text{Ref. P. 29})$$

$$F_{cr2} = 31,500 \quad (\text{Ref. I - Fig. B5.12})$$

$$F_{cs} = \frac{(19,500 \times .875 \times .058) + (31,500 \times .942 \times .087)}{(.875 \times .058) + (.942 \times .087)} \quad (\text{Ref. I - P. B5.7 (5)})$$

$$F_{cs} = \frac{990 + 2580}{.0507 + .0819} = \frac{3570}{.1326} \quad (\text{Ref. P. 29})$$

$$F_{cs} = 26,900 \text{ psi}$$

Effective Skin Width

$$w = 1.7 \times .053 \quad \sqrt{\frac{10.3 \times 10^6}{2.69 \times 10^4}} \quad (\text{Ref. I - P. B4.2(6)})$$

$$w = 1.77 \text{ in.}$$

Section Properties

(Ref. P. 29)

Item	A	y'	Ay'	Ay' ²
13	1.77 x .053	.094	.026	.0024
14	.875 x .058	.051	.082	.0042
15	1.00 x .058	.058	.553	.0321
16	1.00 x .058	.058	.553	.0321
Σ	.261	$\bar{y} = .272$	.0708	.0358

$$\bar{y} = \frac{\Sigma Ay'}{\Sigma A} \quad (\text{Ref. I - P. A3.11})$$

$$\bar{y} = \frac{.071}{.261} = .272 \text{ in.}$$

$$I_{xx} = \Sigma Ay'^2 - \Sigma A\bar{y}^2 \quad (\text{Ref. I - P. A3.11})$$

$$I_{xx} = .036 - (.261 \times (.272)^2)$$

$$I_{xx} = .036 - .0193 = .0167 \text{ in.}^4$$

Section Properties - Wing Station 179.4

$$\rho = \sqrt{\frac{I_{xx}}{A}} \quad (\text{Ref. I - P. A3.5})$$

$$= \sqrt{\frac{.0167}{.261}} \quad (\text{Ref. P. 30})$$

$$= .253 \text{ in.}$$

Column Allowable Buckling Stress

$$F_c = F_{cs} - \frac{F_{cs}^2 \times \left(\frac{l}{\rho}\right)^2}{4 \times c \times \pi^2 \times E} \quad (\text{Ref. I - P. B6.4(3)})$$

$$F_c = 26,900 - \frac{(26,900^2) \times \left(\frac{11.75}{.253}\right)^2}{8.12 \times 10^8} \quad (\text{Ref. P. 30})$$

$$F_c = 26,900 - 1920 = 24,980$$

$$F_c = 24,980 \text{ psi}$$

Revised Effective Skin

(Ref. I - P. B6.4)

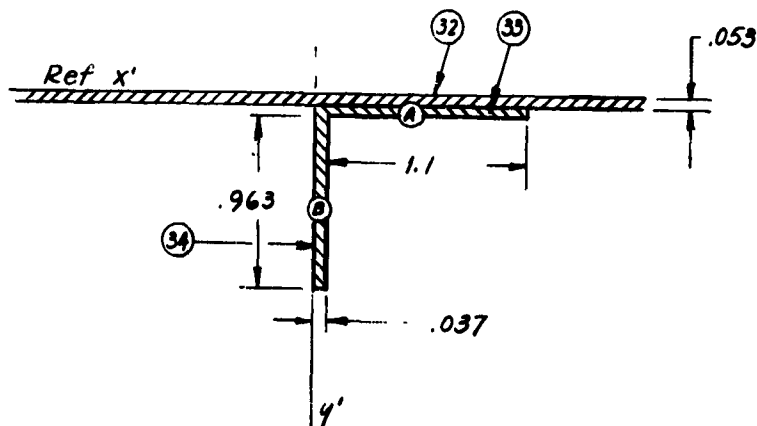
$$w = 1.7 \times .053 \sqrt{\frac{10.3 \times 10^6}{2.498 \times 10^4}}$$

$$w = 1.835 \text{ in.}$$

$$A = 1.835 \times .053 = .097 \text{ in.}^2$$

Aileron Closure Spar - Items 32, 33, 34.

(Ref. P. 20)



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Section Properties - Wing Station 179.4

Buckling Stress

Element A

(Ref. P. 31)

$$\frac{b_1}{t_1} = \frac{1.1}{.037} = 29.8$$

$$F_{cr1} = 5,000 \text{ psi}$$

(Ref. I - Fig. B5.12)

Element B

$$\frac{b_2}{t_2} = \frac{.963}{.037} = 26$$

$$F_{cr2} = 7,000$$

$$F_{cs} = \frac{(5,000 \times 1.1) + (7,000 \times .963)}{1.1 + .963} \quad (\text{Ref. I - P. B5.7(6)})$$

$$F_{cs} = \frac{5,500 + 6,740}{2.063}$$

$$F_{cs} = \frac{12,240}{2.063} = 5940$$

Effective Skin

$$w = 1.7 \times .053 \sqrt{\frac{10.3 \times 10^6}{5.94 \times 10^3}} \quad (\text{Ref. I - P. B4.2(6)})$$

$$w = 3.74 \text{ in.}$$

Section Properties

(Ref. P. 31)

Item	A	y'	Ay'	Ay' ²	x'	Ax'
32 3.74 x .053	.198	.0265	.0052	.00014	.5685	.1125
33 1.10 x .037	.0407	.0715	.0029	.00021	.5685	.0231
34 1.0 x .037	.037	.5053	.0187	.00945	.0185	.0007
Σ	.2757	y = .103	.0268	.00980	x = .485	.1363

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Section Properties - Wing Station 179.4

(Ref. I - P. A3.11)

$$\bar{y} = \frac{.0268}{.2757} = .103 \text{ in.} \quad (\text{Ref. P. 32})$$

$$\bar{x} = \frac{\sum A x_1}{\sum A} = \frac{.1363}{.2757} = .485 \text{ in.} \quad (\text{Ref. I - P. A3.11})$$

$$I_{xx} = .0098 - (.2757 \times (.103)^2) \quad (\text{Ref. P. 32})$$

$$I_{xx} = .0098 - .0029$$

$$I_{xx} = .0069 \text{ in.}^4$$

$$\rho = \sqrt{\frac{.0069}{.2757}} = .158 \text{ in.} \quad (\text{Ref. I - P. A3.5})$$

Buckling Stress of Column

(Ref. I - P. B6.4(3))

$$F_c = 5,940 - \frac{(5,940)^2 \times \left(\frac{11.75}{.158}\right)^2}{8.12 \times 10^8} \quad (\text{Ref. P. 32})$$

$$F_c = 5,940 - \frac{35.6 \times 10^6 \times (74.4)^2}{8.12 \times 10^8}$$

$$F_c = 5940 - \frac{1940}{8.12}$$

$$F_c = 5940 - 239 = 5,701 \text{ psi}$$

Revised Effective Skin

(Ref. I - P. B6.4)

$$w = 1.7 \times .053 \sqrt{\frac{10.3 \times 10^6}{5.701 \times 10^3}}$$

$$w = 3.82 \text{ in.}$$

Area of Skin

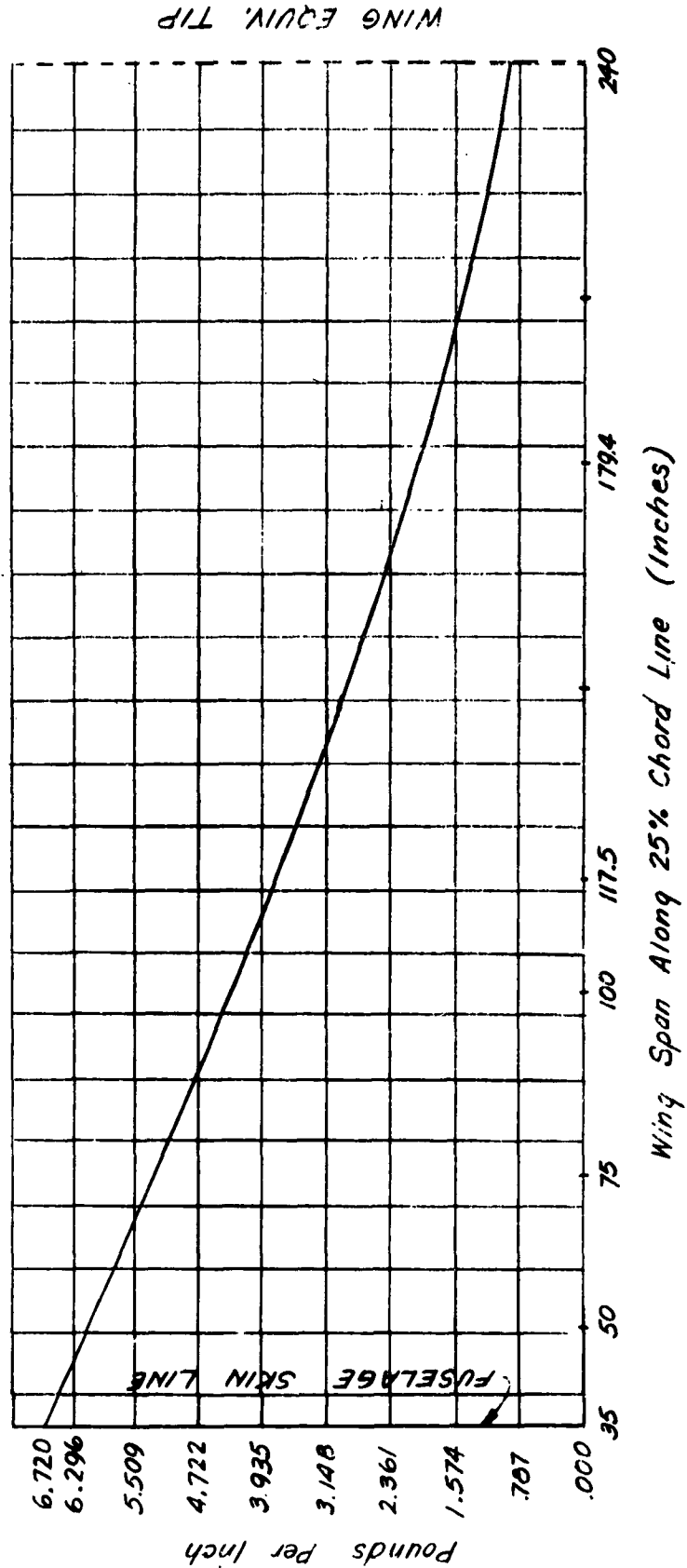
$$A = 3.82 \times .053 = .203 \text{ in.}^2$$

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MIG-15 SPAN-WISE DEAD WEIGHT DISTRIBUTION
TOTAL DISTRIBUTED WEIGHT L & R 1383 LBS.
SEMI-SPAN WEIGHT - 692 LBS



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SECTION II

UNIT BENDING MOMENTS

UNIT DEAD WEIGHT BENDING MOMENTS

(Ref. P. 34)

(1) Station	(2) Shear (Lbs.)	(3) * Arm (In.) C.G. to Sta.	(4) Moment (in.lbs.) (2) x (3)
179.4	91.5	26.2	2,400
117.5	-	-	2,400
	91.5	61.9	5,650
	179.5	27.2	4,880
Σ	271.0	-	12,930
35.0	-	-	12,930
	271.0	82.5	22,300
	421.0	37.2	15,600
Σ	692.0	-	50,830 +

*NOTE - The moment arms for above items were measured along 25% chord line.

+ To account for Wing sweepback of 35° at 25% Chordline.

$$M_w = 50,830 \times \cos 35^\circ = 41,600 \text{ in.lb.}$$

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WING PLANTFORM

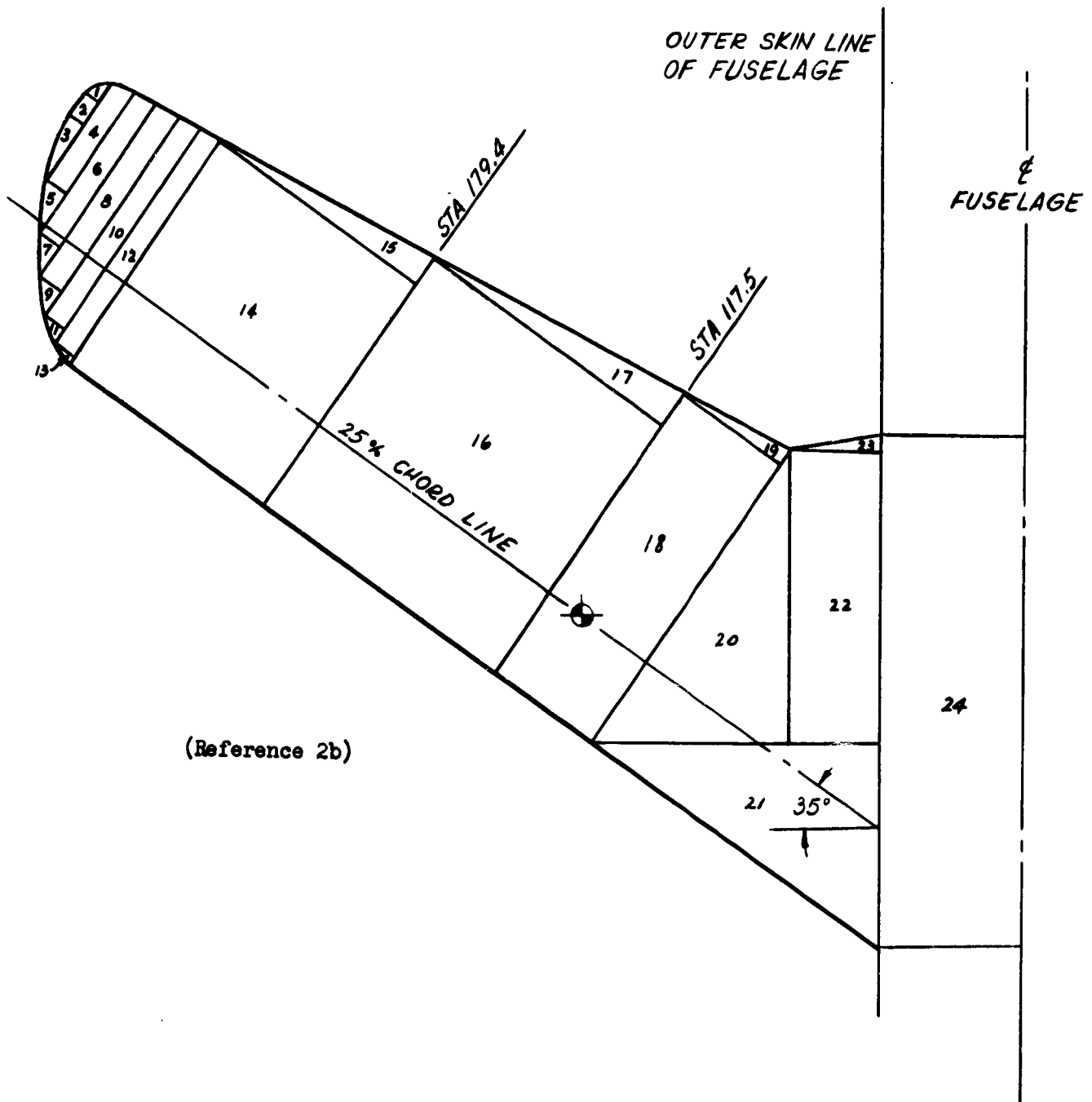


Fig. 4

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UNIT BENDING MOMENTS FOR 1000 LB. AIRLOAD

Wing Tip (Sta. 266.5) to Wing Station 179.4

(1) Item (Ref. P. 36)	(2) Area in. ²	(3) Load (lbs.) .0625 x (2)	(4) *Arm (Inches) CG to Sta. 179.4	(5) Moment (in. lbs.) (3) x (4)
1	5.4	.34	74.5	25.2
2	17.4	1.08	75.0	81.0
3	18.7	1.17	74.5	87.0
4	107.0	6.7	71.5	478.0
5	26.5	1.66	70.5	117.0
6	163.5	10.25	66.3	680.0
7	19.5	1.22	65.3	79.6
8	206.0	12.9	61.3	790.0
9	16.3	1.02	60.5	62.7
10	242.5	15.2	56.3	855.0
11	10.5	.66	55.3	36.5
12	267.5	16.7	51	852.0
13	5.5	.34	50.5	17.4
14	2780.0	174.0	24.5	4260.0
15	179.0	11.2	16.3	182.5
Σ	4065.0	254.4		8605.0

Wing Sta. 179.4 to Wing Sta. 117.5

(1) Item (Ref. P. 36)	(2) Area in.	(3) Load (lbs.) .0625 x (2)	(4) *Arm (Inches) CG to Sta. 117.5	(5) Moment (in. lbs.) (3) x (4)
1-15	4065.0	-	-	8605.0
16	3730.0	234.0	59.0	15000.0
17	186.6	11.7	29.5	6900.0
Σ	7982.0	500.0	19.6	230.0
				30,735.0

*NOTE - The moment arms for the above items were measured along the 25% chord line.

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UNIT BENDING MOMENTS FOR 1000 LB. AIRLOAD

Wing Station 117.5 to Fuselage Skin Line (Wing Sta. 35)

(1) Item (Ref. P. 36)	(2) Area in. ²	(3) Load (lbs.) $q \times (2)$	(4) *Arm (Inches) CG to Sta. 35	(5) Moment (in.lbs.) (3) $\times$ (4)
1-17	7982.0	-	-	24,800 †
18	-	500.1	58.5	29,200
19	1690.0	106.0	48.5	5,140
20	34.2	2.1	25.0	54
21	1220.0	76.5	31.0	2,370
22	1170.0	73.6	19.5	1,440
23	1040.0	65.2	9.0	586
24	15.8	1.0	6.0	6
Σ	13,152.0	824.4	-	63,596
Σ	2,910.0			
Σ	16,062.0 ‡			

*NOTE - The moment arms were measured as perpendicular distances from the fuselage skin line to the centroids of the above items.

† To account for the wing sweepback of 35° at 25% chordline.

$$\therefore \Sigma M_a = 30,735 \times \cos 35^\circ = 24,800 \text{ in.lb. (Ref. P. 37)}$$

‡ Area of section from Sta. 35 to fuselage centerline.

$$\therefore q = \frac{1000}{16062} = .0625 \text{ psi}$$

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SECTION III

DETERMINATION OF LOAD FACTORS

For any point on a section, the load factor

$$N = \frac{IF_c}{y (M_a - M_w)}$$

Where:

- I = Moment of inertia of section
- M_a = Airload moment, in.-lbs. for 1 g
- M_w = Dead weight moment, in.-lbs. for 1 g
- y = Distance from principal axis from drawing (Ref. 2a)
- F_c = Allowable stress

At Sta. 117.5

$$\begin{aligned} M_a &= 30735 \quad \times \quad 5.1 \quad = \quad 156,900 && (\text{Ref. P. 37}) \\ M_w &= 12930 && (\text{Ref. P. 35}) \\ I &= 192.4 && (\text{Ref. P. 9}) \\ N &= \frac{192.4 F_c}{143800 y} = .001338 \frac{F_c}{y} \end{aligned}$$

At Sta. 179.4

$$\begin{aligned} M_a &= 8605 \quad \times \quad 5.1 \quad = \quad 43,900 && (\text{Ref. P. 37}) \\ M_w &= 2400 && (\text{Ref. P. 35}) \\ I &= 66 && (\text{Ref. P. 22}) \\ N &= \frac{66 F_c}{41500 y} = .001592 \frac{F_c}{y} \end{aligned}$$

The calculations are included in a table on page

At Sta. 0

$$\begin{aligned} M_a &= 63,596 \times 5.1 = 324,000 && (\text{Ref. P. 38}) \\ M_w &= 41,600 && (\text{Ref. P. 35}) \end{aligned}$$

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Determination of Load Factors

Sta. 117.5

Item	(1)		(2)	(3)	(4)
Ref. p - (7)	F _c		y*	F _c / y	N
		Ref.		(1)/(2)	.001338x(3)
2	41380	P. 11	4.3	9630	12.9
3	41380	P. 11	4.4	9420	12.6
4-10	42830	P. 15	4.42	9700	13.0
20-27	37260	P. 18	2.03	18400	24.6
15	43040	P. 13	4.1	10750	14.4

Sta. 179.4

Item	(1)		(2)	(3)	(4)
Ref. p - (20)	F _c		y*	F _c / y	N
		Ref.		(1)/(2)	.001592x(3)
3	41200	P. 24	3.8	10830	17.3
4-8	39350	P. 28	4.12	9570	15.2
9	41200	P. 24	4.5	9150	14.6
11	41200	P. 24	4.80	8580	13.7
13-16	24980	P. 31	4.35	5850	9.15

*NOTE - These values are scaled from drawing.

(Ref. 2a)

At Sta. 0

(Ref. P. 6)

$$I_{xx} = 74.7 \text{ in.}^4$$

$$y = 4.29$$

$$N = \frac{IF_c}{y (M_a - M_w)}$$

$$= \frac{74.7 (174000)}{4.29 (324000 - 41600)} = 10.75$$

(Ref. P. 39)

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 - (a) No. 391-5-1013
 - (b) No. 391-5-1002
3. Work Statement Exhibit "A"
Enclosure (A) to C.A.L. letter RFF:emc-446 - 10/31/52
4. TR-AC-6 - C.A.L. Bill of Material Study

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WASHINGTON, DC

23 June 2010

HAF/IMIO (MDR)
1000 Air Force Pentagon
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Bobby Sammons.
P.O. Box 1680
Cloudcroft, NM 88317-1680

Dear Mr. Sammons

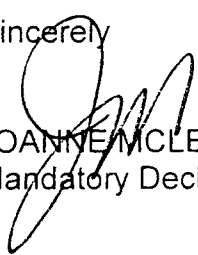
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The review for the documents have been completed and the declassification has been downgraded to UNCLASSIFIED and copies are attached for your information.

Address any questions concerning this review to the undersigned at (703) 692-9979 and refer to our case number 07-MDR-076.

Sincerely


JOANNE MCLEAN
Mandatory Declassification Review Specialist

- 2 Attachments
1. Letter, Request for Documents
 2. 10 DTIC Documents

cc: DTIC w/o documents