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STRUCTURAL SIZING OF A
SOLAR POWERED AIRCRAFT

David W. Hall and Stan A. Hall

LOCKHEED MISSILES & SPACE COMPANY
Sunnyvale, California 94086

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SYMBOLS

		UNITS
A	Cross-sectional Area	sq ft
AR	Aspect Ratio	-
b	Wingspan	ft
C	Chord or Coefficient	ft or dimensionless
C_C	Chordwise Coefficient	-
C_D	Drag Coefficient	-
C_L	Lift Coefficient	-
C_{L_α}	Wing Lift Curve Slope	per degree
C_M	Pitching Moment Coefficient	-
C_N	Normal Force Coefficent	-
d	Incremental Distance or Force	-
E	Young's Modulus	psi/in
F	Force or Stress	lb or psi
f	Aerodynamic Efficiency	-
h	Height or Depth	ft or in
I	Moment of Inertia	$lb\text{-}in^4$
j	Aspect Correction Factor	-
K	Constant	varies
L	Length	-
M	Moment	in-lb
N	Normal Force	lb
n	Load Factor	-
P	Load	lb
q	Dynamic Pressure	psf
R	Resultant Force	-

		UNITS
S	Wing Area	sq ft
T	Torque	in-lb
t/c	Thickness-to-Chord Ratio	-
U	Freestream Velocity	fps
V	Airspeed	fps
W	Weight	lb
X	Longitudinal Distance	in
α	Angle of Attack	deg
ϵ	Aerodynamic Twist	deg
μ	Moment	in-lb
π	3.14159	-
ρ	Density	lb/ft^3

SOLAR HAPP WEIGHT ESTIMATION

INTRODUCTION

Status of Previous Work

Previous weight estimation techniques used to size solar HAPPs (High Altitude Powered Platforms) have been based on algorithms accepted in the aerospace industry (References 1 through 4). These methods were modified where appropriate to reflect the very lightweight materials being used and to agree closely with a thorough preliminary design of another solar HAPP done in 1980 by Stanhall Aerosystems. The results of this work have been proprietary and remain unpublished.

The work done for NASA in FY82, which culminated in the description of the methodology needed to design solar HAPP's, lays out the equations used to arrive at a rough weight statement. Since the primary purpose of this work was to analyze the interactions of power train components to assess the effects of improvements in the state-of-the-art, these methods were adequate to fill in this very important gap in relating a power train to an overall vehicle. The algorithms developed to describe power train interactions were, in fact, thorough enough that confidence in their accuracy should be within $\pm 10\%$. This is not true of either the aerodynamic or structural algorithms.

Purpose of Current Work

The purpose of the work described in this report is to build a more accurate structural weight estimation model to be used with the power train methodology previously done or with other conceptual design efforts.

Scope

The current work analyzes three wing bracing schemes, and scales one with gross weight, wing loading, aspect ratio, and wingspan. The work does not include revisions to either power train or aerodynamic analytical methods described in NASA CR 3699 (Ref. 5).

DESCRIPTION OF WORK

Vehicle Designs

The conceptual HAPP RPV (Remotely Piloted Vehicle) which was chosen for detailed structural analyses in this work is a modification of the MK20 vehicle analyzed in Reference 5. The wing is the same, as is the power train. Changes include addition of a high horizontal tail supported by twin verticals which are mounted on tailbooms. These surfaces replace the separate vertical and horizontal surfaces of the MK20. Figure 1 presents a general arrangement of the basic vehicle analyzed here and referred to in the text as the MK21.

Basic vehicle parameters such as wingspan, aspect ratio, wing area, gross mass, wing thickness-to-chord ratio, and horizontal and vertical tail volumes are the same for both the MK20 and MK21. Mass parameters other than structure are also the same for consistency. The basic MK21 was then modified with three bracing schemes:

- Fully cantilevered (MK21A);
- Externally braced with struts (MK21B); and
- Externally braced with wires (MK21C).

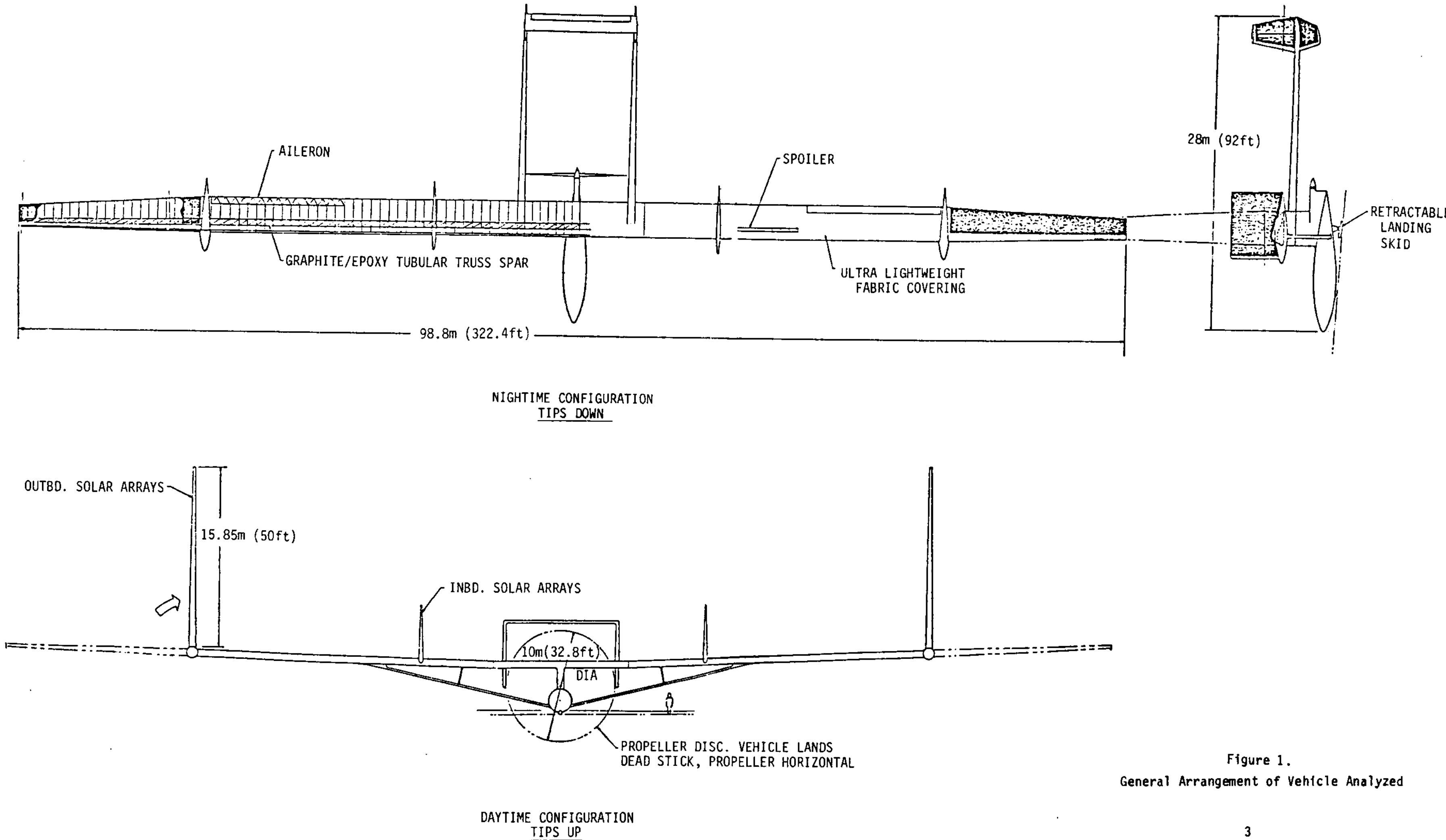


Figure 1.
General Arrangement of Vehicle Analyzed

Design of non-wing components was done once, and the results were used with all three designs.

Weights of Non-Spar Component Parts

Since a change in bracing scheme in the wing would only affect wing spar, strut, and wire bracing weights, all other structural components in the aircraft could be left constant. This includes wing leading and trailing edges, wing ribs, ailerons, and spoilers, all of which will be discussed here.

Wing Leading and Trailing Edges. The wing leading and trailing edge concepts used in this work are shown in Figure 2. The leading edge has been designed

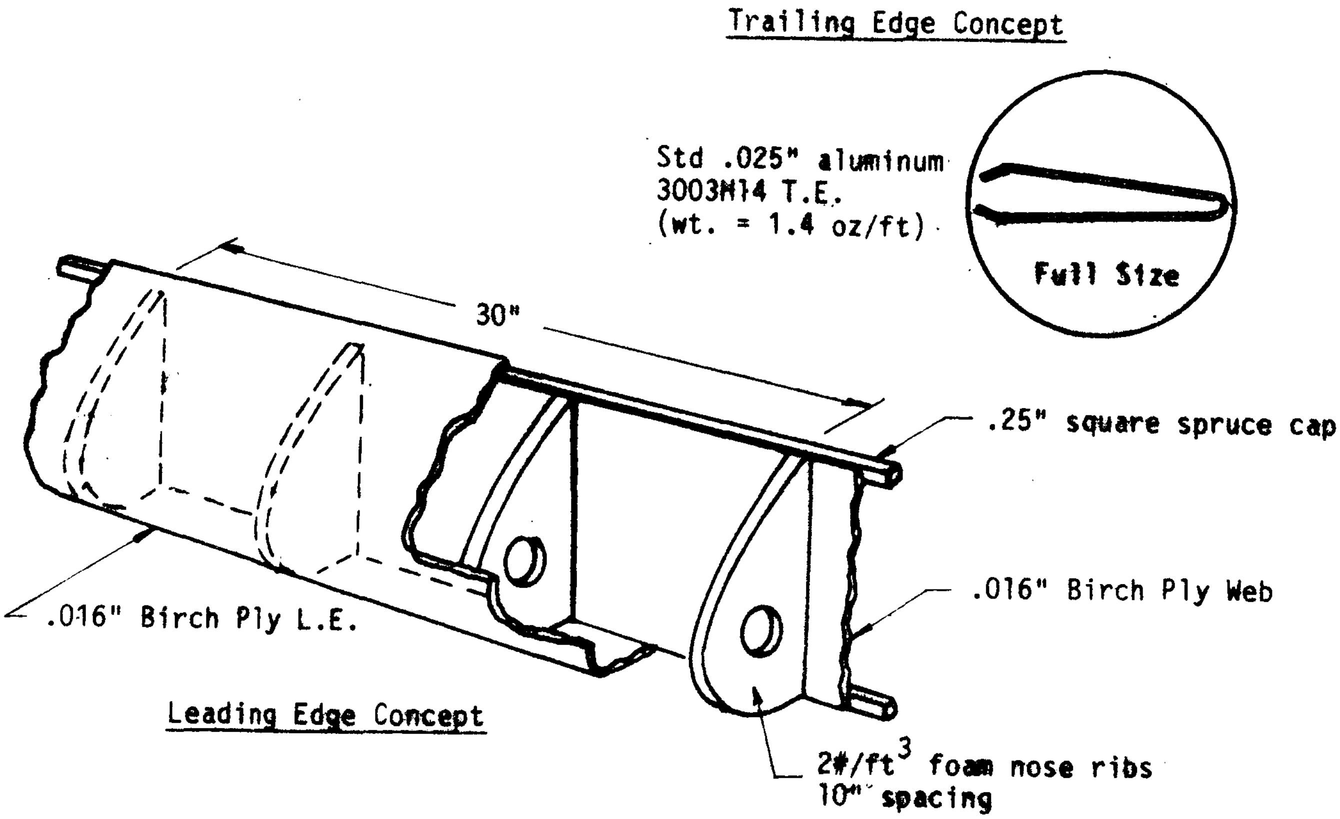


Figure 2. Wing Leading and Trailing Edge Concepts

to hold shape in order to minimize variations in airfoil characteristics along the wing. Basic structure is birch plywood with spruce caps and foam leading edge partial ribs every ten inches. The trailing edge structure is shaped birch plywood. The partial ribs are 0.300 inch thick styrofoam with a density of two pounds per cubic foot. Each piece would weigh 0.0105 pound and all pieces would be the same size. If rib spacing is ten inches, then 193 would be required. Spruce caps would be one-quarter inch square and a total of 1934.4 inches long each for a total weight of 3.92 pounds. The 0.016 inch birch plywood web would be 16.2 inches deep and would weigh 15.23 pounds over the entire span. Since the plywood comes in 50 inch square sheets, one-inch wide gussetts will be required every 50 inches for an additional weight of 0.31 pound. The 0.016 inch birch plywood leading edge skin would cover the entire span and be 24 inches wide for a weight of 22.57 pounds. One-inch gussetts would again be required for an additional 0.46 pound weight for 39. Total weight of these gussetts for the web and leading edge, then, is 0.77 pound. This brings the total weight of one wing leading edge, including 15% for adhesives, to

$$\frac{W_{WING}}{2}_{LE} = 1.15 \left(W_{RIB_{LE}} + W_{CAP_{LE}} + W_{WEB_{LE}} + W_{GUSSETT_{LE}} + W_{SKIN_{LE}} \right)$$

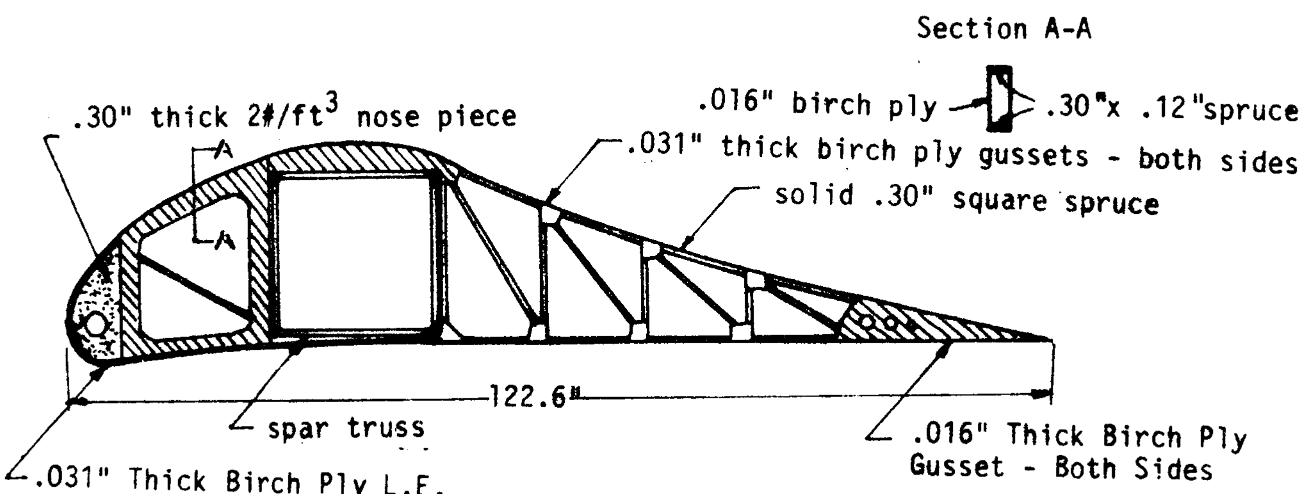
$$= 1.15 \times (2.03 + 3.92 + 15.23 + 0.77 + 22.57)$$

$$= 51.21\ \#$$

$$W_{WING_{LE}} = 102.42\ \#$$

The trailing edge would be made up of 0.025 inch thick 3003 H14 aluminum sheet weighing 1.4 ounces per foot. For a 124 foot run, this would be 10.86 pounds, or 21.72 pounds for both sides.

Wing Ribs. The airfoil section used is a Liebeck L1003 (Ref. 6) of 20% thickness-to-chord ratio. Figure 3 shows the makeup of a typical wing rib. Materials are birch plywood and spruce rod. Appendix A presents a photo of the authors holding a full-scale wing rib built of these materials. Ribs in both the untapered and tapered sections of the wing are similar in construction. Total length of 0.300 inch square spruce members is 500 inches, and density is 0.0162 pound per cubic inch, so the weight of these members is 0.73 pound per rib. The 0.300 x 0.12 inch spruce members will weigh 0.074 pound at the same density. The 0.031 inch birch gussetts will weigh 0.099 pound for a total area of 0.704 square foot ahead of the 40% chord rib center of gravity, 1.07 square feet aft, and a total weight of 0.29 pound per rib. Rib weight in the untapered section of the wing will be, then, 1.31 pounds including 10% adhesive weight.



Airfoil: 20% Liebeck L1003
Weight: 1.31#

Figure 3. Typical Wing Rib (Scale: 1" = 24")

The average weight of a rib in the tapered section of the wing will be approximated by averaging the weight of a constant-section rib and the weight of a wingtip rib. Given the same geometry and construction technique, the wingtip rib will be a ratio of chord lengths squared, or

$$W_{WING_{RIB}} = \left(W_{WING_{RIB}}\right)_{ROOT} \left(\frac{C_{TIP}}{C_{ROOT}}\right)^2$$

$$W_{WING_{RIB}} = 0.507^{\#}$$

An average rib, then, is

$$\left(W_{WING_{RIB}}\right)_{AVG} = \frac{\left(W_{WING_{RIB}}\right)_{ROOT} + \left(W_{WING_{RIB}}\right)_{TIP}}{2}$$

$$\left(W_{WING_{RIB}}\right)_{AVG} = 0.909^{\#}$$

Since plywood thickness stays the same in ribs and is not tapered with decreasing chord, this number will be increased about 20% to 1.10 pounds to be conservative.

Each wing half is made up of 43 constant chord ribs and 21 tapered ribs. Wing rib weight for each wing half, then, would be 79.96 pounds, or 159.92 pounds for both wing halves together.

Ailerons. The MK21 as currently envisioned is conventionally controlled by ailerons (x-axis), elevators (y-axis), rudders (z-axis), and spoilers (x and z axes). Each aileron is 450 inches long and is made up of an aluminum truss with the trailing edge being an aluminum sheet. Covering is doped fabric. The aileron main spar is 0.020 inch thick 3003H14 aluminum channel measuring 5.4 inches high by 0.600 inch wide. Ribs are formed sheet approximately 29 inches long by 5.4 inches high. Figure 4 shows details of aileron construction. All aluminum pieces have lightening holes varying from 3.75 inch diameter in the spar to an inch in the ribs.

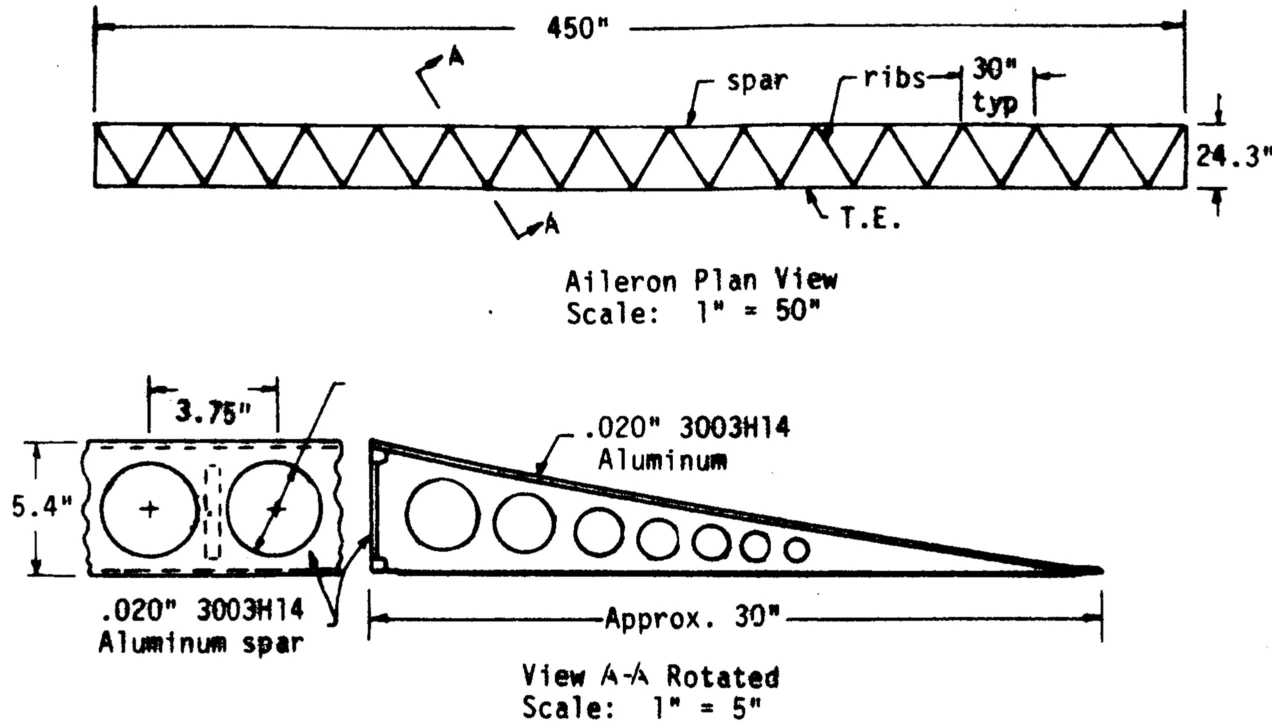


Figure 4. Aileron Structural Concept

The aileron spar will be formed from 6.6 inch wide sheet and will weigh 5.94 pounds without lightening holes or 4.13 pounds with 82 lightening holes of 3.75 inch diameter. The ribs will be formed from 29 inch long tapered blanks weighing 0.2262 pound each. With seven lightening holes tapered from 2.75 to 1.00 inches, this weight will be reduced to 0.1903 pound. For 32 ribs, this will be 6.09 pounds per aileron. The aileron trailing edge will be formed from the same material and will be identical in concept to the wing trailing edge. Weight will be 3.28 pounds for the trailing edge yielding a structural weight of 13.5 pounds per aileron. Covering is accounted for in wing weight.

Spoilers. The MK21 HAPP will use spoilers for added roll control and glide path control. Figure 5 presents details of spoiler construction with wood and foam as the primary materials for both the spoilers and their related structure. The spoiler front spar will be made from a 29 inch long x 0.38 inch wide x 1.00 inch high piece of spruce weighing 0.178 pound. The rear spar will measure 29 x 0.12 x 1.00 inches and will weigh 0.56 pound. Ribs

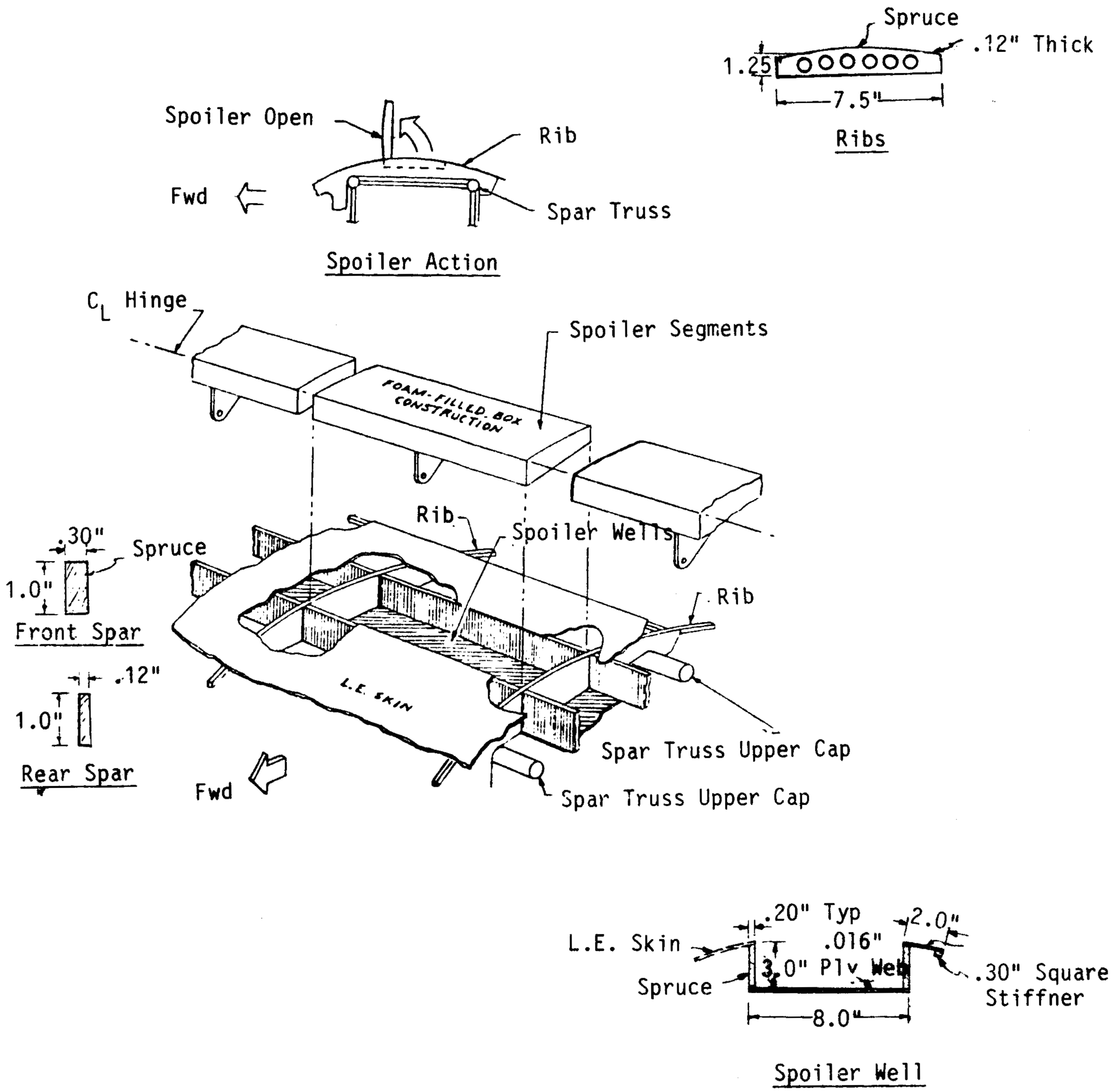


Figure 5. Spoiler Arrangement Concept

will be 7.5 inches long spruce x 0.12 inch thick and will weigh 0.018 pound each. A total of 5 will be required for a weight of 0.090 pound for each spoiler. Upper and lower skins will be 0.016 inch birch plywood weighing 0.211 pound. The foam is 2 pounds per cubic foot density, and 0.315 pound will be required for each spoiler. Total spoiler weight will be 0.98 pound. Associated control horns and hinges will boost this to 1.28 pounds. Six spoilers (one wing panel) will weigh 7.68 pounds.

Figure 5 also shows details of the spoiler wells made from birch and spruce. Total weight of well sides plus stiffeners is 0.775 pound per spoiler. Six wells would weigh 4.65 pounds. Total weight of spoilers plus wells for both wing halves is 24.66 pounds.

Tail booms. The MK21's load diagram, or V-n diagram, is presented in Figure 6. It has been recalculated from that shown in Ref. 5 in order to be in closer agreement with Part 23 of the Federal Aviation Regulations. The positive and negative limit loads are +2 and -1 g, respectively. The critical design conditions are the nighttime configuration at the low speed end and the daytime configuration at the high speed end since cruise speed varies during each 24 hour cycle. At sea level, in the nighttime configuration, the stall speed for a $C_{L_{MAX}}$ of +1.5 is 18 fps; the corresponding negative angle of attack (AOA) stall speed for a $C_{L_{MAX}}$ of -0.7 is 26 fps. The limiting high speeds are established as percentages of daytime and nighttime cruise speeds extrapolated from altitude by keeping cruise dynamic pressures constant. The salient corners for structural design purposes are:

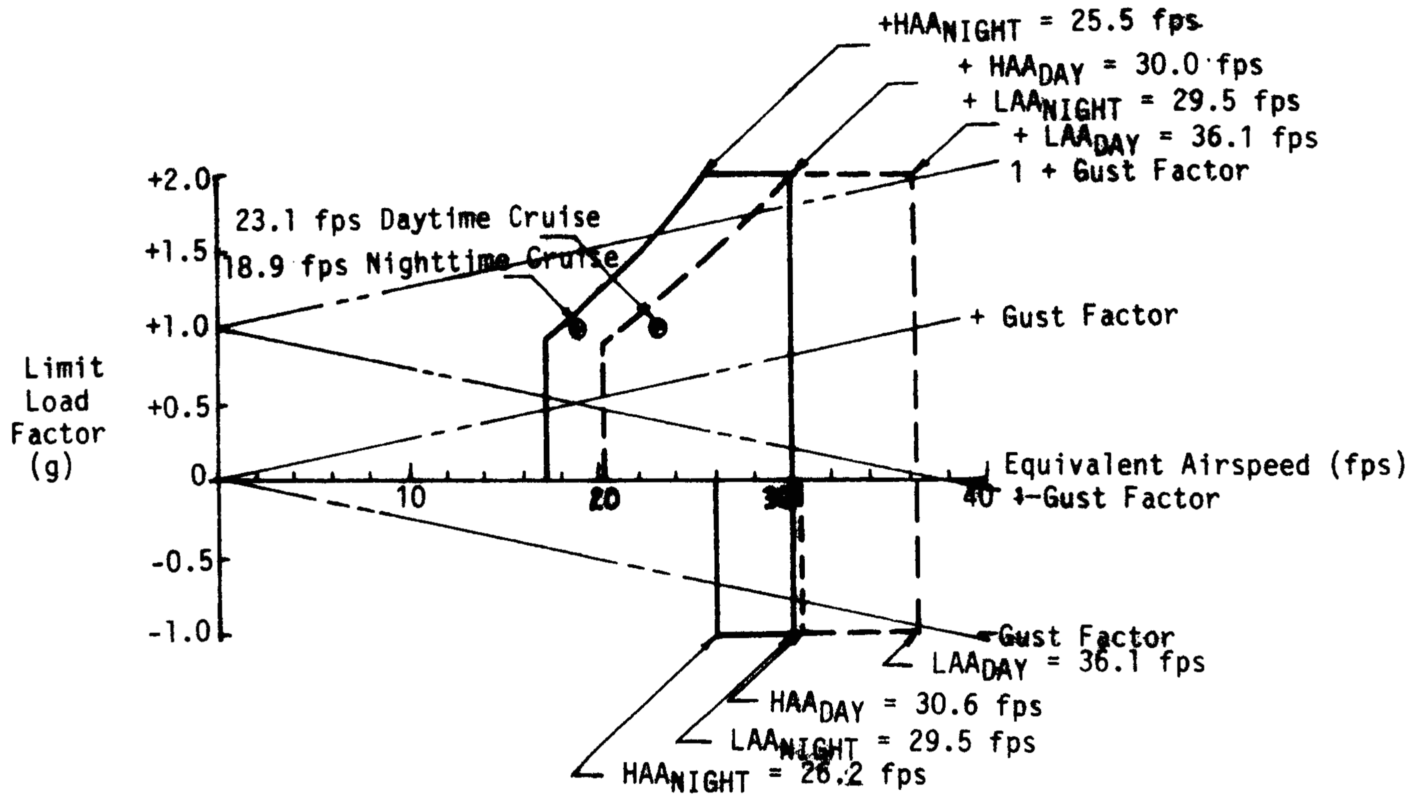


Figure 6. Velocity - Load Diagram for MK21 HAPP

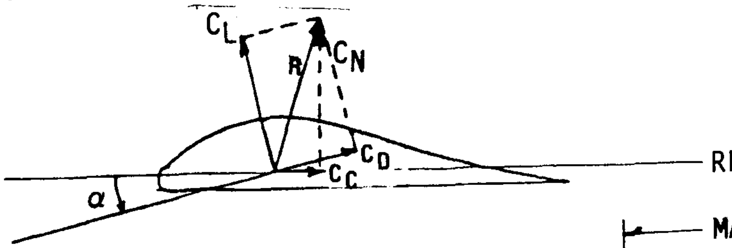
- Positive High Angle of Attack in the nighttime configuration ($+HAA_{NIGHT}$) of 25.5 fps at + 2g's;

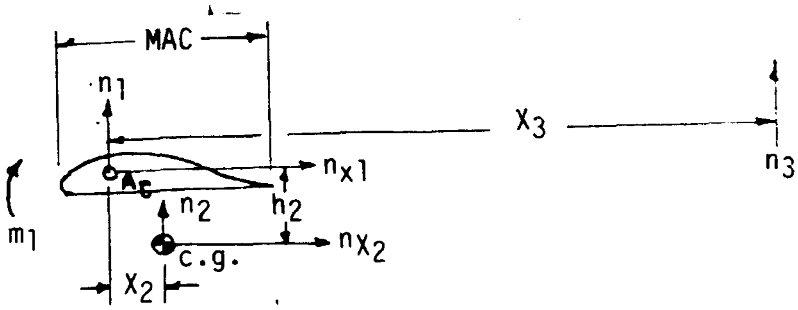
- Positive Low Angle of Attack in the daytime configuration ($+LAA_{DAY}$) of 36.1 fps at + 2g's;

- Negative High Angle of Attack in the nighttime configuration ($-HAA_{NIGHT}$) of 26.2 fps at -1g; and

- Negative Low Angle of Attack in the daytime configuration ($-LAA_{DAY}$) of 36.1 fps at -1g.

In order to size the tail boom structure it is first necessary to determine the gust loads which will be encountered by the horizontal and vertical tails. The illustration below defines the coordinate system used and shows forces and moments acting on both the wing and the horizontal tail. The forces acting on the horizontal tail may be resolved into normal (C_N) and chordwise (C_C) components, which are defined as:





$$C_N = C_D \sin\alpha + C_L \cos\alpha$$

$$C_C = C_D \cos\alpha - C_L \sin\alpha$$

A summary of pertinent data used is given in Table 1. The tail load factor,

TABLE 1

SUMMARY OF CALCULATED TAIL PARAMETERS

		FLIGHT CONDITION (NIGHTTIME)				FLIGHT CONDITION (DAYTIME)			
NO.	ITEM	+HAA	+LAA	-HAA	-LAA	+HAA	+LAA	-HAA	-LAA
1	w = Gross wt. - lbs.	1757.4	1757.4	1757.4	1757.4	1757.4	1757.4	1757.4	1757.4
2	v = Velocity - fps	34.8	36.09	28.7	36.09	34.8	36.09	34.8	36.09
3	q = .00119 V^2 = .00119x$(2)^2$	1.44	1.55	.98	1.55	1 .44	1.55	.98	1.55
4	s = w/s = 1/s	.57	.57	.57	.57	.80	.80	.80	.80
5	q/s = (3)/(4)	2.53	2.72	1.72	2.72	1.80	1.94	1.23	1.94
6	n_1 = Load factor (wing)	3.0	3.0	-1.5*	-1.5*	3.00	3.00	-1.5	-1.5
7	C_{N*} = (6)/(5)	1.19	1.10	-.87	-.55	1.67	1.55	-1.22	-.75
8	C_C = C_D cos - C_L sin	.0337	.0373	.0357	.0373	.0134	.0189	-.0309	.0189
9	n_{x1} = (8) x (5)	.0853	.1015	.0614	.1015	.0241	.0367	-.0380	.0367
10	C_m	-.03	-.03	-.03	-.03	-.03	-.03	-.03	-.03
11	m_1 = (10) x (5)	-.076	-.082	-.052	-.082	-.054	-.058	-.037	-.058
12	n_z = Tail load factor	.0311	.0287	-.0400	-.0485	.0396	.0377	-.0360	-.0396
13	n_2 = - (6) - (12)	-3.031	-3.02	1.540	1.549	-3.04	-3.04	1.54	1.54
14	n_{x2} = - (9)	-.085	-.1015	-.0614	-.1015	-.0241	-.0367	.0380	-.0367
15	T = (1) x (12) = lbs. tail load	54.66	50.44	-70.30	-85.23	69.59	66.25	-63.27	69.59
16	C_L	.395	.367	.581	.367	.555	.516	.816	.516
17	C_{DO}	.029	.0315	.0190	.0315	.020	.022	.010	.022
18	C_{Di}	.0018	.0015	.0038	.0015	.0035	.0030	.0075	0030
19	C_D	.0308	.0330	.0228	.0330	.0235	.0250	.0175	.0250
20	α, deg.	-.42	-.67	1.27	-.67	1.04	.68	3.40	.68
21	COS	1.000	.9999	.9998	.9999	.9998	.9999	.9982	.9999
22	SIN	-.0073	-.0117	.0222	-.0117	.0181	.0119	.0593	.0119
23	C_C	.0337	.0373	.0357	.0373	.0134	.0189	-.0309	.0189
24	C_N	0.394	0.3666	0.5814	0.3666				

*Conservative

n_3, may then be calculated using the solution of

$$n_3 = \frac{1}{X_3 - X_2}\left(m_1 - n_x h_2 + n_1 x_2\right)$$

Results are presented in Table 2.

TABLE 2
SUMMARY OF TAIL LOAD FACTORS
NIGHTTIME

(1) FLT. COND.	(2) X3-X2	(3) 1/(2)	(4) m1	(5) $n_{X1} h_2$	(6) $n_1 X_2$	(7) (4)-(5)+(6)	(8)=n_3 (3)x(7)
+HAA	5.83	.1715	-.076	.0427	.30	.181	.0311
+LAA	5.83	.1715	-.082	.0508	.30	.167	.0287
-HAA	5.83	.1715	-.052	.0307	-.15	-.233	-.0400
-LAA	5.83	.1715	-.082	.0508	-.15	-.283	-.0485

DAYTIME

(1) FLT. COND.	(2) X3-X2	(3) 1/(2)	(4) m1	(5) $n_{X1} h_2$	(6) $n_1 X_2$	(7) (4)-(5)+(6)	(8)=n_3 (3)x(7)
+HAA	5.54	.1807	-.054	.0121	.285	.2189	.0396
+LAA	5.54	.1807	-.058	.0184	.285	.2086	.0377
-HAA	5.54	.1807	-.037	.0190	-.143	-.199	-.0360
-LAA	5.54	.1807	-.058	.0184	-.143	-.2194	-.0396

Gust loads may be arrived at using the FAR Part 23.341

$$n = 1 \pm \frac{K_{GUST}\ U\ V_{a_N}}{498\ (w/s)}$$

where

$$K_{GUST} = \frac{2(w/s)}{\rho \bar{C}\ a_N g}$$

The wind studies shown in Appendix A of Ref. 5 yield a maximum gust at altitude of 3.9 mps, or 12.8 fps. Using this value for U in the equation above yields the gust envelope shown in Figure 6. The vertical tail gust load turns out to be the sizing criterion for the tailbooms given the high inertia of the vehicle directionally as opposed to pitch with wingtips up. Loads on the tailbooms are shown in Figure 7. Figure 8 summarizes the combined loads in one typical tailboom bay.

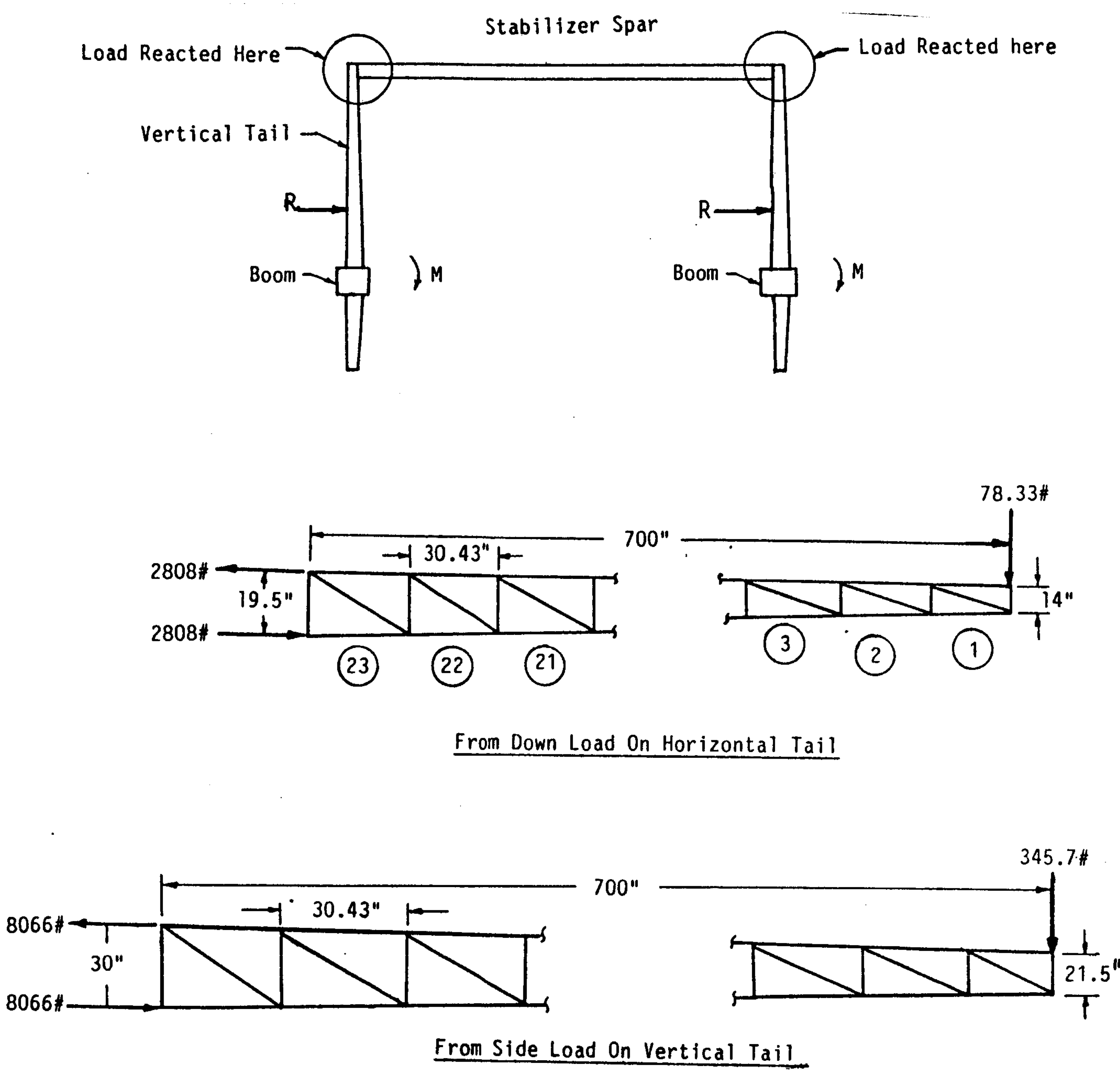


Figure 7. Critical Loads in One Tailboom

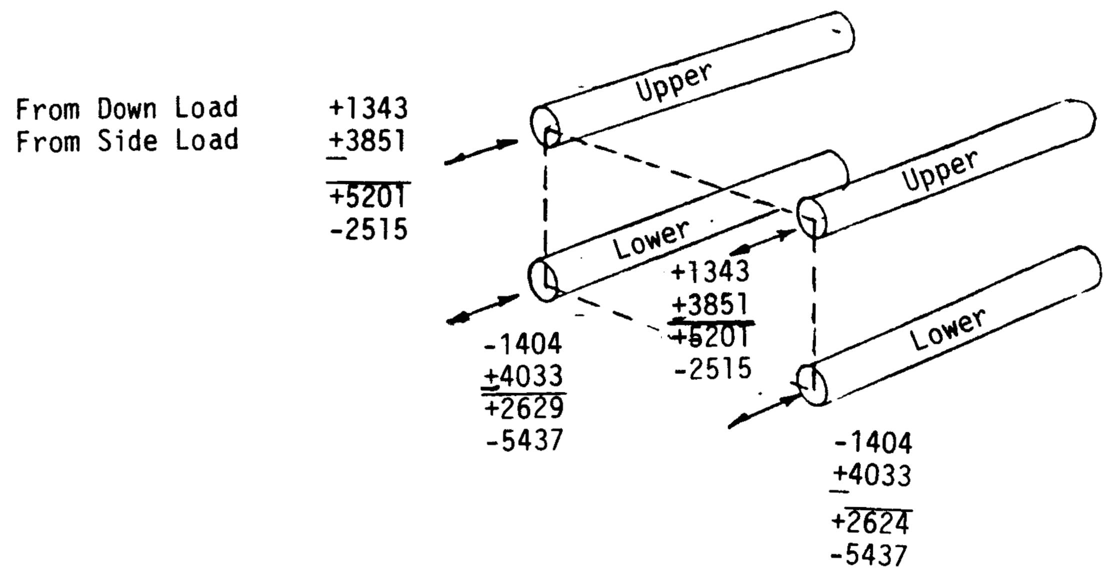
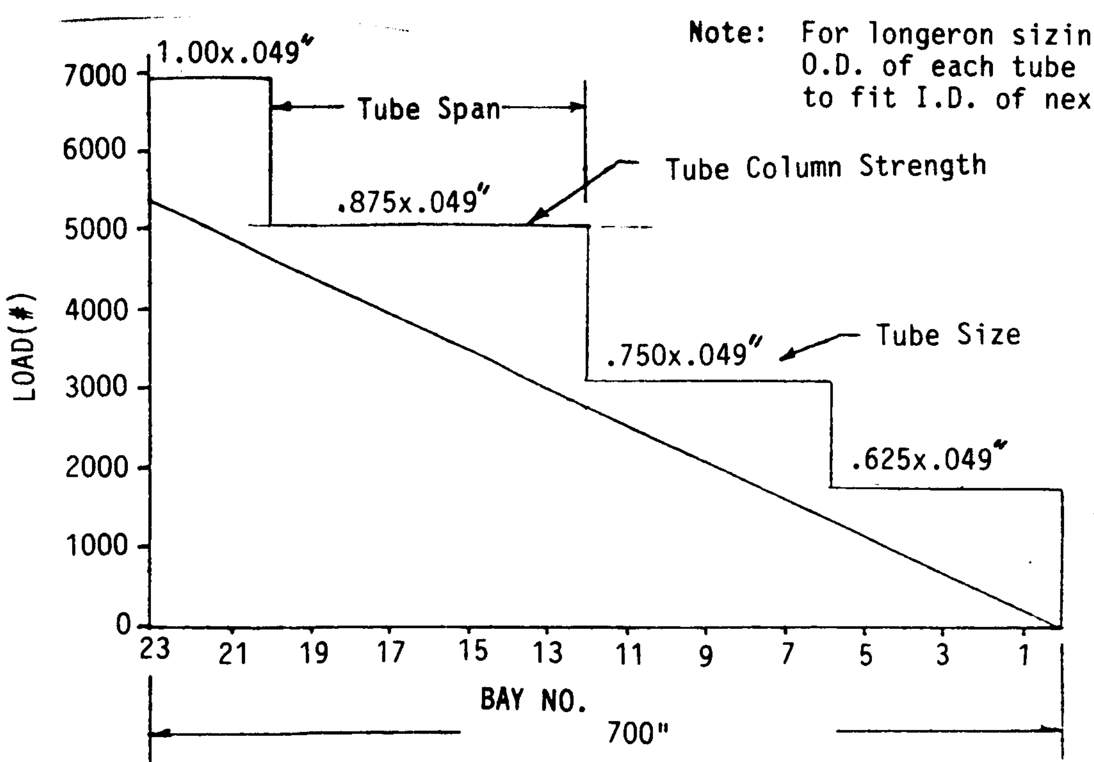


Figure 8. Summary of Loads in Tailboom

Table 3 presents longeron loads for each bay. The longeron tubes in all bays but 1, 2, and 3 may be treated as short columns. Tube sizes were then chosen to provide the lightest possible member to meet the net column loads (Figure 9).

TABLE 3. SUMMARY OF LONGERON LOADS IN TAILBOOMS

BAY NO.	UPPER LONGERONS		LOWER LONGERONS	
23	5201 lbs	-2515 lbs	2629 lbs	-5437 lbs
22	4964	-2400	2515	-5201
21	4728	-2286	2400	-4964
20	4492	-2172	2286	-4728
19	4256	-2058	2172	-4492
18	4019	-1943	2058	-4256
17	3783	-1829	1903	-4019
16	3546	-1714	1829	-3783
15	3310	-1600	1714	-3546
14	3074	-1486	1600	-3310
13	2837	-1371	1486	-3074
12	2601	-1257	1371	-2837
11	2360	-1148	1257	-2601
10	2129	-1029	1148	-2360
9	1892	-914	1029	-2129
8	1656	-800	914	-1892
7	1420	-686	800	-1656
6	1182	-572	686	-1420
5	946	-458	572	-1182
4	709	-343	458	-946
3	473	-229	343	-709
2	237	-115	229	-473
1	0	0	-115	-237



Note: For longeron sizing purposes, O.D. of each tube is designed to fit I.D. of next larger tube

Figure 9. Distribution of Longeron Sizes Along Length of Boom

Once longeron tube sizes had been determined, trusses could be sized to transfer net loads. The highest load in any truss member is 351 pounds. (It can be computed from the longeron loads shown in Figure 7.) The longest column is 42.73 inches. A half inch outside diameter (O.D.) tube of 0.049 inch wall thickness made of graphite epoxy will provide adequate margin of safety. Boom weights were estimated and the results are presented in Table 4.

TABLE 4. SUMMARY OF TAILBOOM COMPONENT WEIGHTS

Item	Number	Area	Volume	Weight
SIDE TRUSS				
Upper & Lower Longerons				
1.00x0.049x95	2	0.1464in^2	27.8in^3	1.70#
0.875x0.049x250	2	0.1272	63.6	3.88
0.750x0.049x192	2	0.1079	41.4	2.53
0.625x0.049x183	2	0.0887	32.5	1.98
Verticals				
0.500x0.049x16.75(avg)	23			1.63
Diagonals				
0.500x0.049x34.84(avg)	23			3.39
Top & Bottom Trusses				
Cross Members				
0.500x0.049x25.75(avg)	23			2.51
Diagonals				
0.500x0.049x39.98(avg)	23			3.9
TOTAL WT OF 1 SIDE OF BOOM TRUSS				21.52#
TOTAL WEIGHT OF BOTH SIDES OF BOOM TRUSS				43.04
JOINTS & ADHESIVES (15%)				6.43
TOTAL WEIGHT OF 1 BOOM TRUSS				49.47#
WOODEN STRINGERS 0.25x0.50x700		0.125	350	5.67#
WOODEN STRINGERS ADHESIVES (15%)				0.85
TOTAL WT OF STRINGERS & ADHESIVES FOR 1 BOOM				6.52#
FABRIC & DOPE		413ft^2		8.03#
TOTAL WEIGHT OF 1 TAILBOOM				64.02#
TOTAL WEIGHT OF BOTH TAILBOOMS				128.04#

Vertical Tail Design. The areas of both the horizontal and vertical tail surfaces were kept constant from the MK20 to the MK21 as were tail volumes to maintain static stability. Figure 10 presents details of the vertical fin design. The ultimate load shown is a fraction of the total fin load of 346 pounds. This translates to a tension load in one fin spar truss of 586 pounds and a compression load in the other of 848 pounds.

The airfoil chosen for the vertical fin is a NASA 63_2-015. Two alternate construction techniques for ribs were examined. The first, shown in Figure 10 (center, left), is a rib of aluminum weighing 2.63 pounds (with lightening holes) for 6 ribs. The second is shown in Figure 10 (center, right) and is made of spruce and birch plywood. It weighs 2.10 pounds for 6 ribs, or 21% lighter than the aluminum rib. See Appendix B for a further discussion of lightweight building materials. The fin leading edge is a 0.625 x 0.028 wall x 155 inch graphite epoxy tube. The fin shape is maintained with doped fabric covering. The rudder and trailing edge are made similarly to the ailerons. Table 5 summarizes vertical tail weights.

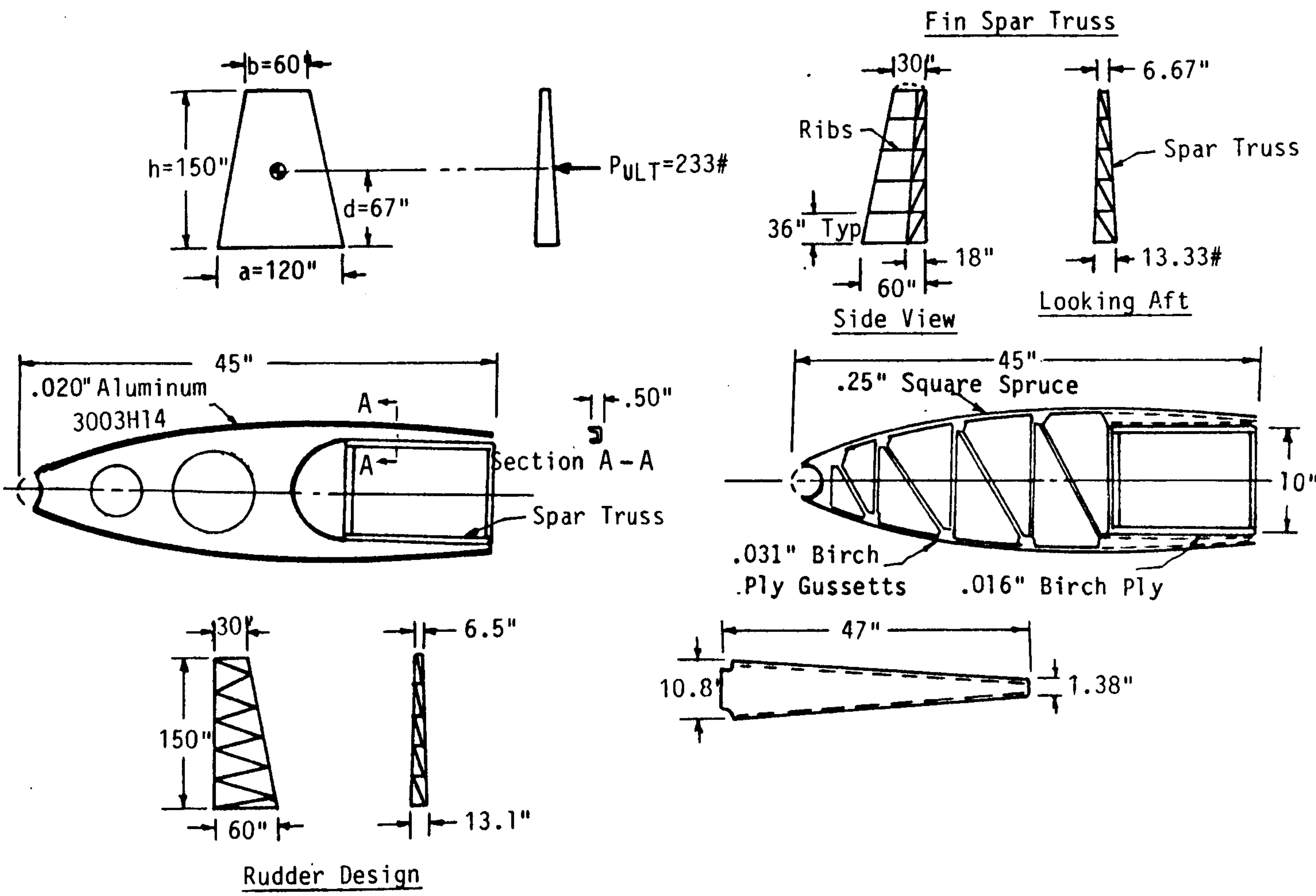


Figure 10. Vertical Tail Design

TABLE 5. SUMMARY OF VERTICAL TAIL WEIGHTS

ITEM	NUMBER	AVERAGE LENGTH	TOTAL LENGTHS	AREA	VOLUME	WEIGHT
FIN SPAR TRUSS						
Lower Caps	4	77in.	308in.		$20in^3$	1.219#
Upper Caps	4	75	300		15.33	0.935
CHORD MEMBERS	12	13.5	162		8.28	0.505
CHORD DIAGONALS	10	32	320		16.35	0.998
CROSS MEMBERS	12	10	120		6.13	0.374
CROSS DIAGONALS	10	32	320		16.35	1.000
JOINTS & ADHESIVES						0.75
WT OF 1 FIN TRUSS						5.78#
RIBS (SPRUCE & BIRCH)				$1.53ft^2$		
Caps			94	$0.063in^2$		0.095
Verticals			48	0.063		0.049
Diagonals			50	0.063		0.051
Chord Members			28	0.063		0.028
0.031 Plywood			--	42		0.041
0.016 Plywood			--	84		0.041
RIB WEIGHT						0.304#
ADHESIVE (15%)						0.046
TOTAL RIB WEIGHT						0.350#
TOTAL WT OF 6 RIBS FOR 1 VERTICAL						2.10 #
FIN LEADING EDGE WT						0.496#
FABRIC COVERING & DOPE				$95ft^2$		1.847#
VERTICAL FIN (similar construction)				45		9.86 #
RUDDER						
Caps	2	150.0	300			0.961
Cross Members	6	9.8	59			0.189
Diagonals	5	31.0	155			0.496
Ribs	12	47.0		$286in^2$		5.84
Joints & Adhv(15%)						0.247
Trailing Edge			12.83ft			1.12
Fabric & Dope				$93ft^2$		1.81
TOTAL WT OF 1 VERTICAL FIN						31.81#
TOTAL WT OF BOTH VERTICAL FINS						63.62#

<u>Horizontal Tail.</u> The horizontal tail is structurally analogous to the vertical but is constrained and loaded differently. If a factor of 2 is applied to account for this difference, then the horizontal will weigh approximately 19.00 pounds.

Fuselage Pod. The fuselage pod shown in the general arrangement is there to enclose power train and payload items and may not be necessary on all versions of solar HAPPs. The main sizing load is ground impact at a 15^{0} nose down angle. Figure 11 presents fuselage pod load and construction details. The

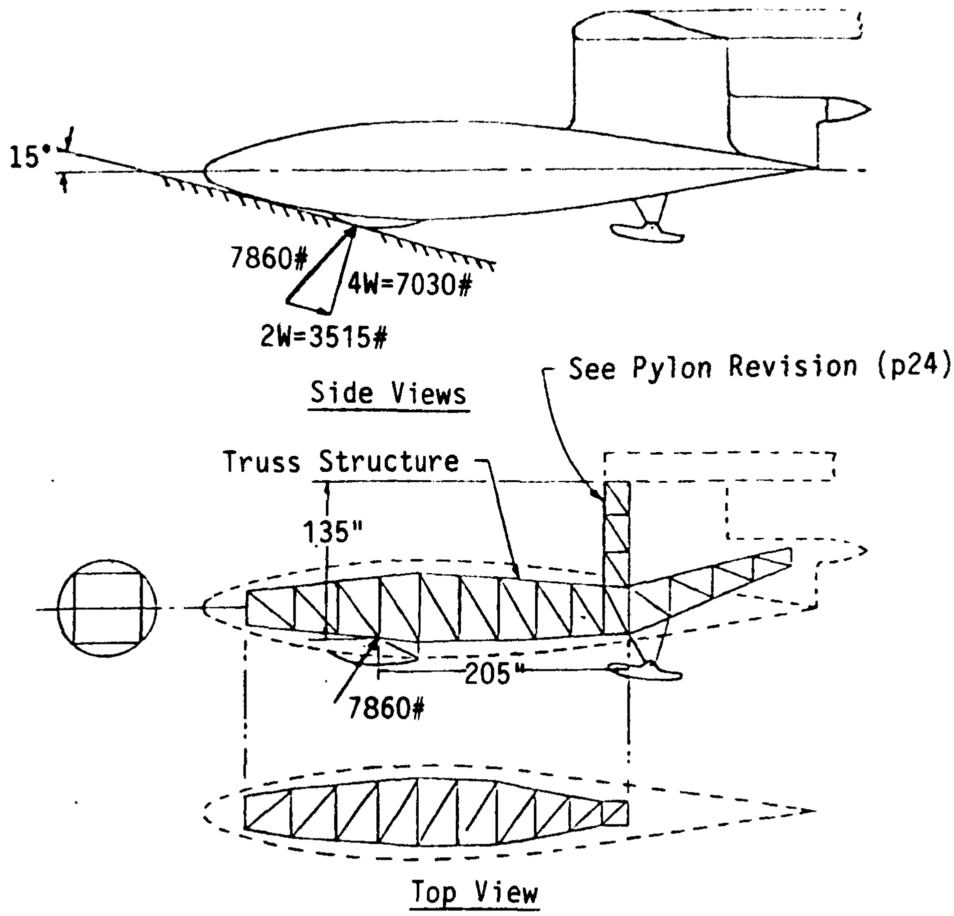


Figure 11. Fuselage Pod Load and Construction Details

trusses in the pod may be broken into 3 sections. The forward section carries negligible loads and, hence, can be made out of the lightest practical size tubes for manufacturing and handling, 0.500 inch O.D. by 0.028 inch thick.

The mid-section will carry a maximum load of 6100 pounds in compression. The smallest size tube available to handle this, 1.25 x 0.035 wall x 33, will handle almost 7500 pounds, so the structure will be somewhat overdesigned in this section. Lower longerons must handle a 2600 pound tension load. A tube size of 1.25 x 0.028 wall will be used to facilitate joining to other truss members. Verticals will be 1.25 x 0.035 x 52 inches and will carry a compression load of 3600 pounds.

The aft section will absorb a 14,100 pound compression load and will be 1.62 x 0.049 wall x 30 inches. Lower longerons will be 1.62 x 0.028 wall for consistency of construction with vertical pieces which are 1.62 x 0.028 wall x 40 inches. Diagonals will all be in tension with the maximum tension load being 5600 pounds. Tube sizes of 0.500 x 0.035 wall will be adequate to handle this with the exception of one diagonal side brace, which has a 21000 pound tension load and must, therefore, be 1.25 x 0.049 wall tube.

Pod upper and lower trusses will be similarly sized since the landing load is expected to be the worst case load.

The pod fairing will be made up of spruce, birch plywood, fiberglass and doped fabric as shown in Figure 12. Both nose and tail fairings will be fiberglass. The 12 spruce fairing strips will be 0.25 x 0.80 x 385 inches and the 52 supports will be 0.25 x 0.25 x 70 inches. Birch plywood will be 0.031 inch thick and each support will be approximately 0.59 square foot. Including joints and adhesives, total weight of fairing strips and supports will be 20 pounds. Fabric and dope will add 9.76 pounds. Figure 12, bottom, presents drawings of the nose cone and tail cone. Surface area of the nose cone is 21.92 square feet and the tail cone is 47.91 square feet for weights of 3 pounds and 8.13 pounds, respectively.

The landing skid is also a part of the fuselage pod. It will weigh roughly the same as a typical sailplane landing gear, or 27 pounds (0.15W) .

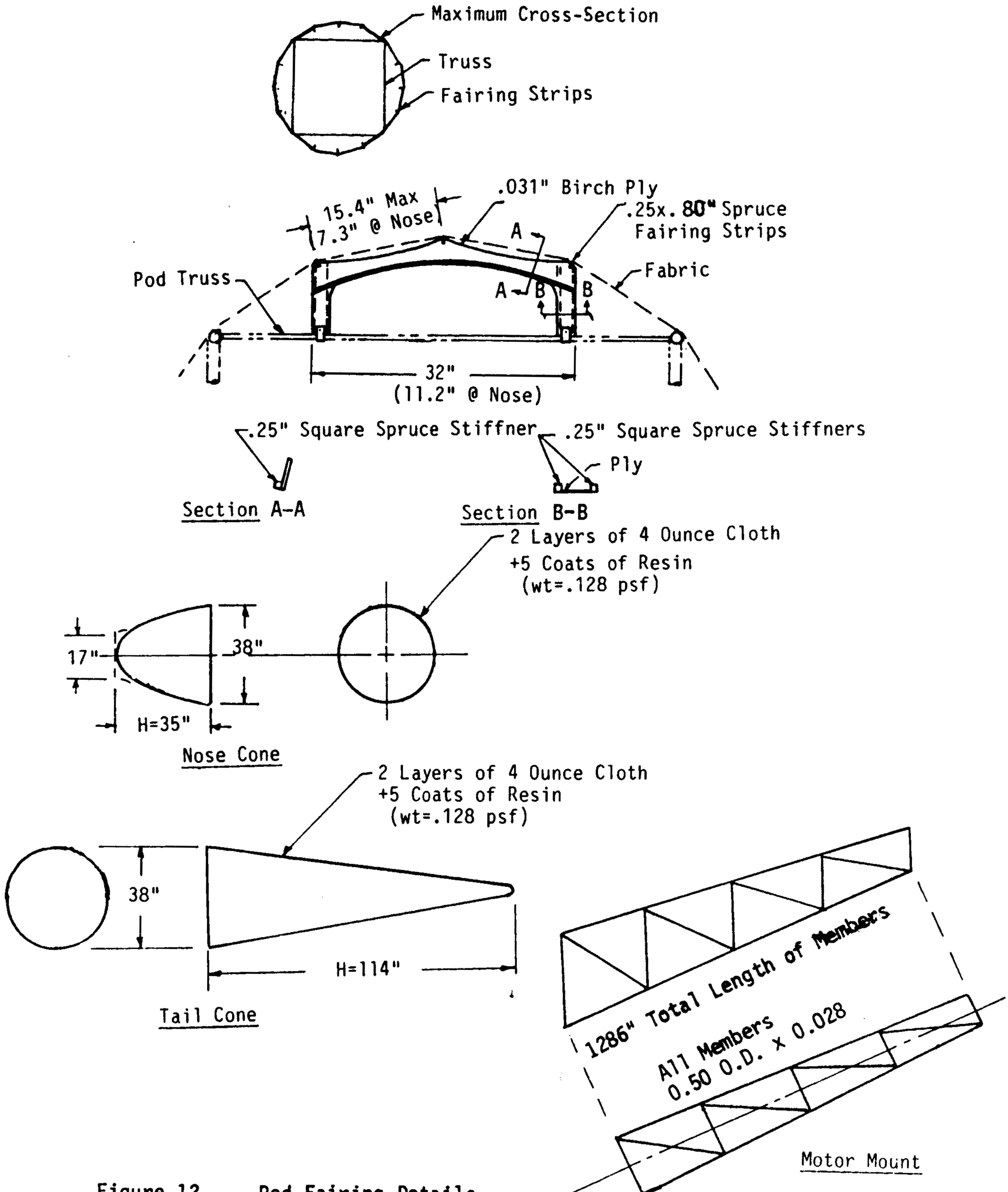
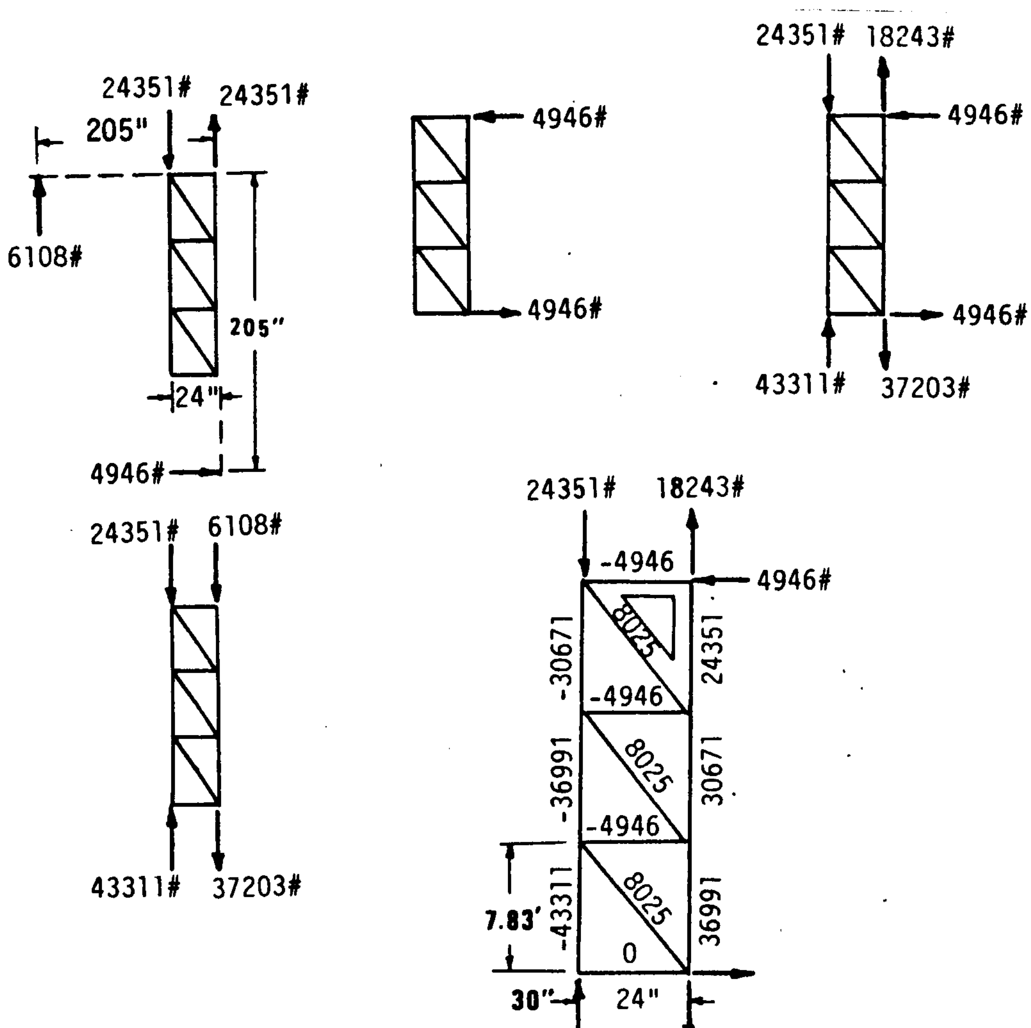


Figure 12. Pod Fairing Details

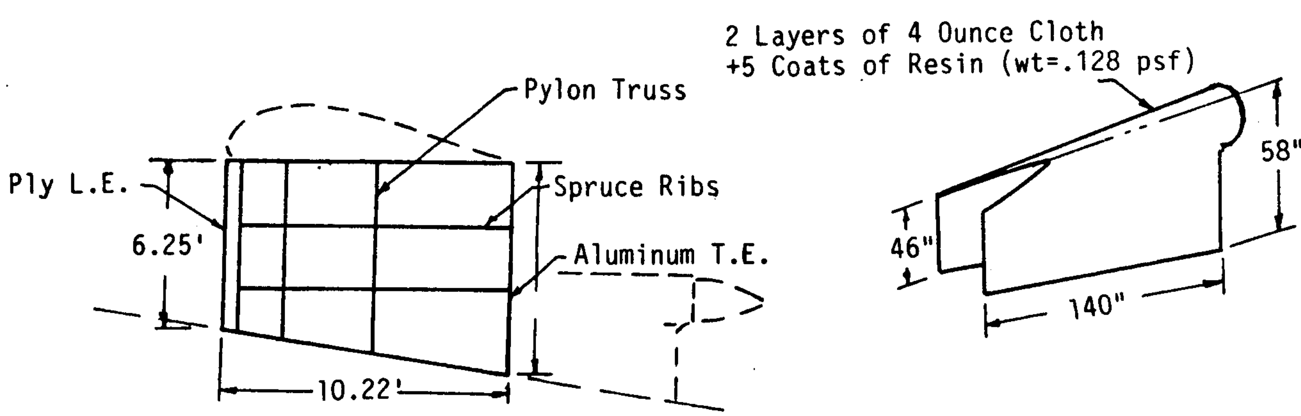
The motor mount is included in the fuselage pod weight. It is a cantilevered truss of 0.500 x 0.028 wall members of 1286 inches length. If this is weighed the same as other structure examined so far with 15% for adhesives, then it will weigh 3.74 pounds. A summary of fuselage pod weight, then, is

Main tubular truss	29.64#
Fairing strips and supports	20.00
Fabric & dope	9.76
Nose cone	3.00
Tail cone	8.13
Landing skid	27.00
Motor mount	3.74
TOTAL	101.27#

Pod Support Pylon. The fuselage pod is attached to the wing by a support pylon which is an aerodynamic fairing around a tubular truss. Figure 13 presents details of the structure envisioned for the pylon and motor fairing as well as critical loads encountered in the 15^{0} nose-down landing case. Given the loads shown in Figure 13, it is possible to estimate tube sizes. The forward caps will experience a 21,656 pound compressive load which can be handled by tubes 1.62 x 0.065 wall x 30.67. Aft caps will experience an 18,496 pound tension load, so 1.62 x 0.028 wall will be used. Chordwise pylon tubes will have 2473 pound compressive loads which can be handled by 0.62 x 0.049 wall x 24 inch tubes. Diagonals will have 4013 pounds in tension and 0.62 x 0.028 wall will be used. Figure 14 presents a summary of tube sizes and shows the revised pylon truss structure envisioned for the MK21. Weights will be:



Loads in Truss Members



Design of Pod Fairing

Figure 13. Pod Support Pylon Details

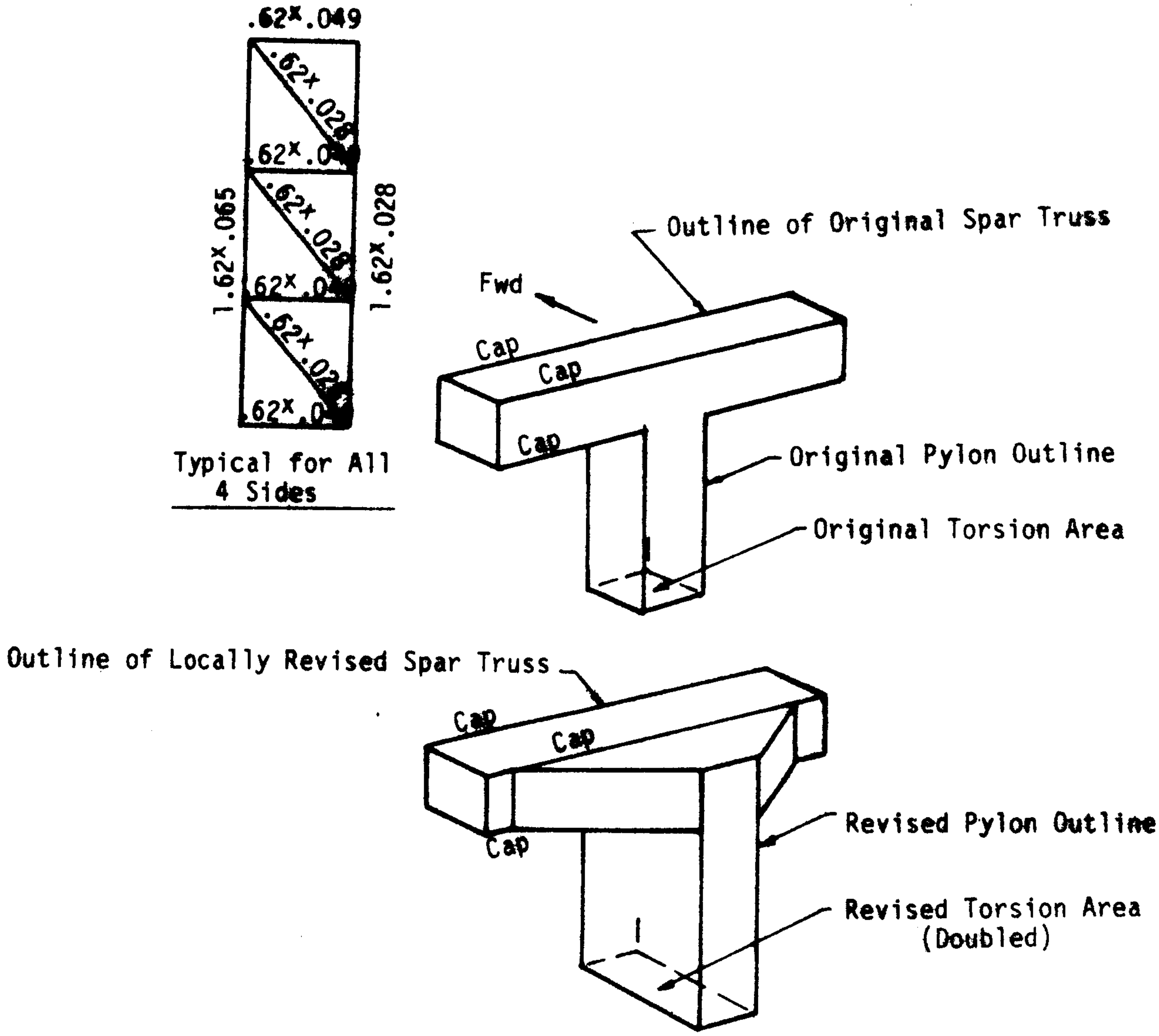


Figure 14. Pylon Tube Size Summary

	ITEM	TOTAL LENGTH	AREA	WEIGHT
Forward Caps	1.62x0.065in	184in	$0.3186 in^2$	3.576#
Aft	1.62x0.028	184	0.1405	1.377
Chordwise members	0.62x0.049	192	0.0887	1.039
Spanwise members	0.62x0.049	192	0.0887	1.039
Diagonals	0.62x0.028	468	0.1050	1.498
Joints & Adhesives (15%)				1.309
TOTAL				9.838#

The pod fairing will have a birch plywood leading edge, spruce ribs, and a trailing edge similar to the ailerons with covering being doped fabric. Applying the same unit weights as comparable wing parts, pod fairing weights are:

ITEM	UNIT WEIGHT	WEIGHT
Leading edge	0.318 #/ft.	1.99#
Spruce ribs	1.31 #	5.24
Trailing edge	1.4 oz/ft	0.69
Fabric & dope	0.01944 psf	2.79
Adhesives		1.07
TOTAL		11.78#

The fiberglass motor fairing will be made up of 2 plies of 4 ounce cloth and 5 coats of resin. Total area is 116 square feet, and weight is 18 pounds.

Summary of Non-Wing Spar Weights

The various parts of the MK21 which have been discussed so far were left constant as wing design was changed to evaluate the effect of bracing concept on wing weight. These parts may be summarized, as below:

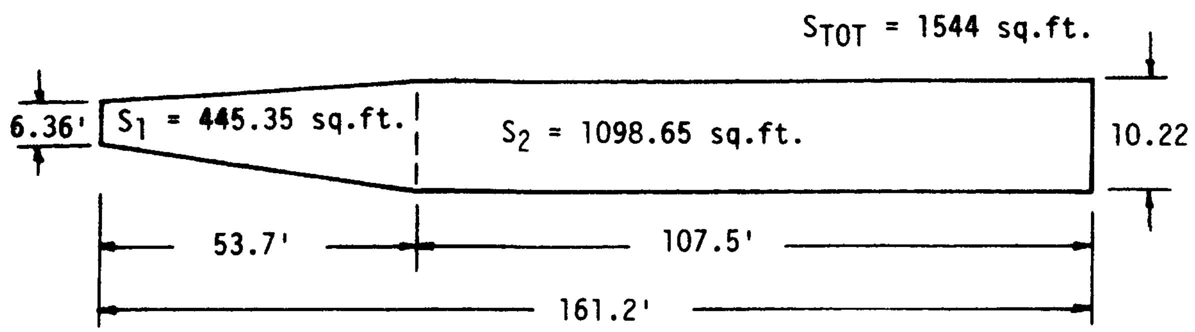
ITEM	WEIGHT	FRACTION OF TOGW
Wing leading edges	102.42#	0.0583
Wing trailing edges	21.72	0.0124
Wing ribs	159.92	0.0910
Ailerons	27.00	0.0154
Spoilers & wells	24.66	0.0140
Tail booms	128.04	0.0729
Vertical fins & rudders	63.62	0.0362
Horizontal tail	19.00	0.0108
Fuselage pod	74.27	0.0423
Landing skid	27.00	0.0154
Pod support pylon	21.62	0.0123
TOTAL	669.27#	0.3808

Bracing Schemes Analyzed

Strut-Braced Wing. Several assumptions have been made to begin design of these alternate wing concepts and they are:

- Wing loading is uniform across the span;
- No tip losses;
- Design load factor is $\pm$ 3.0;
- Vehicle gross weight remains constant at 1757.4 pounds (797Kg); and
- Vehicle wing area and planform remain constant at 3088 square feet (287 square meters)

The wing planform to be used is shown below for one wing half.

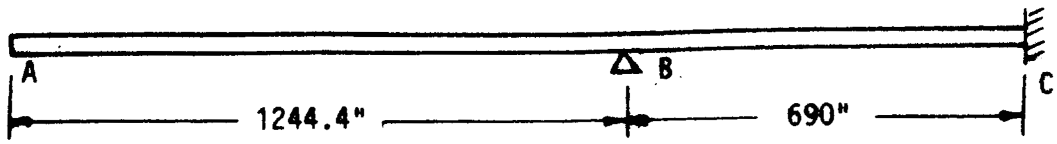


The lift load per panel is 2636 pounds and this is arrived at by applying the design load factor to half the gross weight. Wing dead weight items may be approximated by multiplying the wing panel area by a factor of 0.164 psf which was arrived at in earlier LMSC studies. Add to this the following items:

- Fixed solar panel of 283 square feet, weighing about 170 pounds including solar cells on the panel; and

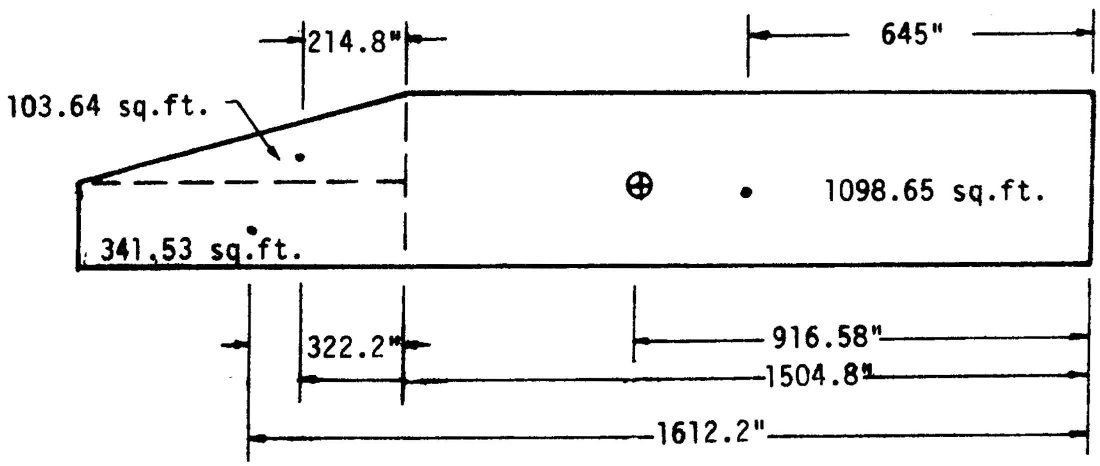
- Movable wingtip and solar cells weighing about 130 pounds.

Total dead weight per side is 1010 pounds. This is a starting point for purposes of load calculation and will be refined as the analysis continues. The lift load may be expressed in terms of a running load in the spar of 1.46 pounds per inch. This will be taken out by the support scheme shown in Figure 15. Figure 16 presents the lift reactions and calculation of the load center of gravity. Similarly, the dead weight items create a running load in the spar and have reactions at the joints shown in Figure 17. Lift and dead weight shears on both tapered and constant chord sections may then be calculated, and the net reactions are presented in Figure 18.



A is free
B is simply supported
C is fixed

Figure 15. Free-Body Representation of Strut-Braced Wing Spar



$$X_{CG} = \frac{[1098.65 \times 645] + [341.53 \times 1612.2] + [103.64 \times 1504.8]}{1544}$$

$$X_{CG} = 916.58"$$

Figure 16. Determination of Load Center of Gravity For Strut-Braced Wing

Wing bending moments from both lift and dead weight may be calculated, Figure 19 presenting the results. The strut attaches to the wing at wing station (W.S.) 690.0 and the resultant bending moment transferred there is 606,262 inch-pounds. The strut also induces an axial load in the spar of 11,488 pounds. If the inboard section of the wing spar is assumed fixed at both

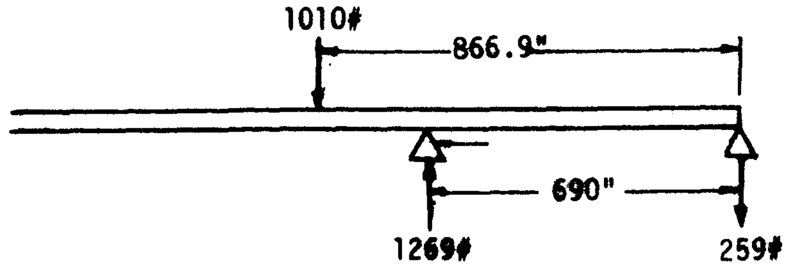


Figure 17. Reactions in Main Spar From Dead Weight Items

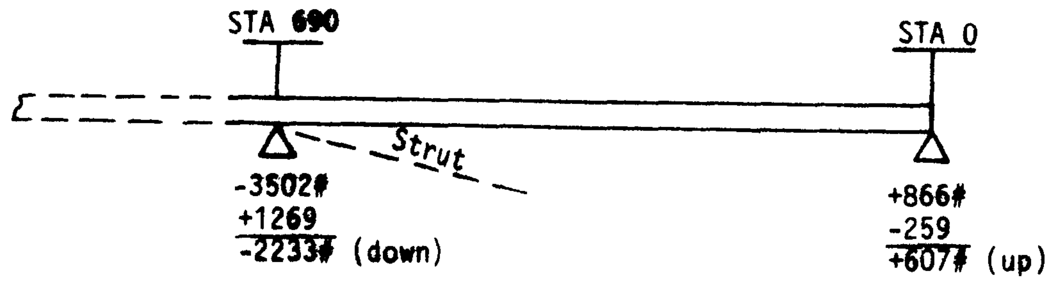


Figure 18. Main Spar Net Running Load Reactions

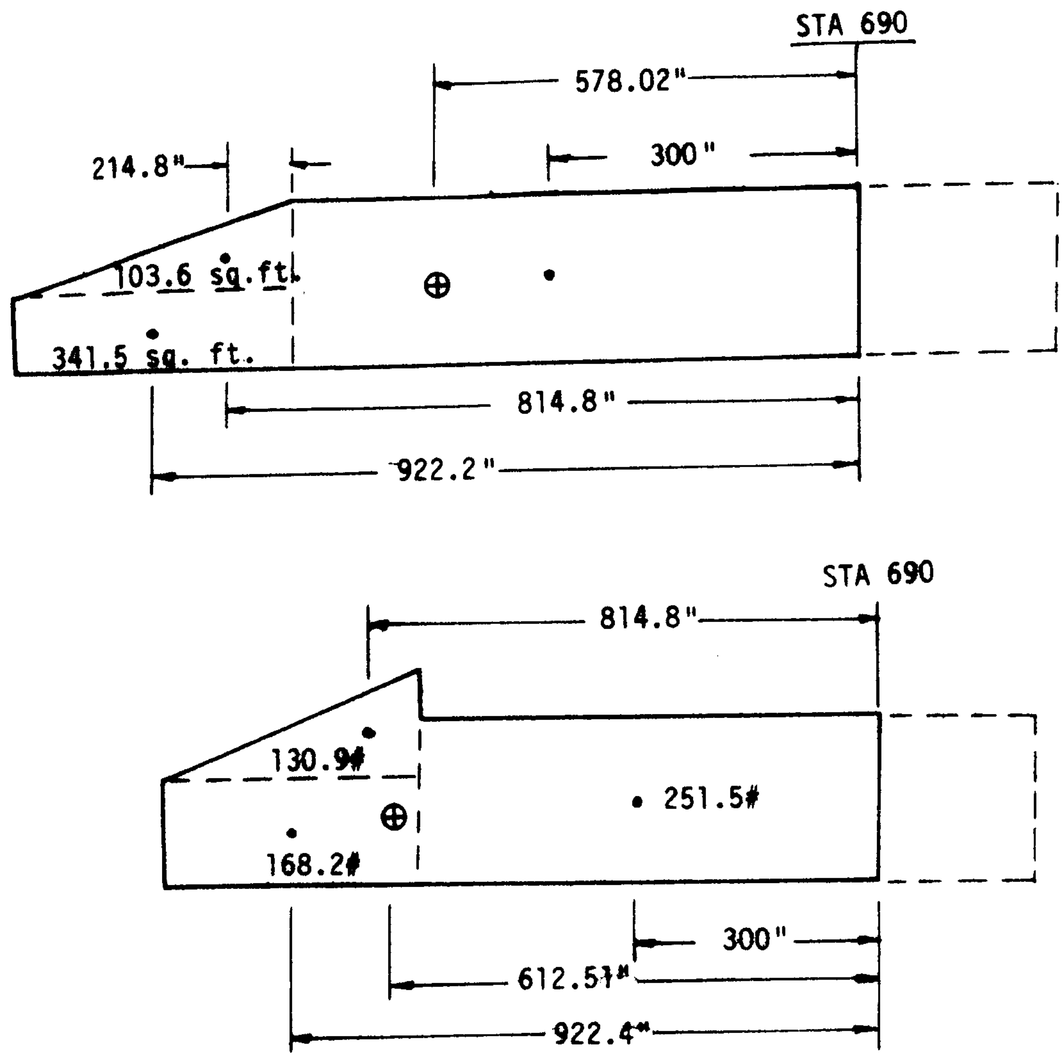


Figure 19. Bending Moments in Main Spar

ends, then the bending moment inboard of the wing strut may be calculated as shown in Figure 20. Peery's method (see Ref. 7, pg. 355) then yields a bending moment as follows:

$$J = \left(\frac{EI}{P}\right)^{1/2} = \left(\frac{3.342\text{x}10^9}{11488}\right)^{1/2} = 539.4$$

$$\frac{L}{J} = \frac{690}{539.4} = 1.28; \quad C_1 = 11.6$$

$$M = \frac{W\,L^2}{C_1} = \frac{0.788\text{x}(690)^2}{11.6} = 32{,}342\text{in\#}$$

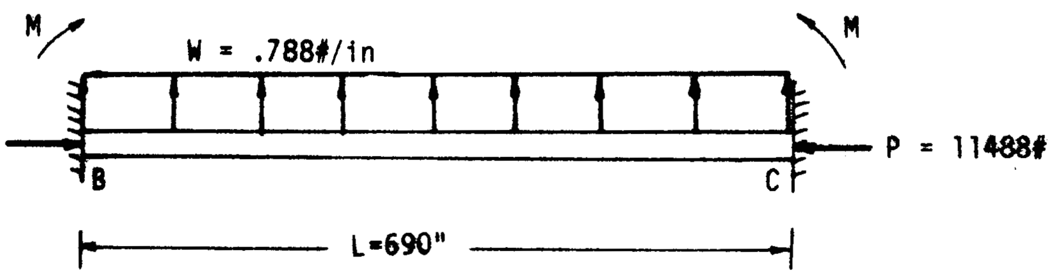


Figure 20. Bending Moment Inboard of Strut

Finally, the moment distribution may be expressed below recalling that A is the wingtip, B is the strut attachment point, and C the left wing/right wing interface. All units are inch-pounds.

		B	C
Initial Moments	+606,202	-32,342	+32,342
Balance, Joint B		-573,920 →	-286,960
Final Moments	+606,207	-606,202	-254,618

The product EI, known as bending stiffness may be calculated for the spar using a value of Young's modulus, E, arrived at in previous work of 30×10^6 psi. Figure 21 presents the spar cross-section to be analyzed.

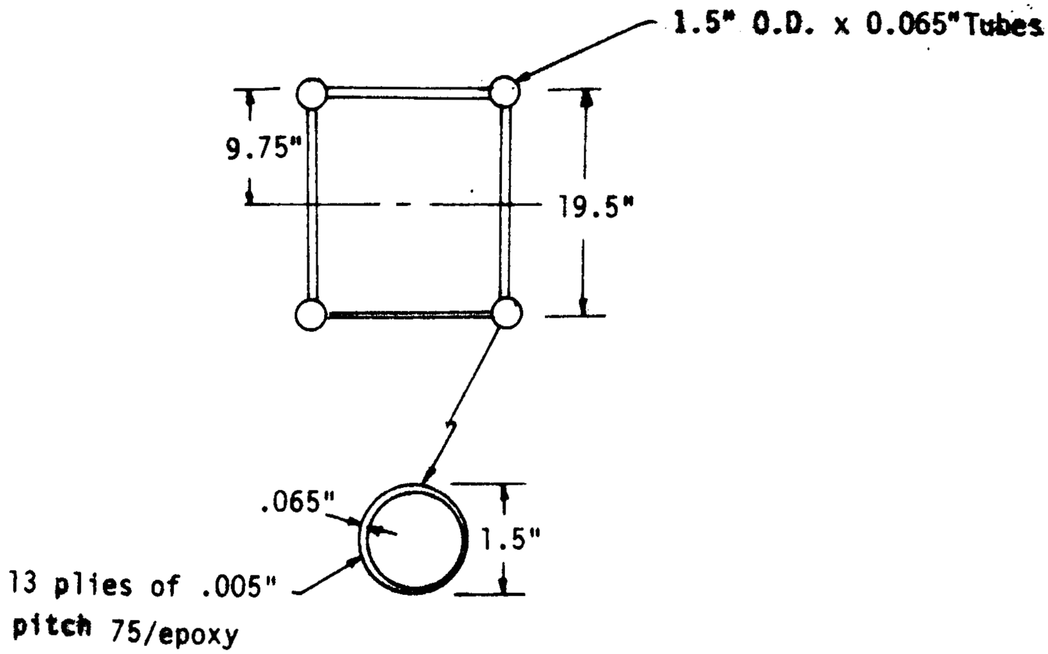
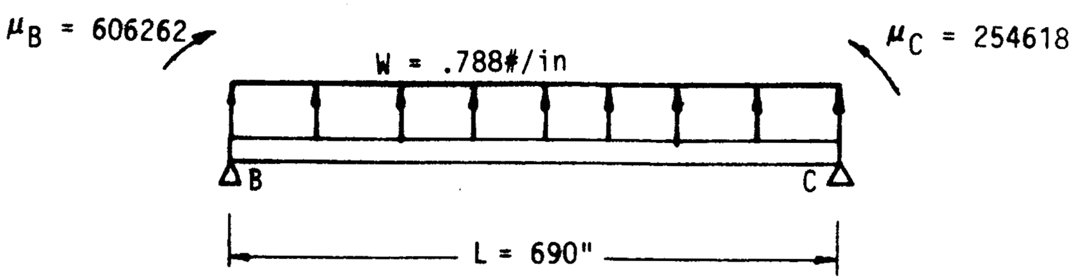


Figure 21. Spar Cross-Section

Continuing with calculation of reactions at the points of support in the strut-braced wing, the reaction to the 1082 pound net load in the wing spar outboard of WS 690 will be a downward shear at WS 690 of equal magnitude. Inboard of the wing strut, the shear and bending moment reactions may be arrived at as follows:



$$R_B = \frac{\mu_C - \mu_B}{L} - \frac{WL}{2}$$

$$= \frac{-254618 - 606262}{690} - \frac{.788 \times 690}{2}$$

$$= -1248 - 272$$

$$= \underline{-1520 \text{ lbs}}$$

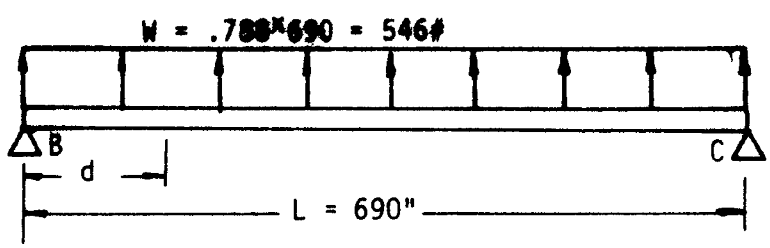
$$R_C = \frac{\mu_B - \mu_C}{L} - \frac{WL}{2}$$

$$= \frac{606262 - (-254618)}{690} - \frac{.788 \times 690}{2}$$

$$= 1248 - 272$$

$$= \underline{976 \text{ lbs.}}$$

Similarly, free bending moments for this section of the spar may be found as follows:



$$M = .5\ W\left(d - \frac{d^2}{L}\right)$$

$$M = .5 \times 544 \left(d - \frac{d^2}{690}\right) = 272 \quad d - \frac{d^2}{690}$$

d	d^2	d^2/L	$d-d^2/690$	M
0	0	0	0	0
138	19,044	27.60	110.4	30,029 in. lbs.
276	76,176	110.4	165.6	45,043
414	171,396	248.4	165.6	45,043
552	304,704	441.6	110.4	30,029
690	476,100	690.0	0	0

Figure 22 summarizes the wing normal shear load distribution and Figure 23 summarizes the wing normal shear bending moment distribution for this strut-braced wing.

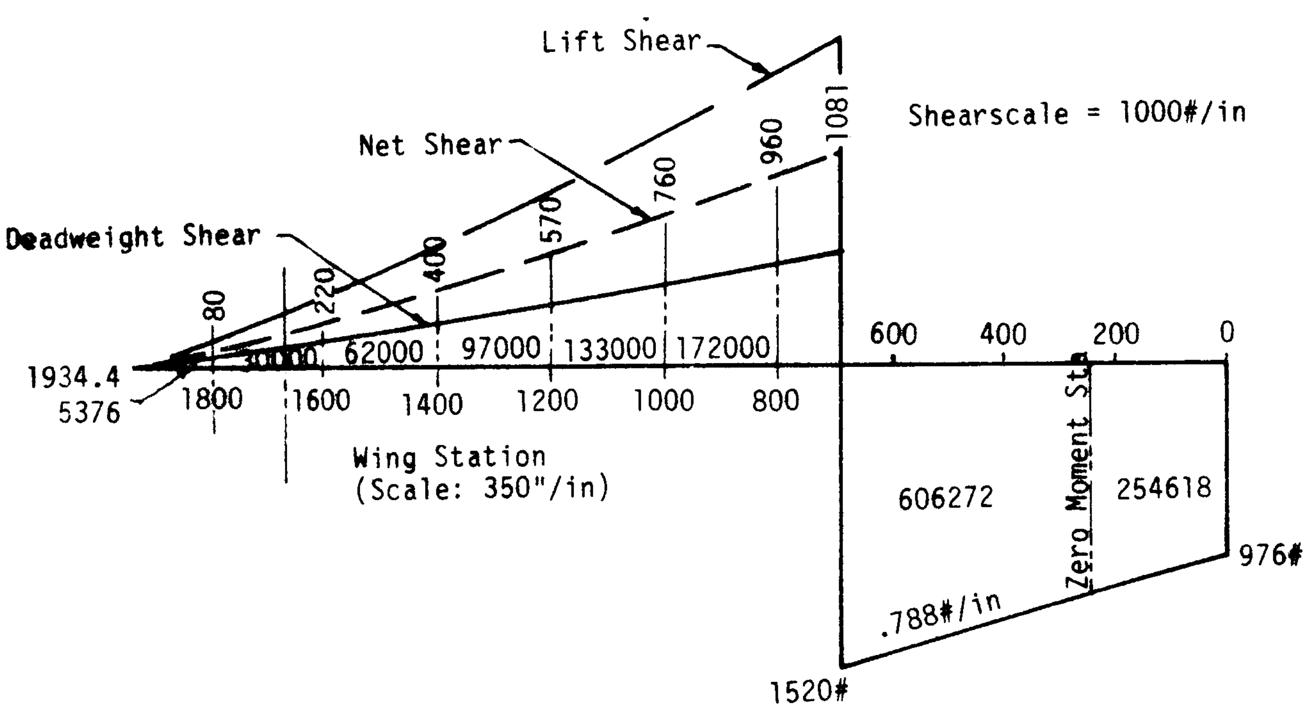


Figure 22. Wing Normal Shear Load Diagram at Ultimate Load Factor

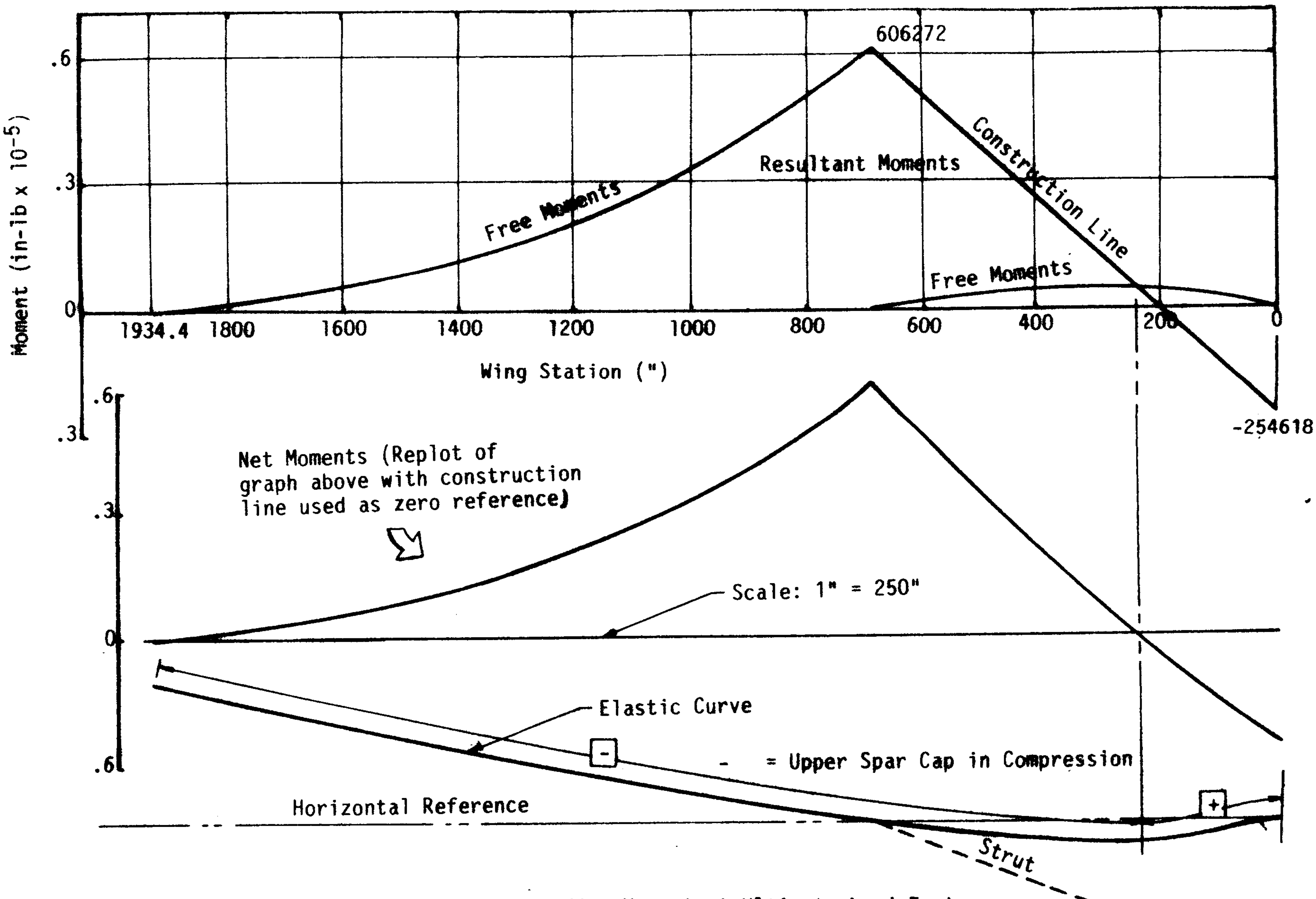


Figure 23. **Normal Bending Moment at Ultimate Load Factor**

Wing chord loads may now be calculated with the basic assumptions that:

- Maximum chord load will occur at maximum lift coefficient ($C_{L_{MAX}}$) and act forward;

- Maximum realistic $C_{L_{MAX}}$ is 1.6.

Defining chord forces are below:

The chord load, $d_1 + d_2$, will be determined as

$$d_1 + d_2 = 2636 \sin\alpha + C_{DW} \; q \; S_{REF} \cos\alpha$$

If angle of attack at C_L MAX is estimated by

$$\alpha_{MAX} = \frac{C_{L_{MAX}}}{C_{L_\alpha}} + \alpha_{OL} + j\epsilon$$

the zero-lift angle of attack is given in Ref. 6 as -4 degrees, and j is identically zero for an untwisted wing. Wing lift-curve slope, then, is

$$C_{L_\alpha} = \frac{f \; a_e}{1 + \left(\frac{57.3 a_e}{\pi AR}\right)}$$

Abbott & VonDoenhoff (Ref 12) define f as 0.99 for a wing of this type. If section lift curve slope, a_0, is 0.12 per degree, then a_e will be

$$a_e = \frac{a_o}{E} = 0.1185/\text{degree} \text{ , where } E = 1.013.$$

(Here E is the ratio of wing semi-perimeter to span.) Angle of attack, then, will be 10.52^{o} at $C_{L_{MAX}}$. The wing drag coefficient will be

$$C_{D_W} = C_{DP} + C_{Di}$$

At $C_{L_{MAX}}$, C_{DP} will be 0.0090 and C_{Di} may be approximated as

$$C_{Di} = C^2_{L_{MAX}} (1 + \delta)$$

The wing efficiency parameter $(1+\delta)$ is defined in Ref. 8 as 1.05, so C_{Di} becomes 0.0255 and the chord load can be calculated as 462.9 pounds acting forward. The chord load distribution may then be approximated as shown in Figure 24 below.

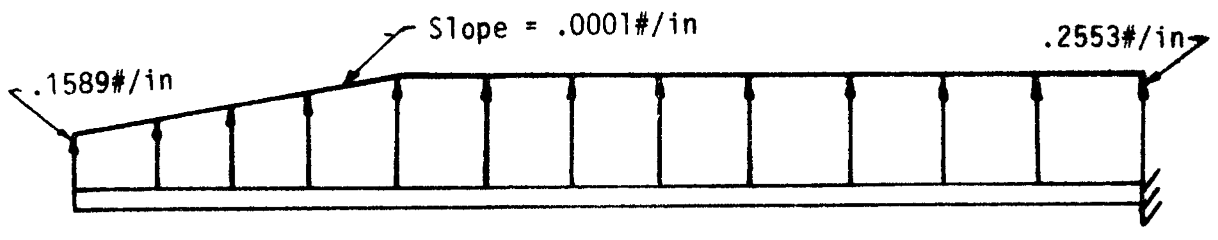


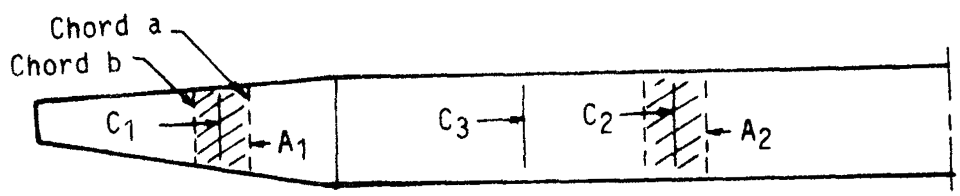
Figure 24. Wing Chord Load Distribution

The chord shear diagram is presented in Figure 25 as is the chord bending moment diagram.

In addition to normal and chordwise loads on the wing spar, torsion will be present due to the basic airfoil pitching moment, $C_{M_{C/4}}$. Torsion, μ, may be defined as

$$\mu = C_{M_{C/4}} \, q \, S \, C_3$$

where S and C_3 are arrived at as numerical iterations across the wing, the product being indicative of the action of a changing moment arm on a constant pitching moment across the wing from root to tip. Schematically, this is shown below.



$$C_3 = \frac{C_1 \Sigma A_1 + C_2 \Sigma A_2}{\Sigma A_1 + \Sigma A_2}$$

$$C_1 = .667 \left[a+b - \frac{ab}{a+b}\right]$$

$$C_2 = 10.22 \text{ ft.}$$

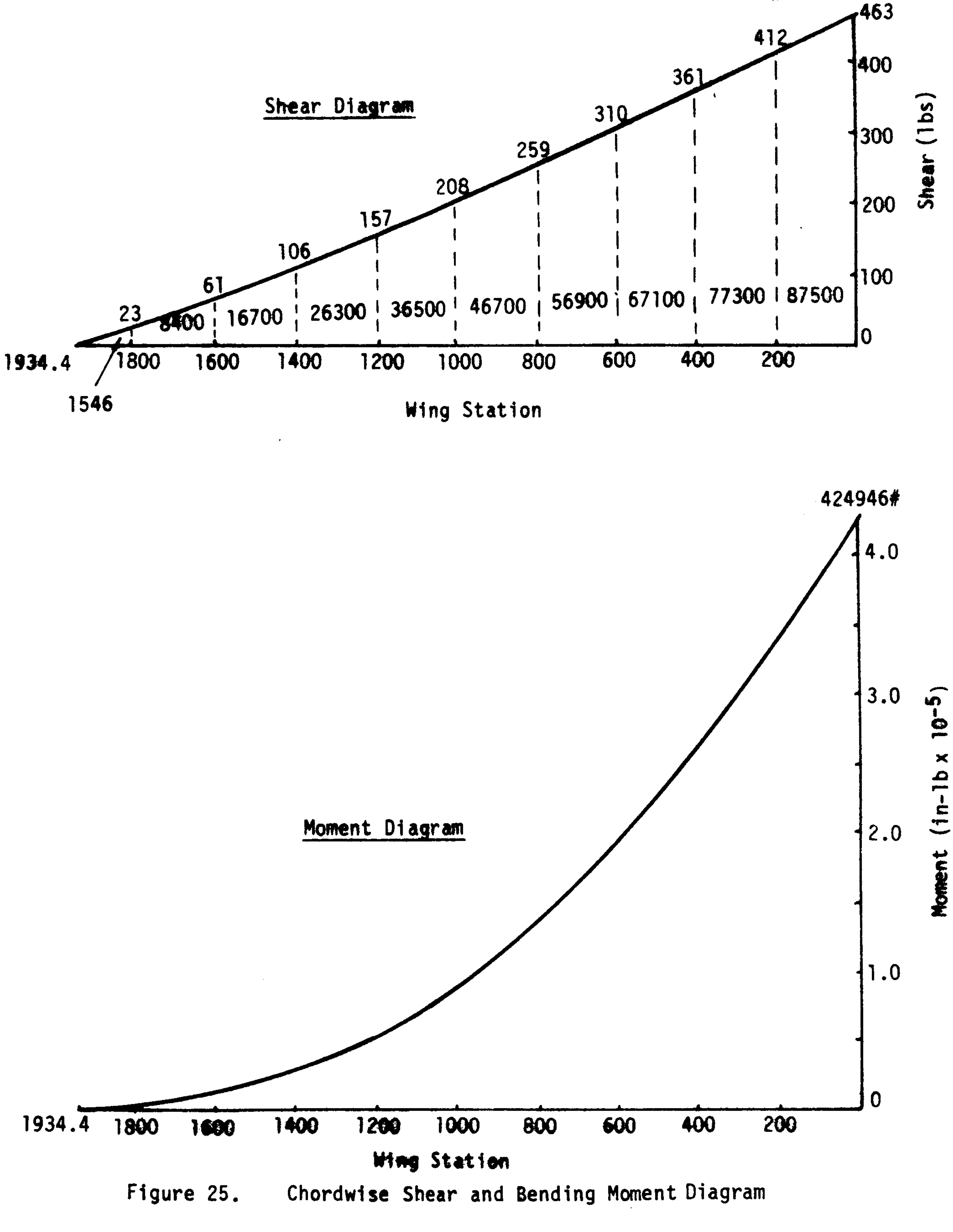
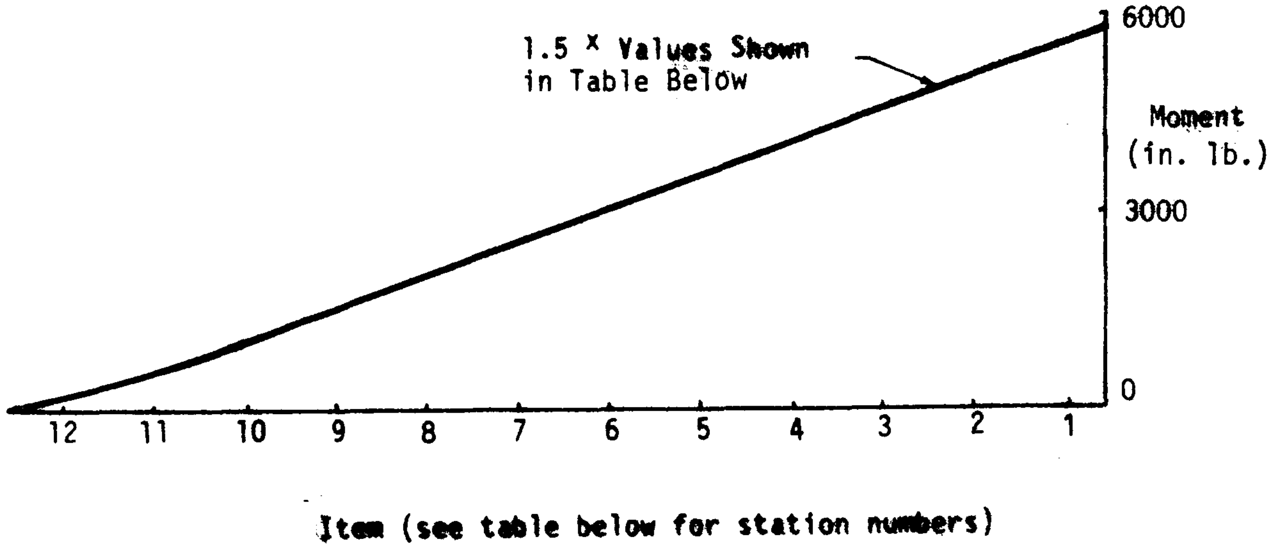


Figure 25. Chordwise Shear and Bending Moment Diagram

Wing torsion due to pitching moment may then be calculated tabularly and the results presented graphically as in Figure 26. Calculations were made at the cruise condition at altitude and a 50% safety factor was added to account for off-design operation.



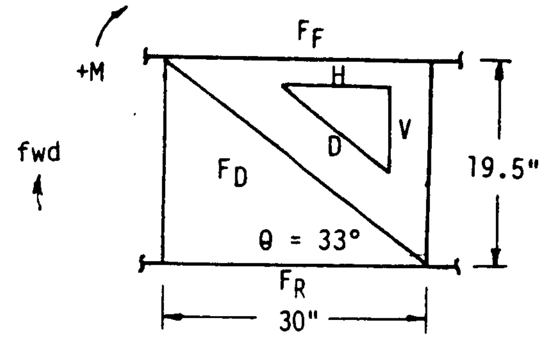
ITEM	STA.	a, FT.	b, FT.	C_1, FT.	C_2, FT.	C_3, FT.	A_1, FT^2	A_2, FT^2	S, FT^2	M, in.lb.
12	1853.9	7.33	6.36	6.89	0	6.89	91.89	0	91.89	-153
11	1692.8	8.29	7.32	7.82	0	7.82	196.68	0	198.68	-375
10	1531.7	9.23	8.29	8.77	0	8.29	314.45	0	314.45	-629
9	1370.6	10.22	9.26	9.75	0	9.75	405.18	0	445.18	-1047
8	1209.4		10.22		10.22	9.86		137.33	582.50	-1385
7	1048.1					9.93		274.66	719.83	-1725
6	886.9					9.98		411.99	857.16	-2063
5	775.6					10.00		549.32	994.49	-2399
4	564.4					10.04		686.66	1131.83	-2741
3	403.1					10.06		823.98	1269.16	-3080
2	241.9					10.07		961.32	1406.49	-3416
1	80.6	10.22	10.22	9.75	10.22	10.08	445.18	1098.65	1543.80	-3753

Figure 26. Wing Torsion Due to Pitching Moment

Normal bending loads in the lift truss may be calculated and are shown in Figure 27. Chord bending loads in the drag truss may also be calculated and those are presented in Table 6. Similarly, torsion loads may be calculated and these are shown in Figure 28 for a typical bay. Note that the caps do not carry any torsion loads. The combined loads in the spar truss due to lift,

TABLE 6. WING CHORDWISE LOADS IN THE DRAG TRUSS

STA.	M	V	H=V/.65	F_F	F_R=F_F-H	F_D=V/.54
0-30	212473	232	357	-10896	10539	426
30-60	206000	228	351	-10564	10213	418
60-90	198500	224	345	-10179	9834	411
90-120	192500	220	339	-9872	9533	404
120-150	185500	216	333	-9513	9180	396
150-180	179000	212	326	-9179	8853	389
180-210	173000	209	322	-8872	8550	383
210-240	166500	205	316	-8538	8222	376
240-270	160500	201	310	-8231	7921	369
270-300	155500	197	303	-7974	7671	361
300-330	148500	193	297	-7615	7318	354
330-360	142500	189	291	-7308	7017	347
360-390	137000	186	286	-7026	6740	341
390-420	131000	182	280	-6718	6438	334
420-450	126000	178	274	-6462	6188	327
450-480	121000	174	268	-6205	5937	319
480-510	116000	170	261	-5949	5688	312
510-540	111500	166	256	-5718	5462	305
540-570	107000	163	251	-5487	5236	302
570-600	102500	159	245	-5256	5011	294
600-630	96000	155	238	-4923	4685	287
630-660	92500	152	234	-4744	4510	281
660-690	87000	148	228	-4462	4234	274



F_F = M/19.5 [Sign is - for Compression]
H = V/Tan 33° = V/.65
F_D = V/Sin 33° = V/.54 [Sign is + for Tension]
F_R = F_F - H [Sign is + for Tension]

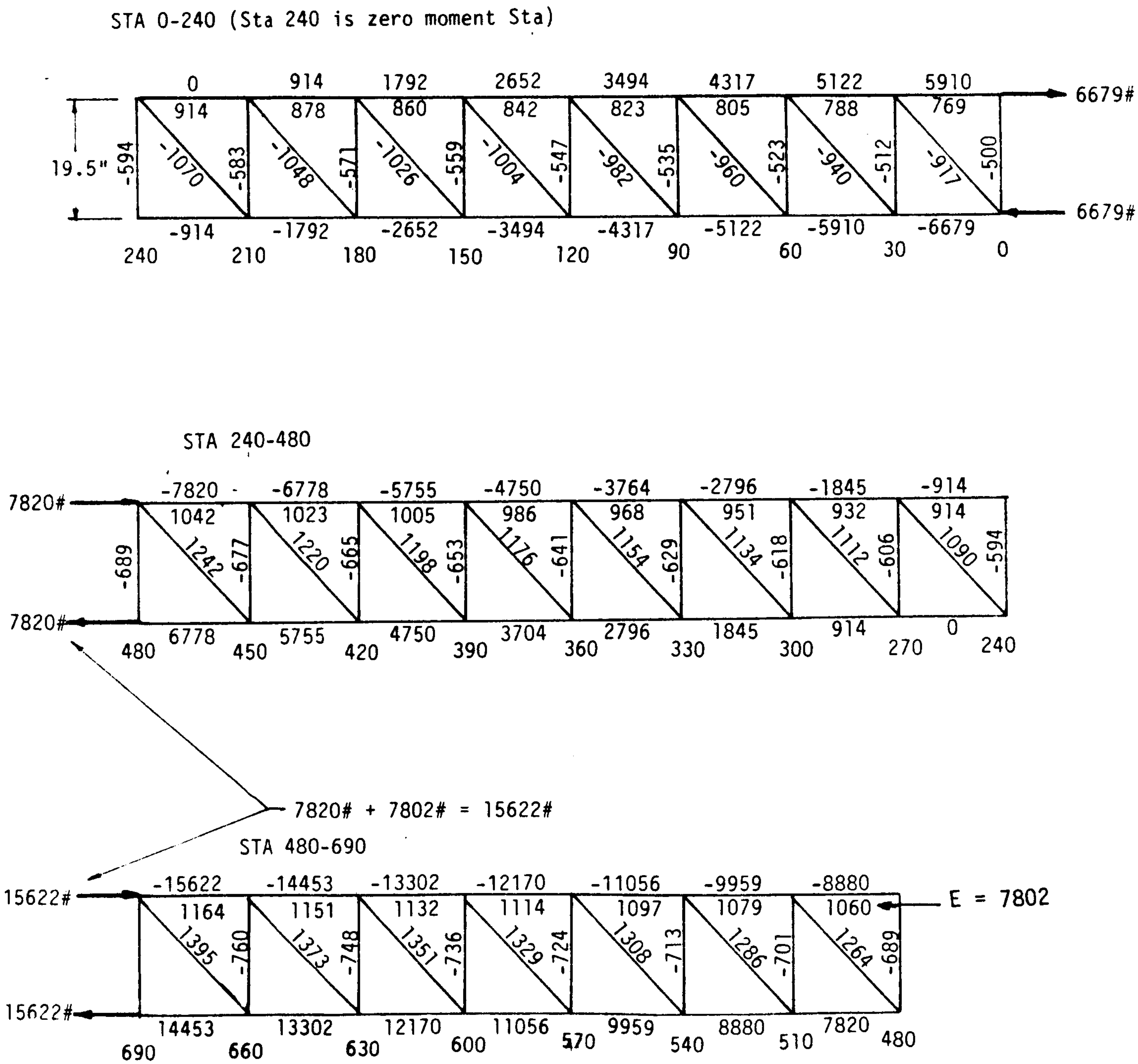


Figure 27. **Wing Normal Bending Loads in the Lift Truss**

Load in Member = $\frac{TL}{2A}$, where: T = Torque, in. lbs.
L = Length of member, in.
A = Enclosed area of torsion Truss = $(19.5)^2$ = 380.25 in.^{2}

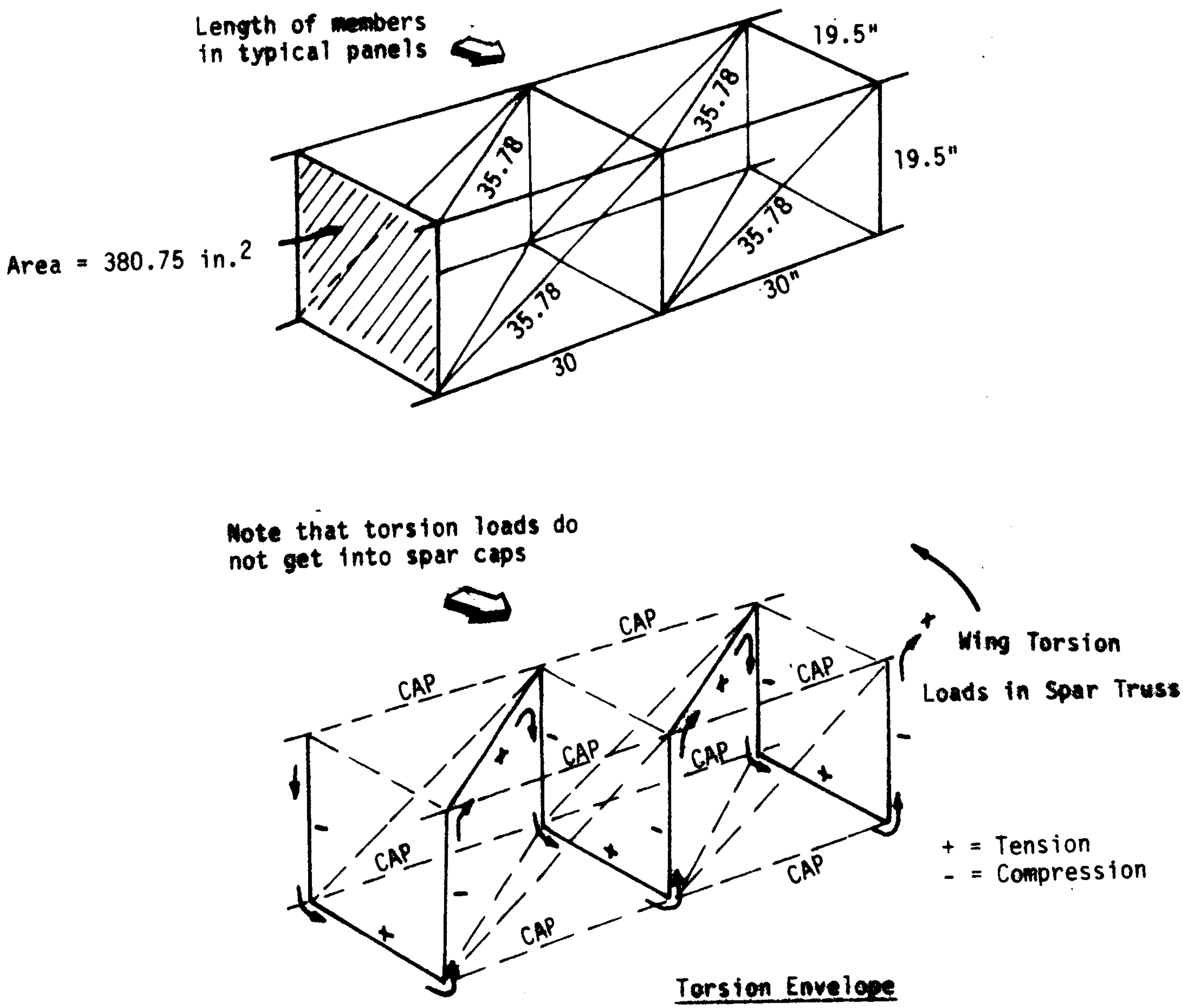


Figure 28. Wing Torsion Loads in Spar Truss

drag, and pitching moment on the wing may then be calculated. Between WS0 and WS690 there are 23 30-inch bays, each one with an average of 14 members, for a total of 322 members. Since calculating the net loads in each of these 322 members is time consuming and costly, only 4 bays will be investigated:

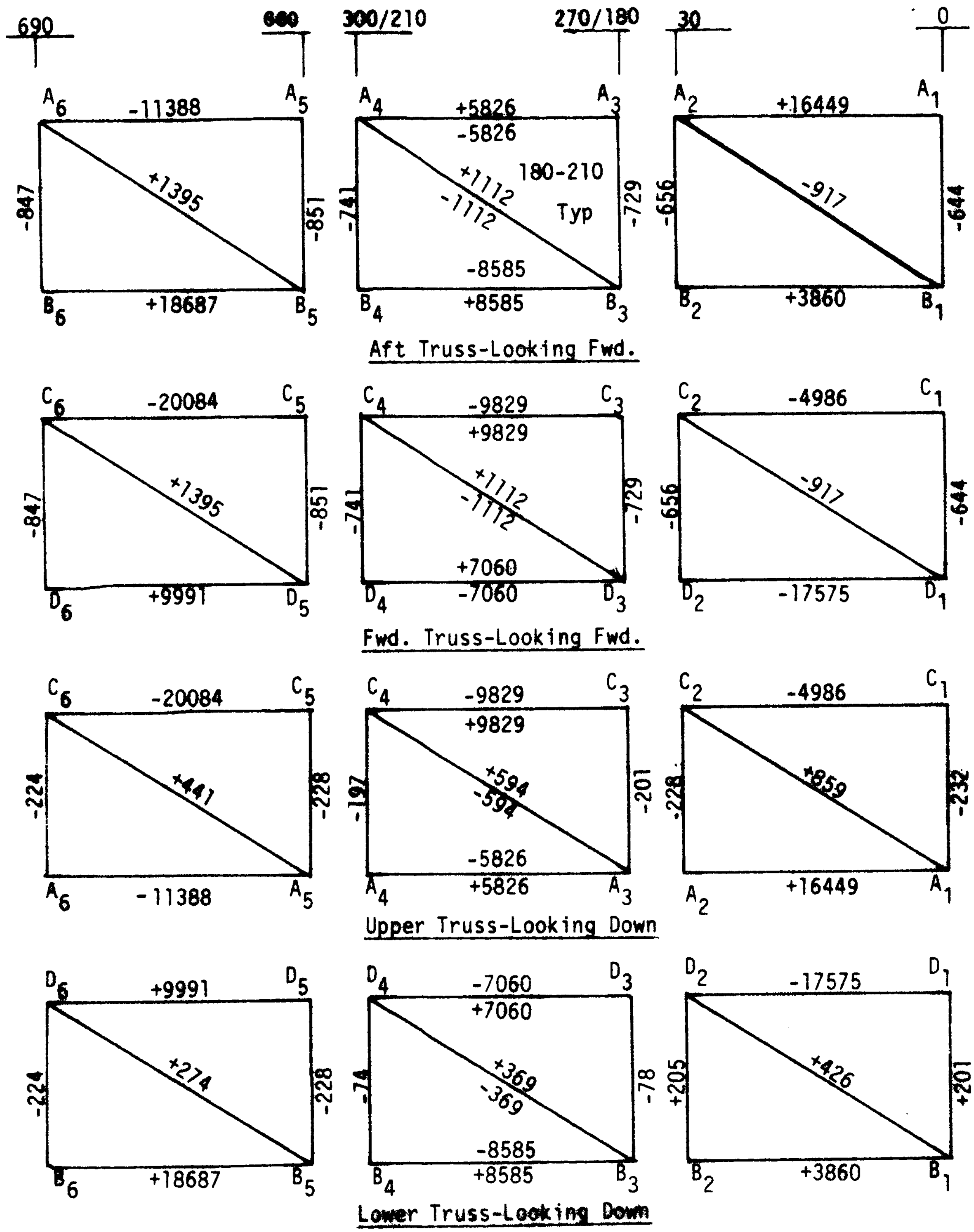
- WS 690-660 which has the highest positive bending moment from lift;
- WS 300-270 which is close to the lowest positive bending moment from lift;
- WS 210-180 which is close to the lowest negative bending moment from lift (the loads in this bay are opposite in sign to those in WS 300-270)
- WS 30-0 which has the highest negative bending moment from lift.

Note that bending moment from lift is reasonably linear from WS0 to WS690. Sizing of truss members in this area will, therefore, assume a linear variation in loads. Figure 29 presents this summary of net loads in wing truss members. These data are presented tabularly in Table 7. Recall that

- Lift loads are based on ultimate load ($n=\pm 3$);
- Drag loads are based on $C_{L_{MAX}}$;
- Torsion loads are based on V_{MAX}

in looking at Figure 27 and Table 7. Since these conditions will not occur simultaneously, this should be a conservative estimate of loads.

The loads in the spar caps may be calculated next. Lift loads (column loads) outboard of WS690 may be calculated assuming that lower cap column loads are half upper cap loads. This assumption is based on the vehicle having a negative load factor of half the positive value. The result is presented below in Table 8.



*For selected bays

NOTE: Assume effect of torsion in STA. 180-210 bay same as 270-300 bay (loads are small)

Figure 29. Summary of Net Load in Wing Truss Members*

TABLE 7. SUMMARY OF NET LOAD IN WING TRUSS AT SELECTED BAYS

NOTE: Lift loads are based on n=3.0. Drag loads are based on $C_{L_{MAX}}$. Torsion loads are based on V_{MAX}. This is likely a worst-on-worst condition, which may be somewhat conservative from the standpoint of structural weight.

STA.	MEMBER	TORSION*	TORSION LOAD**	LIFT LOAD	DRAG LOAD	NET LOAD
0-30	A_1B_1	5630	-144	-500	0	-644
	A_2B_2		-144	-512	0	-656
	A_1A_2		0	+5910	+10539	+16449
	B_1B_2		0	-6679	+10539	+3860
	A_2B_1		0	-917	0	-917
	C_1D_1		-144	-500	0	-644
	C_2D_2		-144	-512	0	-656
	C_1C_2		0	+5910	-10896	-4986
	D_1D_2		0	-6679	-10896	-17575
	C_2D_1		0	-917	0	-917
	C_1A_1		0	0	-232	-232
	C_2A_2		0	0	-228	-228
	C_2A_1		+433	0	+426	+859
	D_1B_1		+433	0	-232	+201
	D_2B_2		+144	0	-228	-84
0-30	D_2B_1	5630	0	0	+426	+426
270-300	A_3B_3	4780	-123	-606	0	-729
	A_4B_4		-123	-618	0	-741
	A_3A_4		0	-1845	7671	+5826
	B_3B_4		0	+914	7671	+8585
	A_4B_3		0	+1112	0	+1112
	C_3D_3		-123	-606	0	-729
	C_4D_4		-123	-618	0	-741
	C_3C_4		0	+1845	-7974	-9829

TABLE 7. SUMMARY OF NET LOAD IN WING TRUSS AT SELECTED BAYS (CONT)

STA.	MEMBER	TORSION*	TORSION LOAD**	LIFT LOAD	DRAG LOAD	NET LOAD
	D_3D_4		0	+914	-7974	-7060
	C_4D_3		0	+1112	0	+1112
	C_3A_3		0	0	-201	-201
	C_4A_4		0	0	-197	-197
	C_4A_3		+225	0	+369	+594
	D_3B_3		+123	0	-201	-78
	D_4B_4		+123	0	-197	-74
270-300	D_4B_3	4780	0	0	+369	+369
660-690	A_5B_5	3553	-91	-760	0	-851
	A_6B_6		-91	-772	0	-863
	A_5A_6		0	-15622	+4234	-11388
	B_5B_6		0	+14453	+4234	+18687
	A_6B_5		0	+1395	0	+1395
	C_5D_5		-91	-760	0	-851
	C_6D_6		-91	-772	0	-863
	C_5C_6		0	-15622	-4462	-20084
	D_5D_6		0	+14453	-4462	+9991
	C_6D_5		0	+1395	0	+1395
	C_5A_5		0	0	-228	-228
	C_6A_6		0	0	-224	-224
	C_6A_5		+167	0	+274	+441
	D_5B_5		+91	0	-228	-137
	D_6B_6		+91	0	-224	-133
660-690	D_6B_5	3553	0	0	+274	+274

$$\text{*TORSION} = M_{\text{STA }0} - \left[\text{WING STA} \times \frac{d\mu}{d\,\text{STA}}\right] = 5630 - \left[\text{WING STA} \times 3.147\right]$$

**TORSION LOAD = TL/2A = TL/760.5
[MOMENT TAKEN AS HIGHEST IN BAY; INBD. STA.]

Truss members inboard of WS690 will be sized to act as short columns except for the last two in the table which will be treated as long columns. Candidate tubes are then:

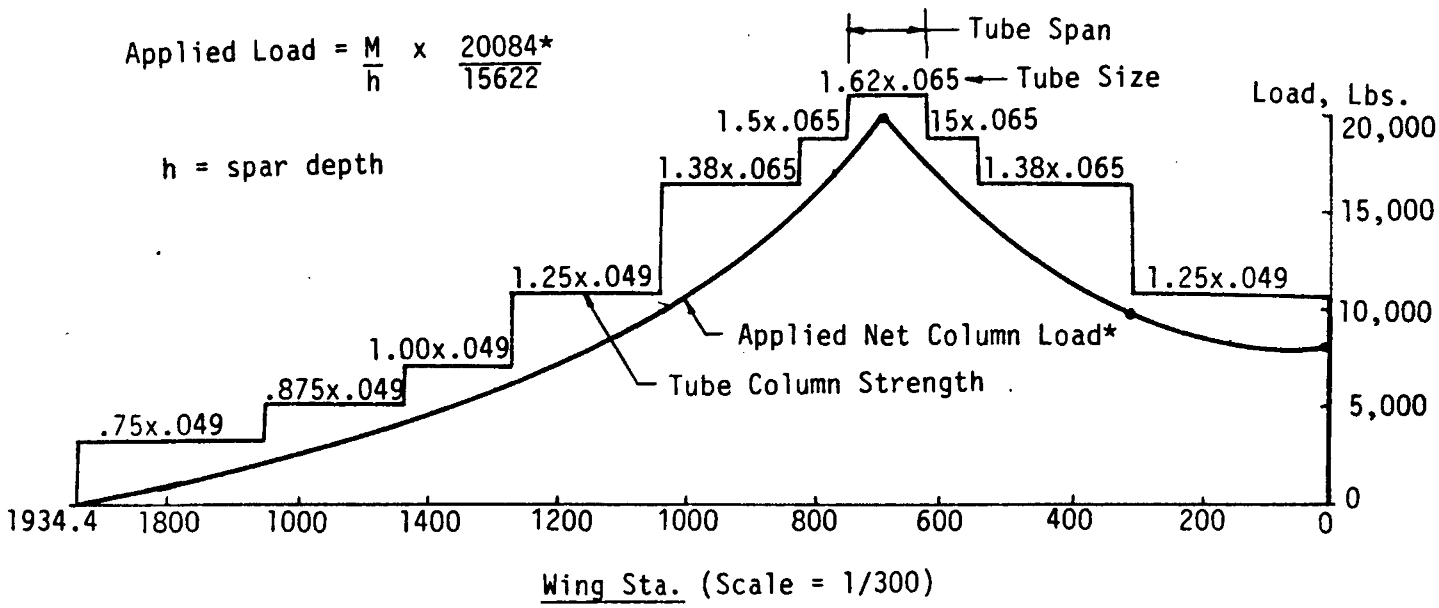
TUBE SIZE	A	ρ	L'/ρ	F_C	STRENGTH= $F_C \times A$
1.62x.065	.3186 in.^{2}	.5520 in.	44.38	66908 psi	21317 lbs.
1.50x.065	.2930	.5079	48.24	64765	18976
1.38x065	.2675	.4637	52.83	62098	16611
1.25x.049	.1844	.4250	57.65	59197	10916
1.00x.049	.1464	.3307	72.77	49278	7206
.875x.049	.1272	.2925	83.76	41279	5251
.750x.049	.1079	.2484	98.63	30438	3339
.625x.049	.0887	.2044	119.86	20610	1828

Note: L' = tube length adjusted for end fixity = $L/\sqrt{1.5}$

Spar cap sizes for both forward and aft spar trusses will be made the same size for ease of manufacturing. Upper caps are designed for the highest column load in the bay (looking down) for the maximum positive load factor case. Lower caps will be designed for the maximum negative load factor case. All other members will be 0.75 O.D.x0.028 wall as sized by the maximum column load in diagonal members. The spar cap size distribution is shown in Figure 30 for both upper and lower caps.

Diagonals and verticals in the lift truss may now be sized assuming all members will have the same O.D. for cost and case of manufacture. It should be noted that diagonals are in tension at all positive flight conditions and in compression in all negative flight conditions. All members will be 0.62 inch in diameter and wall thickness will vary from 0.028 inch the first three bays to 0.022 inch in the rest. Verticals, on the other hand, will vary in diameter from 0.50 inch at the tip to 0.62 inch as column strength dictates.

Upper Cap



* 20084# = Net load in member C5C6, 15622# = Lift load only in member C5C6 ratio, 20084/15622 used as correction to M/h to give rapid estimate of net laods outbd. of sta. 690 due to combined lift, drag & torsion. Loads inbd. of sta. 690

Lower Cap [Compression loads due to negative flt. cond.; n = -1.5]

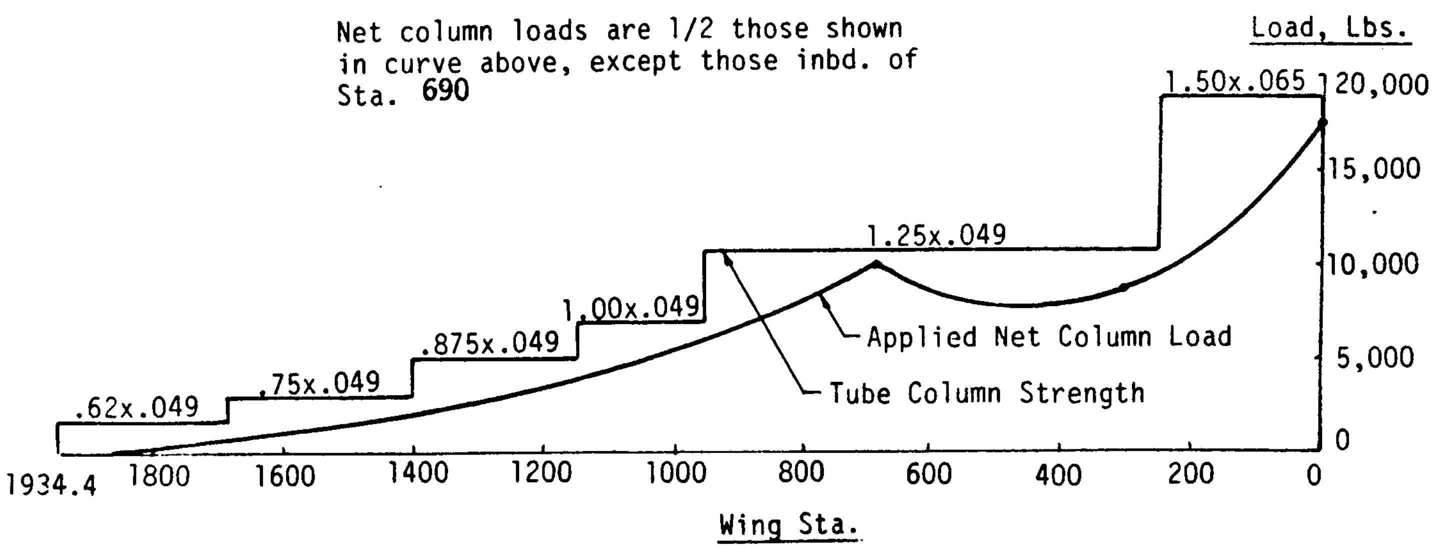


Figure 30a. Spar Cap Size Distribution

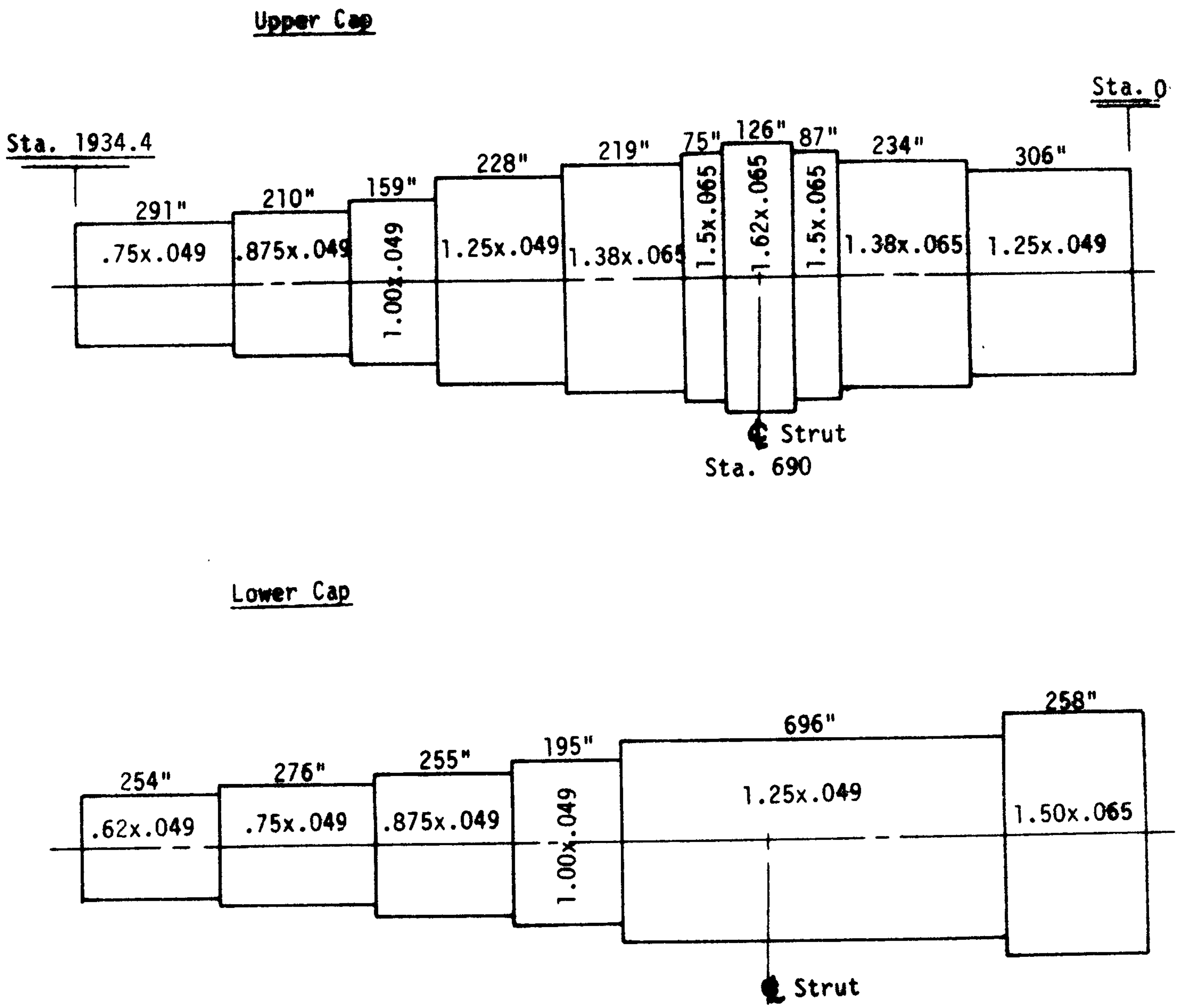


Figure 30b. Summary of Spar Cap Sizes & Lengths
[Scale: Dia = Full, Length = 1/300]

The wing strut may be sized at this point. Referring to the loads shown in Figure 18, the 2233 pound shear load at WS690 translates to an 11488 pound axial load and an 11703 pound tensile load in the strut which intercepts the wingspan at an 11 degree angle. If the column load is half the tensile load, then 5852 pounds must be designed for. The total length of the strut is 703 inches (58.6 feet). If a jury strut is added at the halfway point in the strut, then overall strut size can be as small as 4.00 inch O.D. x 0.120 inch wall. This tube will weigh roughly 70 pounds including fairings and fittings.

Diagonals and chordwise drag truss members can be sized next. The diagonals are roughly 36 inches in length and must absorb a maximum of about 600 pounds. This can be handled by a 0.62 O.D. x 0.022 wall graphite epoxy tube. Chordwise members are roughly 20 inches in length;and the worst load in any member is 232 pounds. The same size tube can handle this load with an excessive margin of safety, but 0.022 inch wall thickness is about the minimum practical size for manufacturing. Outboard of WS690 the same design approach applies. Both diagonal and chordwise members will be 0.50 inch O.D. x 0.022 inch wall thickness. The vertical members will have to mate with caps and so will be 0.62 inch O.D. x 0.022 inch wall thickness. Table 9 summarizes tube thicknesses and gives a weight breakdown for the truss.

Fully Cantilevered Wing. Much theoretical and empirical work has been done on the structural design of fully cantilevered wings for sailplanes. Referring to the shear and bending moment calculations for the strut-braced wing, the strut may be removed and the shears and bending moments recalculated as shown in Figure 31. Wing truss structural details may then be addressed.

Several stations may be chosen and the critical loads calculated. Results are presented in Table 10. Once this is done, tube sizes may be calculated. Results are presented in Table 11. Figures 32 and 33 present an idea of the margin of safety in the caps at each point along the span and how tubes will telescope together.

TABLE 9. SUMMARY OF TUBE THICKNESSES AND WEIGHTS FOR SPAR

Upper Cap Tubes

SIZE	AREA	VOLUME	WEIGHT
1.62X.065X63"	.3186 in^2	20.1 in^3	1.22 lbs.
1.50x.065x75"	.2430	18.2	1.11
1.38x.065x219"	.2075	45.4	2.77
1.25x.049x228	.1849	42.2	2.57
1.00x.049x159"	.1464	23.3	1.42
.875x.049x210"	.1272	26.7	1.63
.75x.049x291"	.1079	31.4	1.92
			12.64 lbs. for 1 truss

Lower Cap Tubes

SIZE	AREA	VOLUME	WEIGHT
1.25x.049x264"	.1849 in^2	48.8 in^3	2.98 lbs.
1.00x0.49x195"	.1464	28.5	1.74
.875x.049x255"	.1272	32.4	1.98
.75x.049x276"	.1079	29.8	1.82
.62x.049x254"	.0887	22.5	1.37
			9.89 lbs. for 1 truss

Diagonals

STA.	NO. MEMBERS	SIZE	AREA	VOLUME	WEIGHT
690-780	3	.62X.028X35.78"	.0525	5.64 in^3	.34 lbs.
780-1290	17	.62x.022x35.78"	.0417	25.4	1.55
1290-1934.4	21	.62x.022x34.20"	.0417	29.9	1.82
					3.71 lbs. for 1 Truss

Verticals

690-1290	20	.50x.022x19.5"	.0330 in2	12.9 in^3	.79 lbs.
1290-1934.4	21	.50x.022x15.82"	.0330	11.00	.67
					1.46 lbs. for 1 Truss

DRAG TRUSS

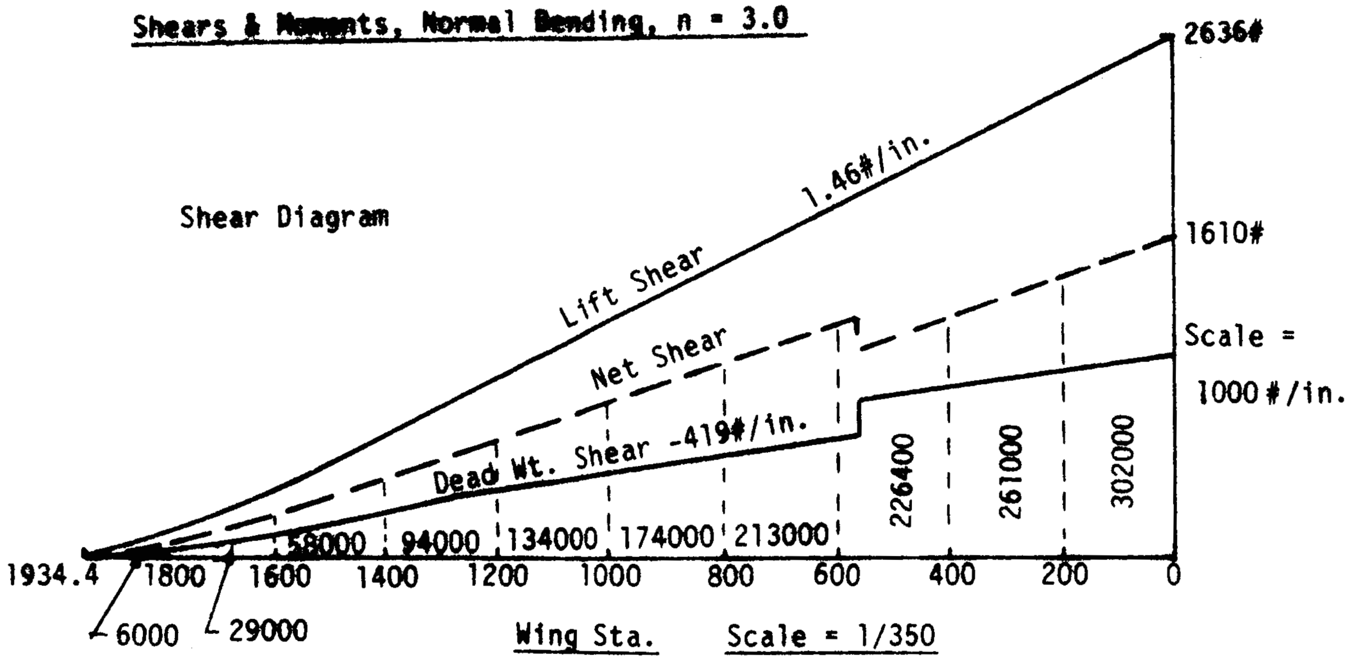
Diagonals

0-870	29	.62x.022x35.78"	.0417 in^2	43.3 in^3	2.64 lbs
870-1290	14	.50x.022x35.78"	.0330	16.5	1.00
1290-1934.4	21	.50x.022x34.20"	.0330	73.6	1.44
					6.08 lbs. for 1 Truss

Chordwise Members

0-690	23	.62x.022x19.5"	.0417 in^2	18.7 in^3	1.14 lbs.
690-1290	20	.50x.022x19.5"	.0330	12.9	.79
1290-1934.4	21	.50x.022x15.82"	.0330	11.0	.67
					2.60 lbs. for 1 Truss

Shears & Moments, Normal Bending, n = 3.0



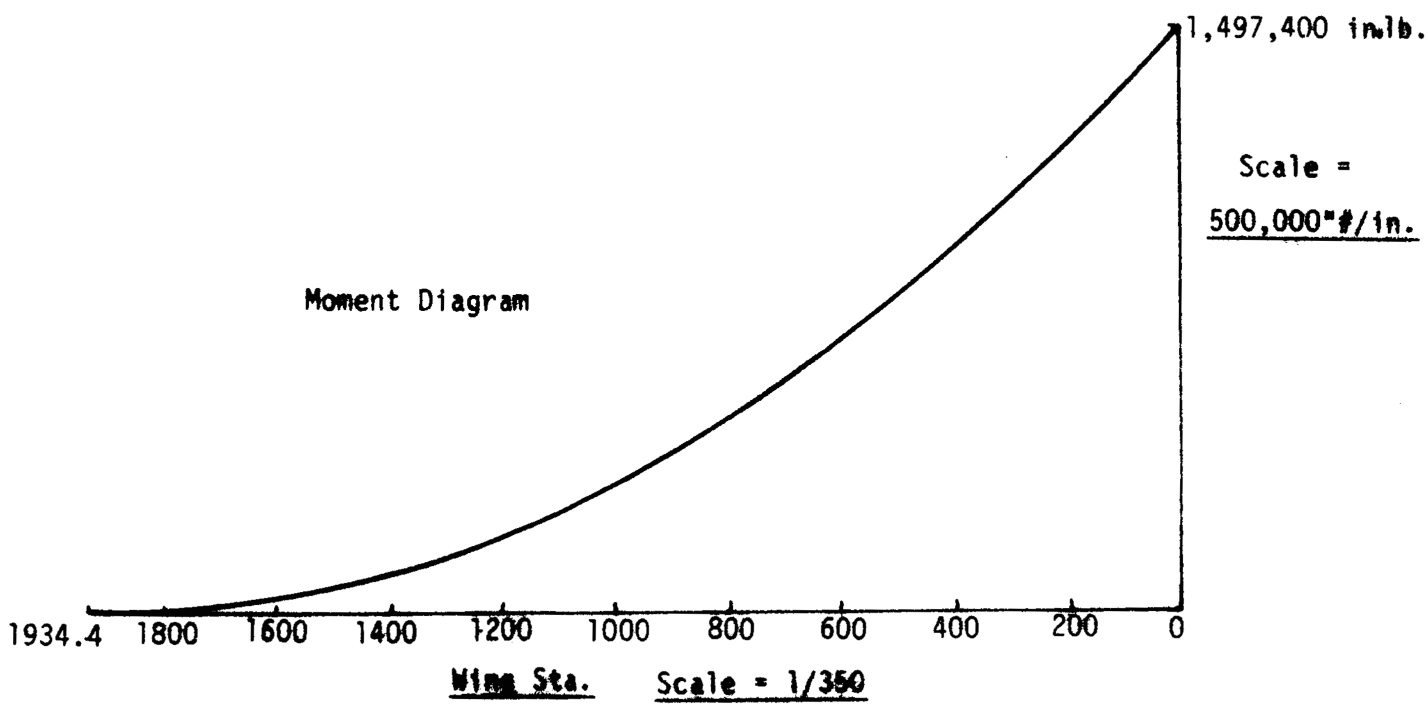


Figure 31. Shear and Bending Moment Diagrams For Fully Cantilevered Wing

TABLE 10. SPAR CAP COLUMN LOADS AT SELECTED WING STATIONS

STA.	LWR. CAP LOAD, LBS	LWR. CAP LOADS x 19.5/1b
0	24,646	26,700 lbs
400	15,262	16,533
800	8,090	8,764
1290	2,372	2,570
1600	666	722

TABLE 11. CANDIDATE TUBES FOR SPAR

TUBE SIZE	A	ρ	L'/ρ	F_C	STRENGTH= F_C x A
3.00x.083	.7606 in^2	1.0317 in.	24.49 in.	76706 psi	56342 lbs.
2.50x.065	.4972	.8612	28.44	74747	37164
2.75x.049	.4158	.9551	25.64	75933	31573
2.75x.058	.4905	.9520	25.72	75899	37228
1.62x.065	.3186	.5520	44.38	66908	21317
1.50x.065	.2930	.5079	48.24	64765	18976
1.38x.065	.2675	.4637	52.83	62098	16611
1.25x.049	.1844	.4250	57.65	59197	10916
1.00x.049	.1464	.3307	72.77	49218	7206
.875x.049	.1272	.2925	83.76	41279	5251
.750x.049	.1079	.2484	98.63	30438	3339
.625x.049	.0887	.2044	119.86	20610	1828
2.75x.065	.5483	.9496	25.79	75872	41600
2.75x.083	.6954	.9434	25.96	75802	52712
2.50x.049	.3773	.8667	28.26	74825	28232

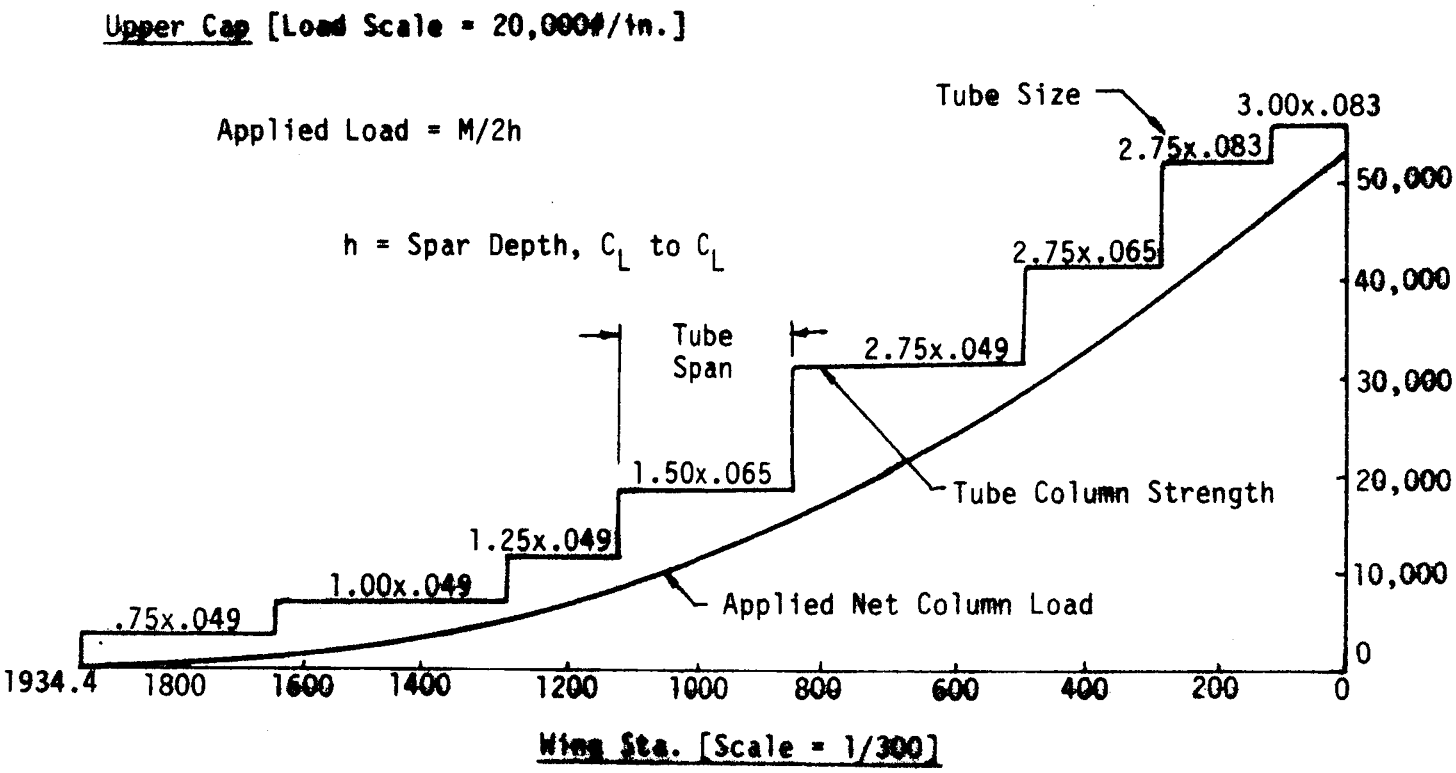


Figure 32. Distribution of Spar Cap Sizes Along Semispan

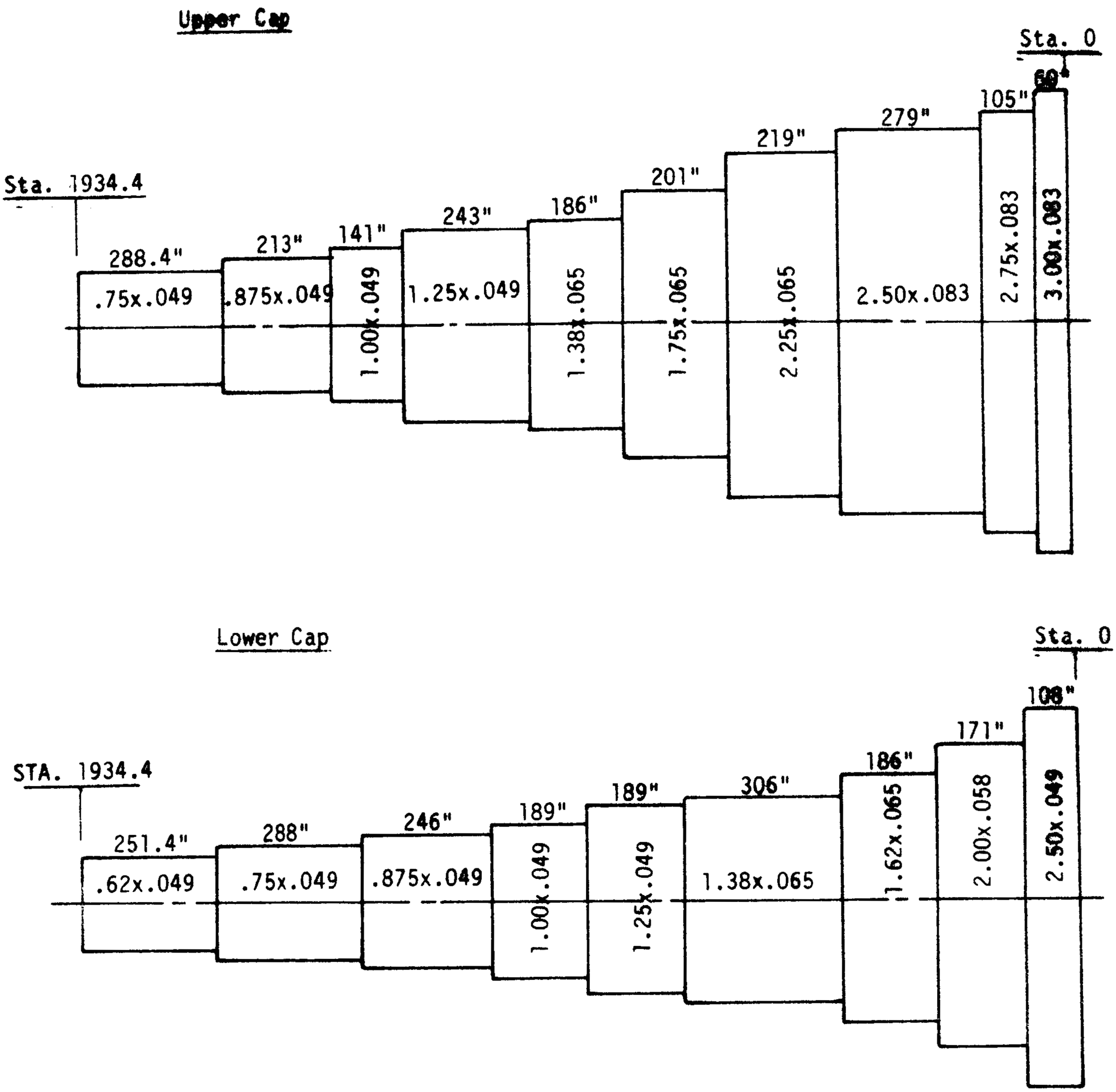


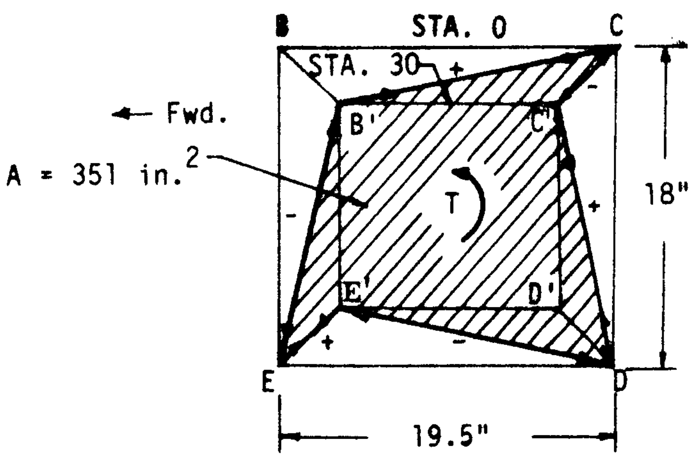
Figure 33. Summary of Spar Cap Sizes [Scale: Dia = Full, Length = 1/300]

Diagonals and verticals may be sized next. All members outboard of WS690 will be the same as for the strut-braced wing since loads are the same for both wings. From WS0 to WS690, the load in any member will be

$$\text{Load in any member} = \text{Torsion Load} \pm \text{Lift Load} \pm \text{Drag Load}$$

Figure 34 presents a slightly distorted view of the bay from WS0 to WS30 with torsional load signs shown. These loads are summarized along with lift and drag loads in Table 12. Members CD and EB will be column critical for the maximum positive load condition and member C'D is column critical for the maximum negative load condition. Members may then be sized and their weights calculated. Results are presented in Table 13 for truss weights.

Wire Braced Wing. Calculation of loads in wire braced structures is more complicated than in the other bracing schemes examined so far. For that reason, the wing will be broken into elements starting at the wingtip. The bracing scheme chosen for analysis is shown in Figure 35. Running loads are shown in Figure 36. Loads in each element will be calculated assuming elements are not connected, then the results will be superimposed to obtain a representative loading for the entire wing.



STA. 30 distorted in sketch to show diagonal members & load signs

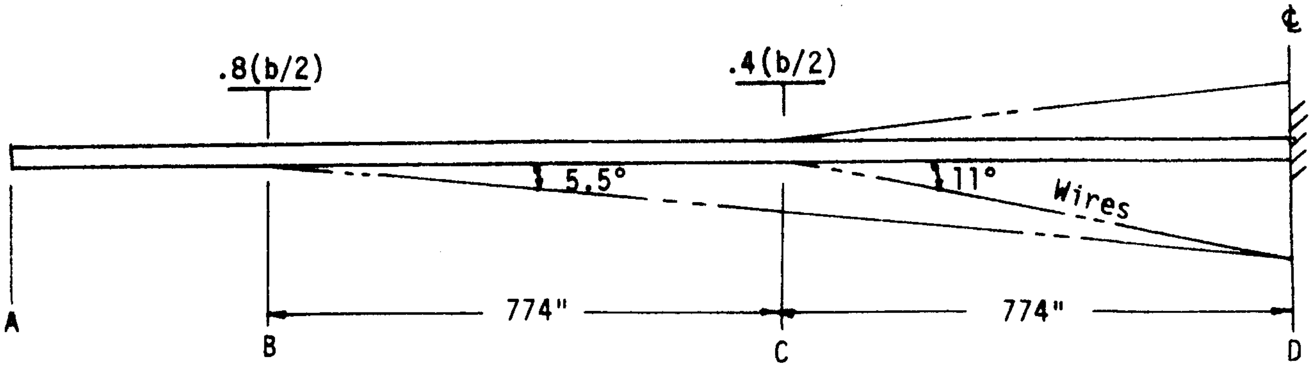
Torsion @ Sta. 30 = 5630 in. lbs.

Torsion Load = TL/2A = 5630L/702 = 8.02L

Member	Length, In.	Torsion Load, Lbs.
B'C	35.78	+287
C'C	30.00	-241
C'D	35.00	+281
E'D	35.78	-287
E'E	30.00	+241
B'E	35.00	-281
		= 0

Figure 34. Loads in Wing Truss Due to Torsion

Bracing Scheme



A is free
B & C are simply supported
D is fixed

Figure 35. Wing Spar Design

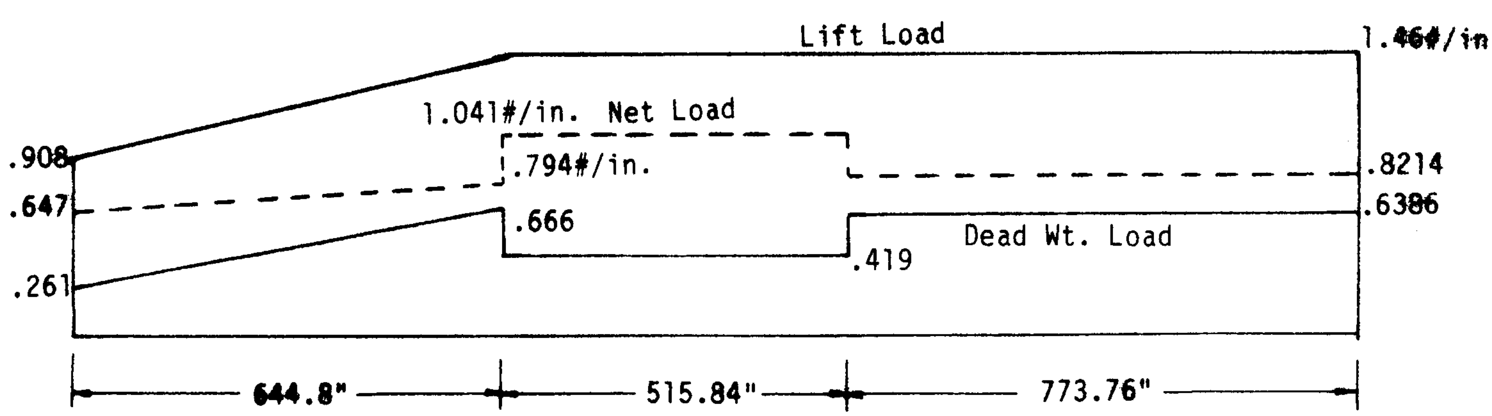


Figure 36. Running Loads in Spar

TABLE 12. NET LOADS IN VERTICAL, CHORDWISE & DIAGONAL MEMBERS

MEMBER	TORSION LOAD	LIFT LOAD	DRAG LOAD	NET LOAD
BC	0 lbs.	0 lbs.	-232 lbs	-232 lbs.
CD	0	-805	0	-805
DE	0	0	-232	-232
EB	0	-805	0	-805
B'C	+287	0	+425	+712
B'E	-281	+1565	0	+1284
C'D	+281	+1565	0	+1846
E'D	-287	0	+425	+138

Element AB is a fully cantilevered section of outboard wing, and the loads which will be transferred to the rest of the wing at its inboard extremity can be calculated accordingly. Element BC can be considered fixed at both ends as can element CD for purposes of bending moment calculations, and both can be considered simply supported for shear load calculations. Figure 37 (top) shows the loadings of each of these sections, and the resultant load centroids are presented at the bottom. Figure 38 presents the shear and free moment diagrams for each wing section. Table 14 summarizes the moment distribution on the wing.

TABLE 13. SPAR WEIGHT SUMMARY FOR CANTILEVER WING

UPPER CAPS

SIZE	AREA	VOLUME	WEIGHT
3.00x.083x60"	.7606 in^2	45.64 in^3	2.78 lbs
2.75x.083x105"	.6954	73.02	4.45
2.50x.083x279"	.6302	175.83	10.73
2.25x.065x219"	.4462	97.72	5.96
1.75x.065x201"	.3441	69.16	4.22
1.38x.065x186"	.2675	49.76	3.04
1.25x.049x243"	.1849	44.93	2.74
1.00x.049x141"	.1464	20.64	1.26
.875x.049x213"	.1272	27.09	1.65
.75x.049x288.4"	.1079	31.11	1.90

= 38.73# for 1 Truss

77.46# for 1 Spar (2 Trusses)

LOWER CAPS

SIZE	AREA	VOLUME	WEIGHT
2.50X.049X108"	.3773 in^2	40.75 in^3	2.49
2.00x.058x171"	.3539	60.52	3.69
1.62x.065x186"	.3186	59.26	3.61
1.38x.065x306"	.2675	81.86	4.99
1.25x.049x189"	.1849	34.95	2.13
1.00x.049x189"	.1464	27.67	1.69
.875x.049x246"	.1272	31.29	1.91
.750x.049x288"	.1079	31.08	1.90
.62x.049x251.4"	.0887	22.30	1.36

= 23.77 for 1 Truss

47.54 for 1 Spar (2 Trusses)

VERTICALS IN LIFT TRUSS

STA.	NO. MEMBERS	SIZE	AREA	VOLUME	WEIGHT
0-1290	43	.62X.022X18"	.0417 in^2	32.28 in^3	1.97 lbs.
1290-TIP	21	.62x.022x14.6"	.0417	12.79	.78
					2.75#(1 Truss)

TABLE 13. SPAR WEIGHT SUMMARY FOR CANTILEVER WING (CONT.)

DIAGONALS IN LIFT TRUSS

0-690	23	.62x.028x35"	.0525 in^2	42.26 in^3	2.58 lbs.
690-1290	20	.62x.022x35"	.0417	29.19	1.78
1290-TIP	21	.62x.022x31.5"	.0417	27.58	1.68
					6.04#(1 Truss)

CHORDWISE MEMBERS IN DRAG TRUSS

0-1290	43	.62x.022x19.5"	.0417 in^2	34.97 in^3	2.13 lbs.
1290-TIP	21	.62x.022x15.82"	.0417	13.85	.85
					2.98#(1 Truss)

DIAGONALS IN DRAG TRUSS

0-690	23	.62x.028x35.78"	.0525 in^2	43.20 in^3	2.64 lbs
690-1290	20	.62x.022x.35.78"	.0417	29.84	1.82
1290-TIP	21	.62x.022x34.10"	.0417	29.86	1.82
					6.28#(1 Truss)

SPAR WEIGHT SUMMARY

ITEM	WT. OF 1 COMPLETE SPAR (2 TRUSSES)
Upper Caps	77.46 lbs
Lower Caps	47.54
Verticals	5.50
Diagonals in Lift Truss	12.08
Diagonals in Drag Truss	12.56
Chordwise Members	5.96
	161.10 lbs

Total Wt. of 1 Spar, incl. 15% for Joints & Misc.
= 1.15 x 161.10
= 185.27#

Total Wt. of Both Spars = 370.53 lbs.

NOTE: Spars on wing with strut weigh 280.22#
If 108# is added for strut W_{TOT} = 388.22#.
So, cantilever spars weigh 17.69# less than
strutted spars with strut.

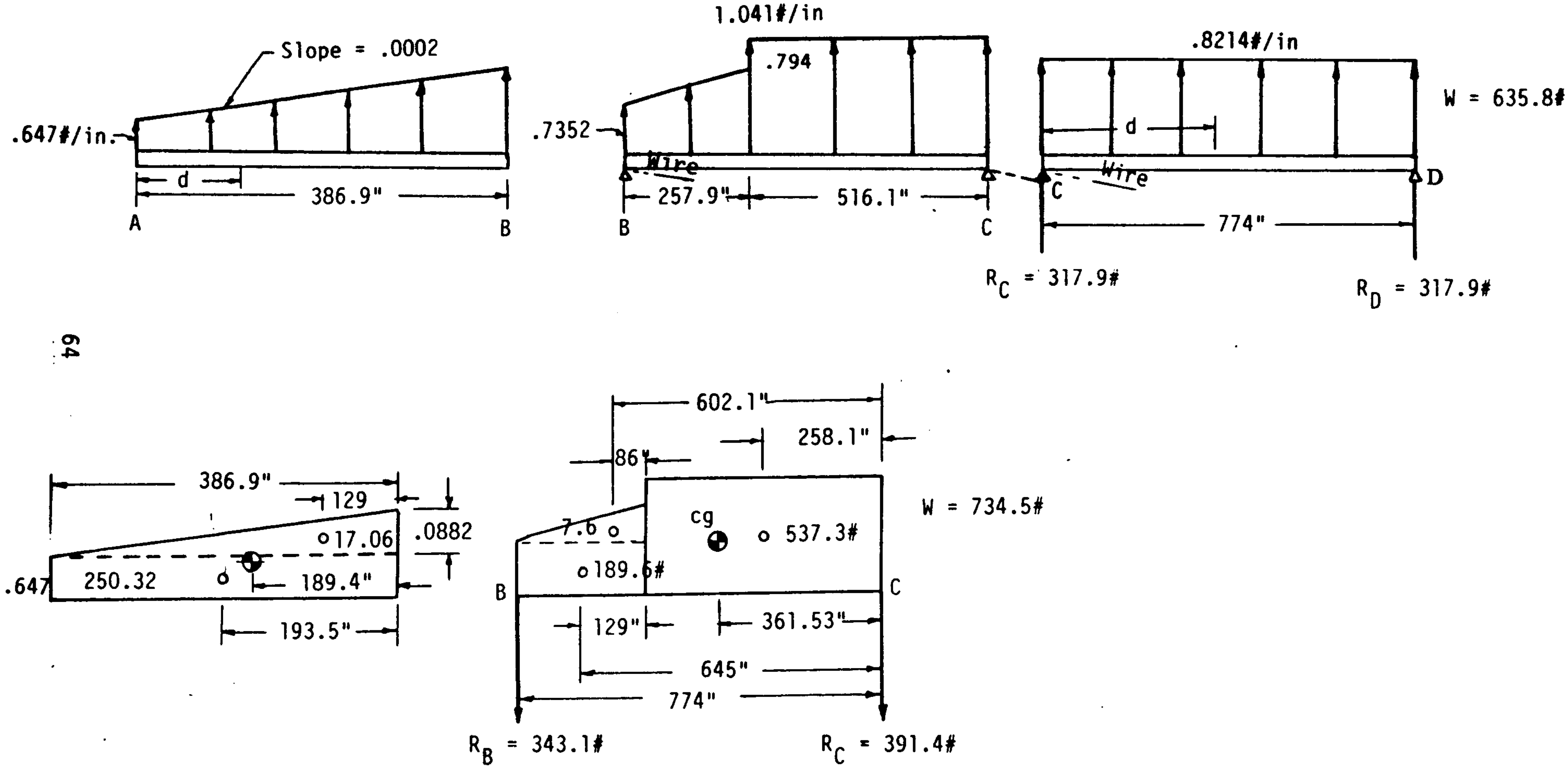
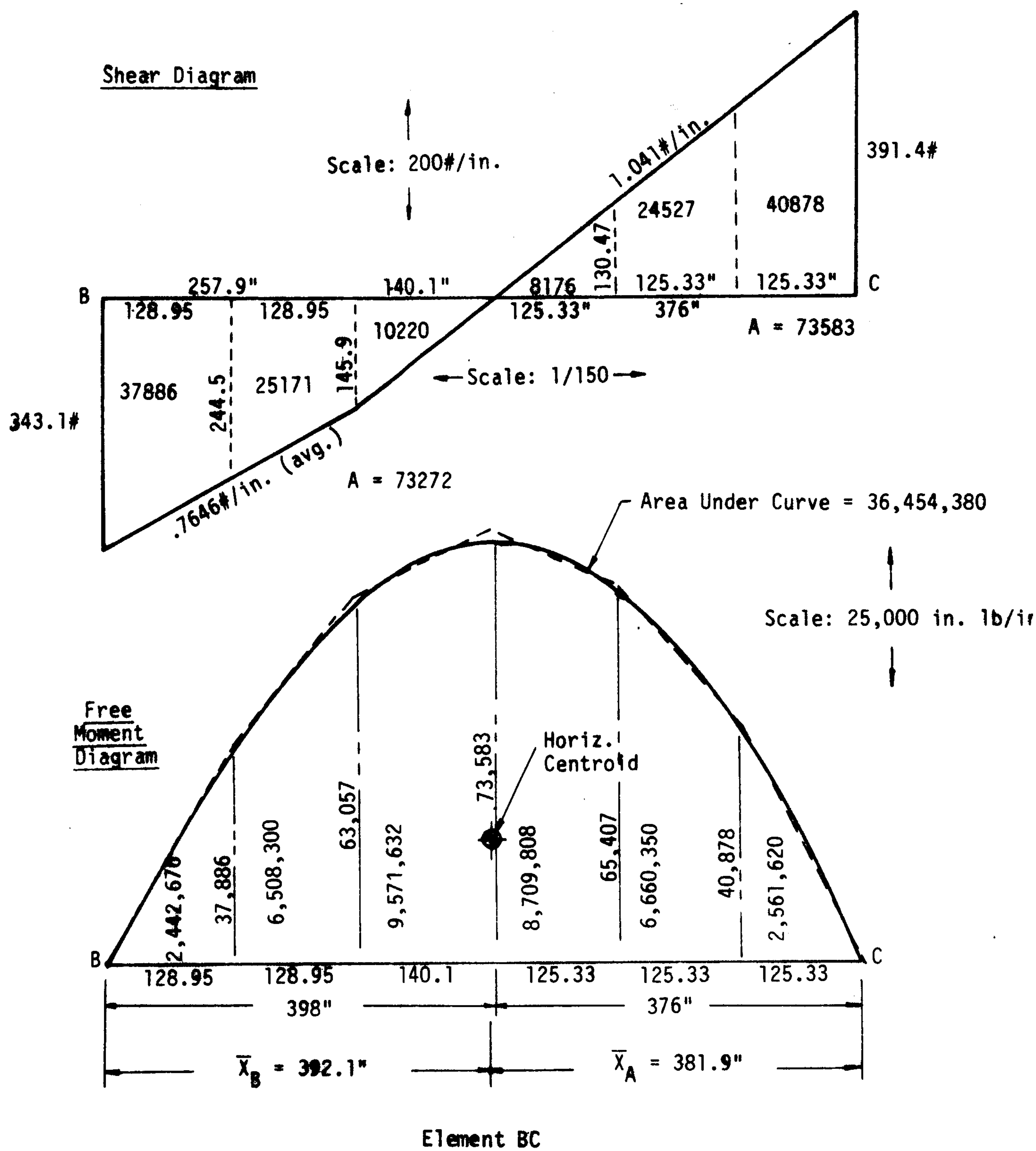


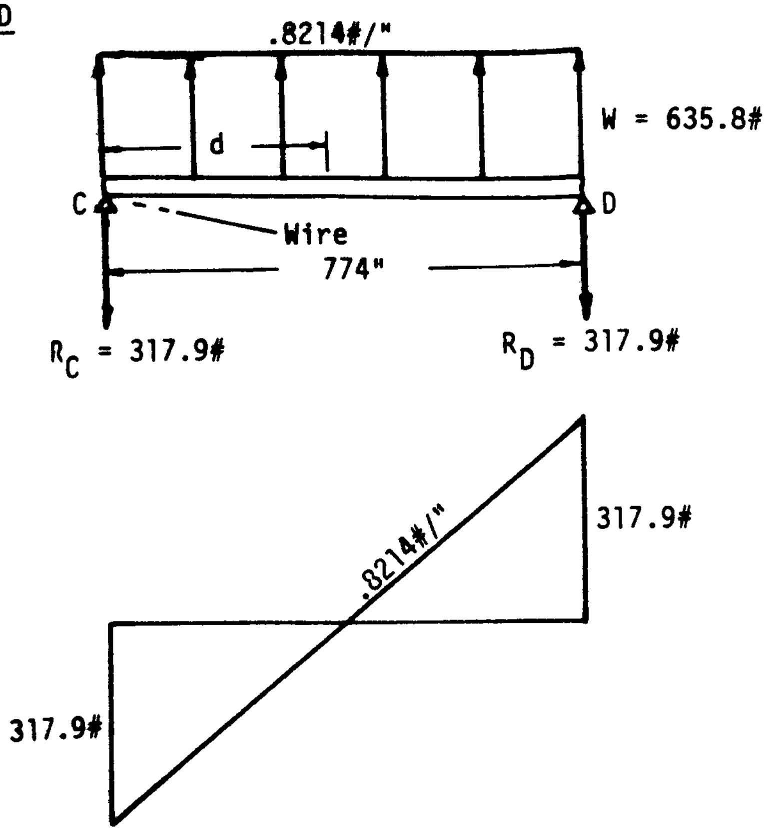
Figure 37. Loads and Centroids of Wing Sections



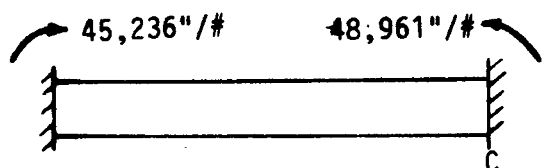
Element BC

Figure 38. Shears and Bending Moments

Element CD

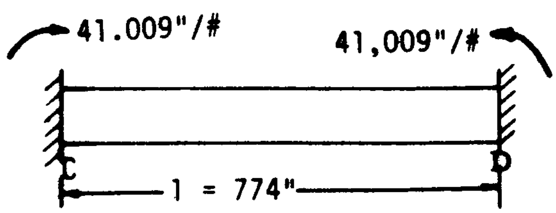


Element BC



Fixed End Bending Moment

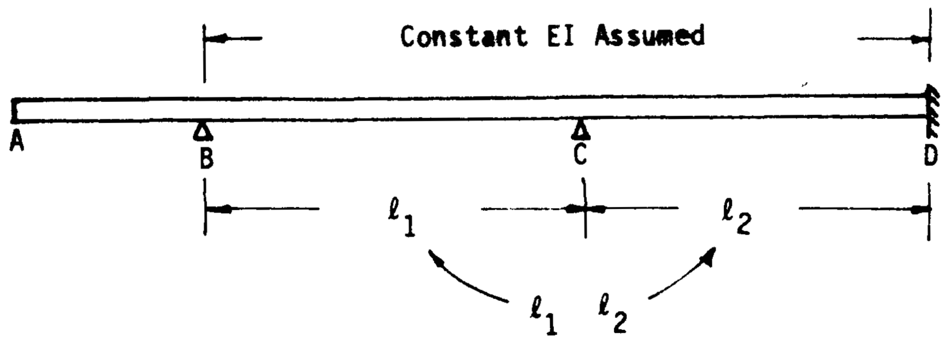
Element CD



Fixed End Bending Moment

Figure 38. Shears and Bending Moments (Cont.)

TABLE 14. MOMENT DISTRIBUTION (NO AXIAL LOADS)



Stiffness Ratio*			.429	.571	
Fixed End Mom.	+50354	-45236	+48961	-41009	+41009
Release B		-5118	-2559		
			+46402	-41009	
Release C			-2316	-3079	.1540
Final Moments	+50354	-50354	+44086	-44088	+39469

*Note that since $\ell_1 = \ell_2$ stiffness ratio for BC = .429 and for CD ratio = .571 (Reference 13, Section V, Subsection 3.53, Case 5)

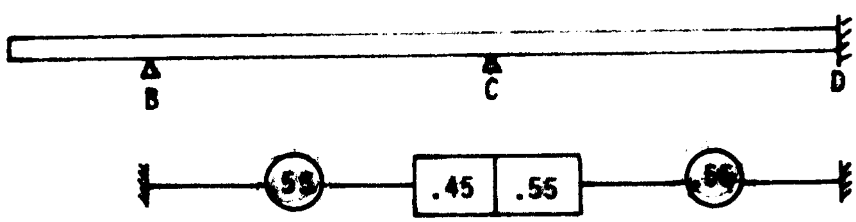
The presence of flying wires in the aircraft structure induces axial loads in the wing spar and these affect both shears and bending moments. Given the bracing geometry shown in Figure 35, these effects may be calculated. These results may be used to estimate EI for the spar. Table 15 summarizes these calculations. Figures 39 and 40 present wing normal bending moments and resultant

TABLE 15. WING EI SUMMARY AND MOMENT DISTRIBUTION WITH AXIAL LOADS

SUMMARY

ELEMENT	C	4 EI/L	$K = \frac{4\,EI}{L}C$	$K/\Sigma K$
BC	.93	10,900,775	10,138,700	.45
CD	.93	13,170,542	12,248,604	.55
		$\Sigma =$	22,387,304	

MOMENT DISTRIBUTION (INCL. AXIAL LOADS)



Fixed End Moment	+50354	-49009	+49009	-40779	+40779
Release B		-1345	-740		
			+48269	-40779	
Release C			-3371	-4120	-2266
Final Moments	+50354	-50354	+44898	-44899	+38513

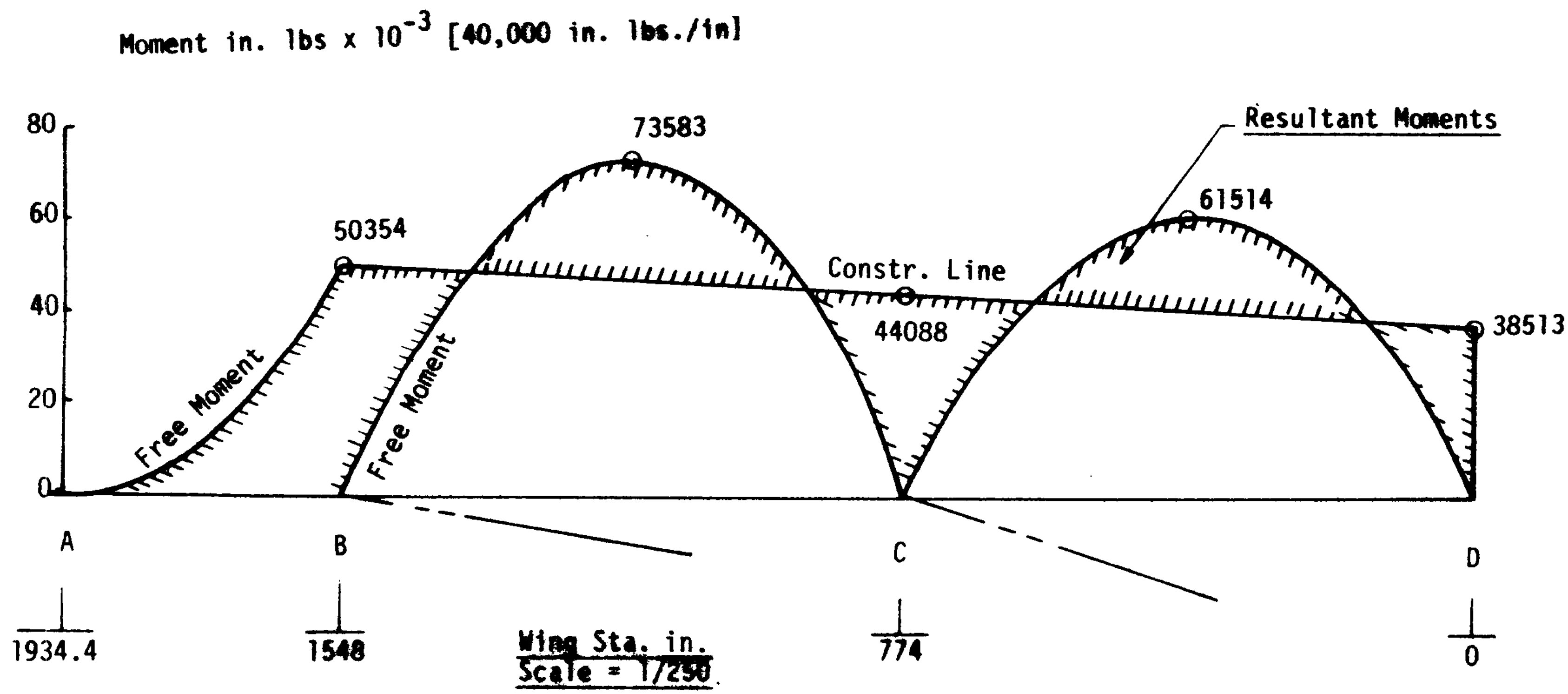


Figure 39. Wing Normal Bending Moments @ n = 3.0
[See Figure 40 for resultant moments]

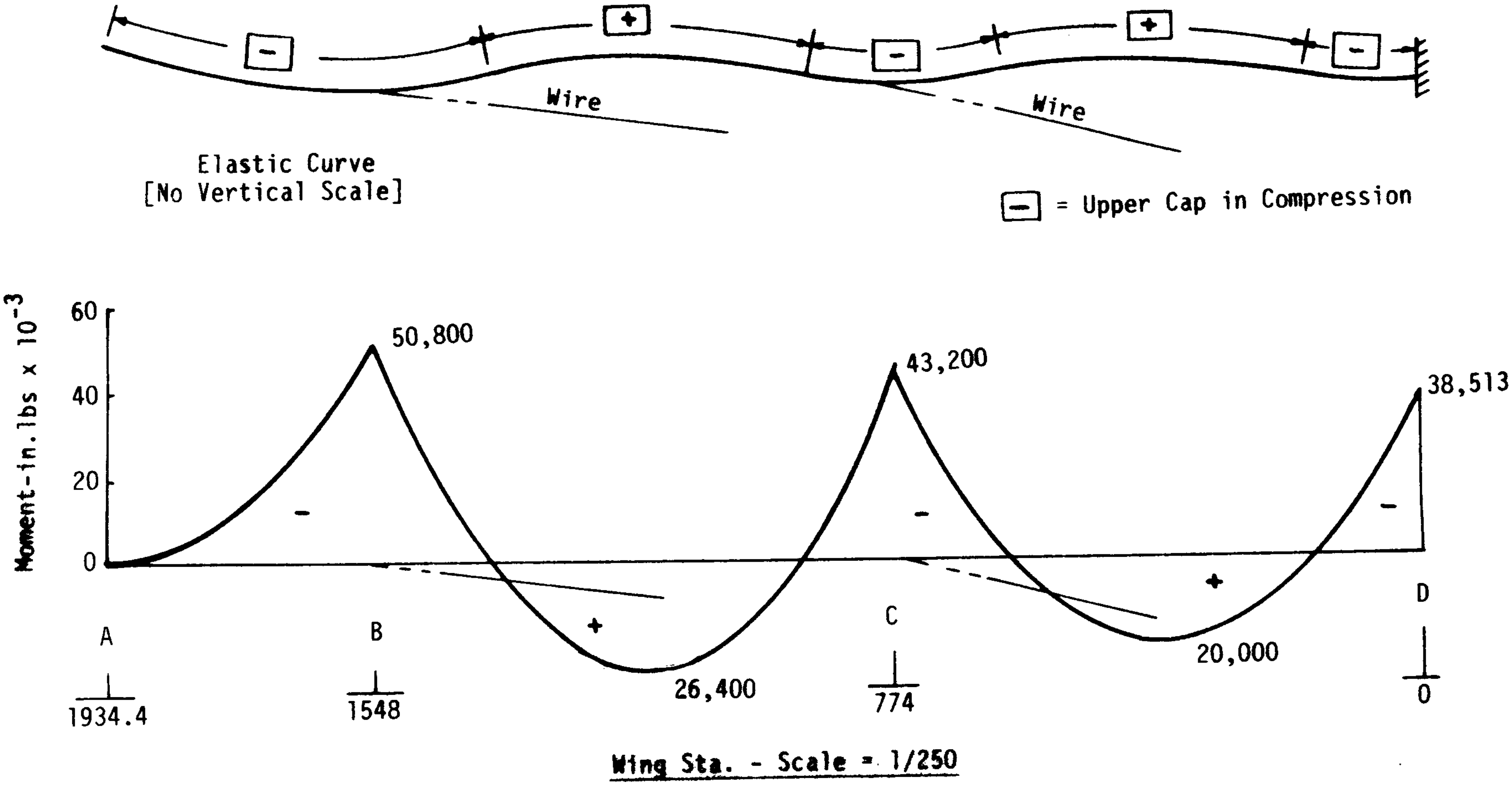


Figure 40. Resultant Normal Bending Moments @ n = 3.0
[Re-plot of Figure 39 graph with construction lines used as zero ref., with signs reversed]

normal bending moments, respectively. With these calculations in hand, normal wing shears may be estimated and these are presented in Figure 41.

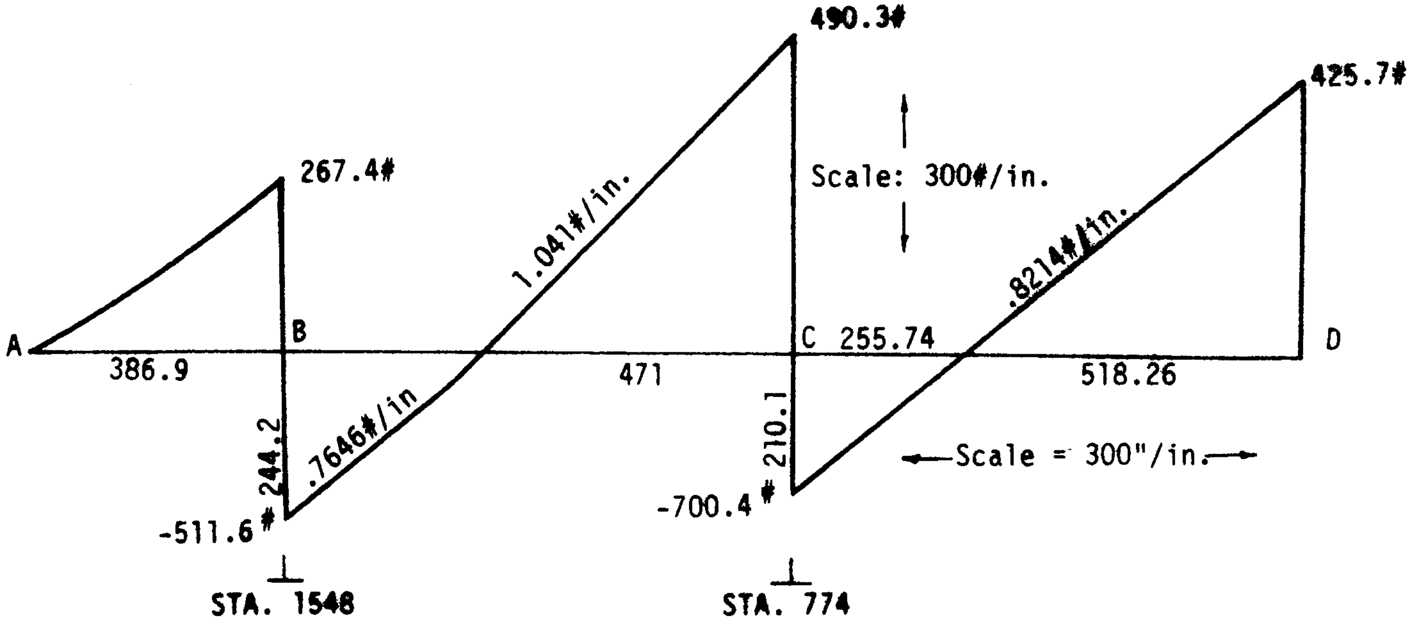


Figure 41. Wing Shear Diagram [normal]

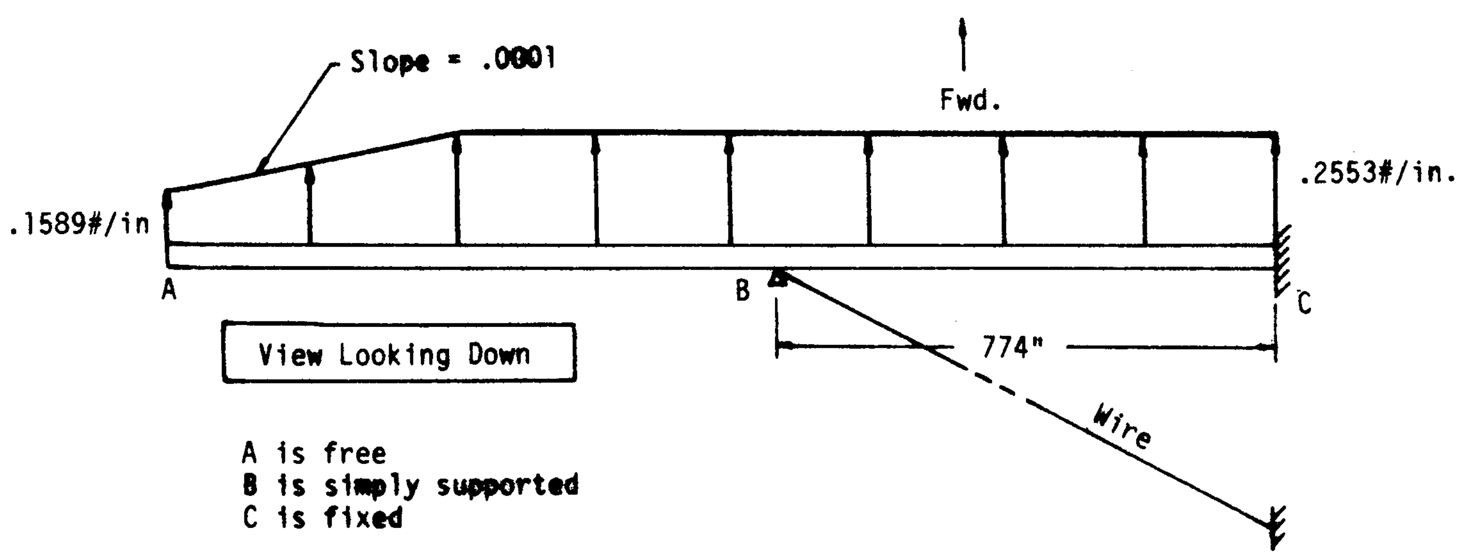


Figure 42. Distribution of Chordwise Shear Loads Along Span

Wing chord moments due to drag may be calculated. The chord load on the wing spar is shown in Figure 42. Chordwise shears and bending moments may then be calculated and these translated to normal and axial loads in the spar. The resultant wing chordwise bending moments are presented in Figure 43,and Figure 44 presents the resultant chordwise bending moments. Chordwise wing shears may be calculated as before and a chord shear diagram (Figure 45) can be constructed.

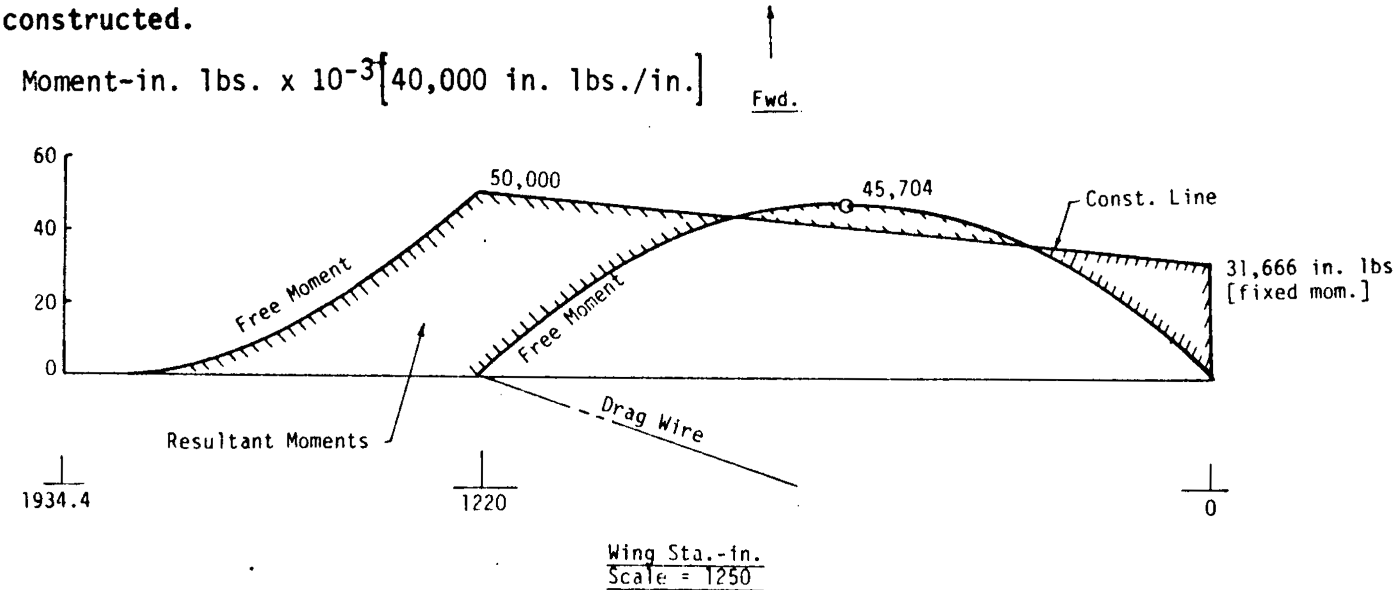


Figure 43. Wing Chordwise Bending Moments
[See Figure 44 for resultant moments]

Next, lift and chord loads in the wing spar truss members may be calculated for selected station members as with the other two bracing schemes. Net loads may then be put together and spar cap sizes may be determined. Figure 46 summarizes the cap sizes chosen. From this, diagonals and verticals may be chosen and spar weight calculated. Table 16 summarizes wing spar weight.

Finally, lift and landing wires may be sized and their weight estimated. Using the same values for non-spar items in the wing then produces the wing weight summary given in Table 17.

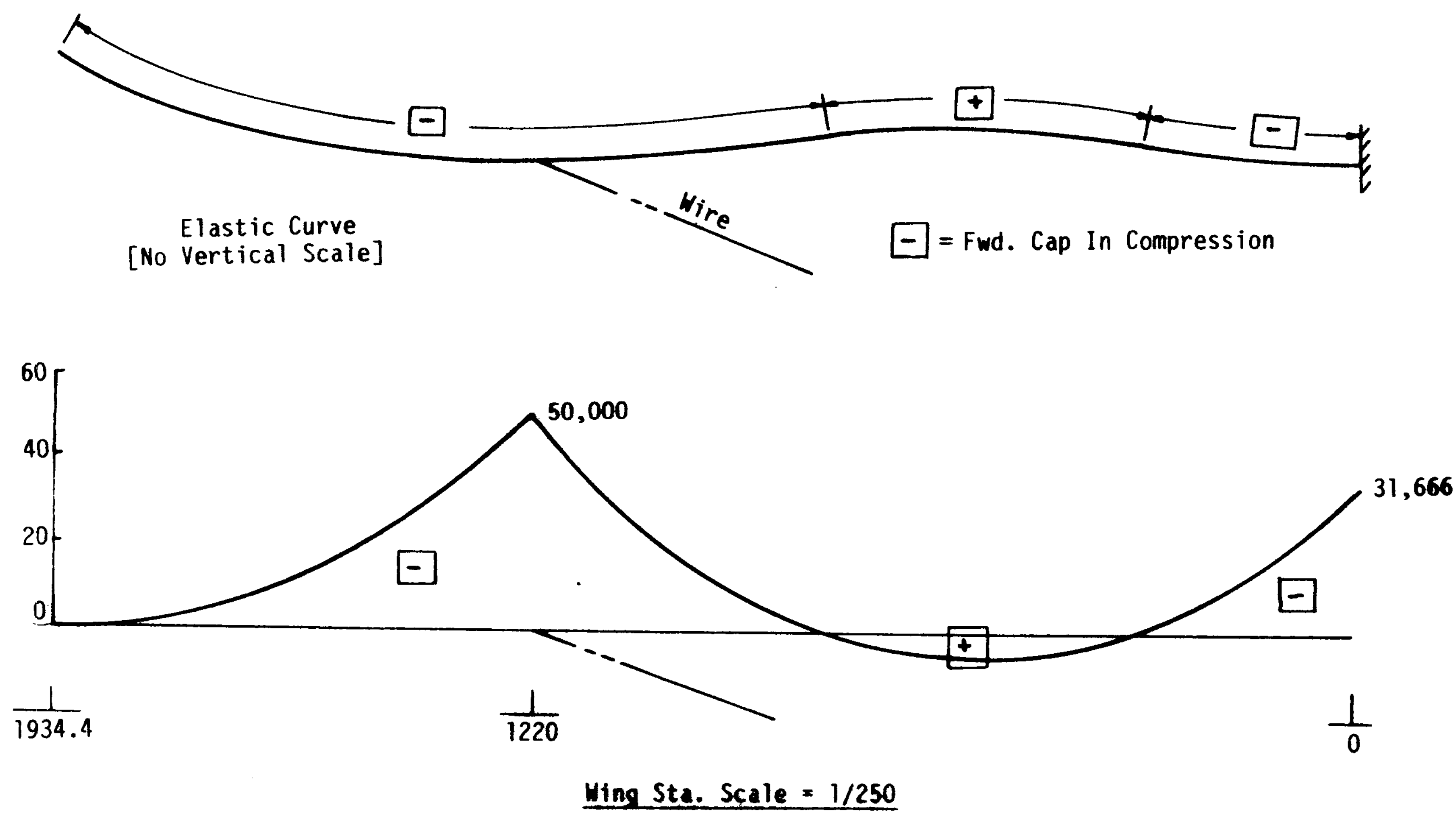


Figure 44. Resultant Chordwise Bending Moments
[Re-plot of Figure 43 graph with construction lines as zero ref., with signs reversed]

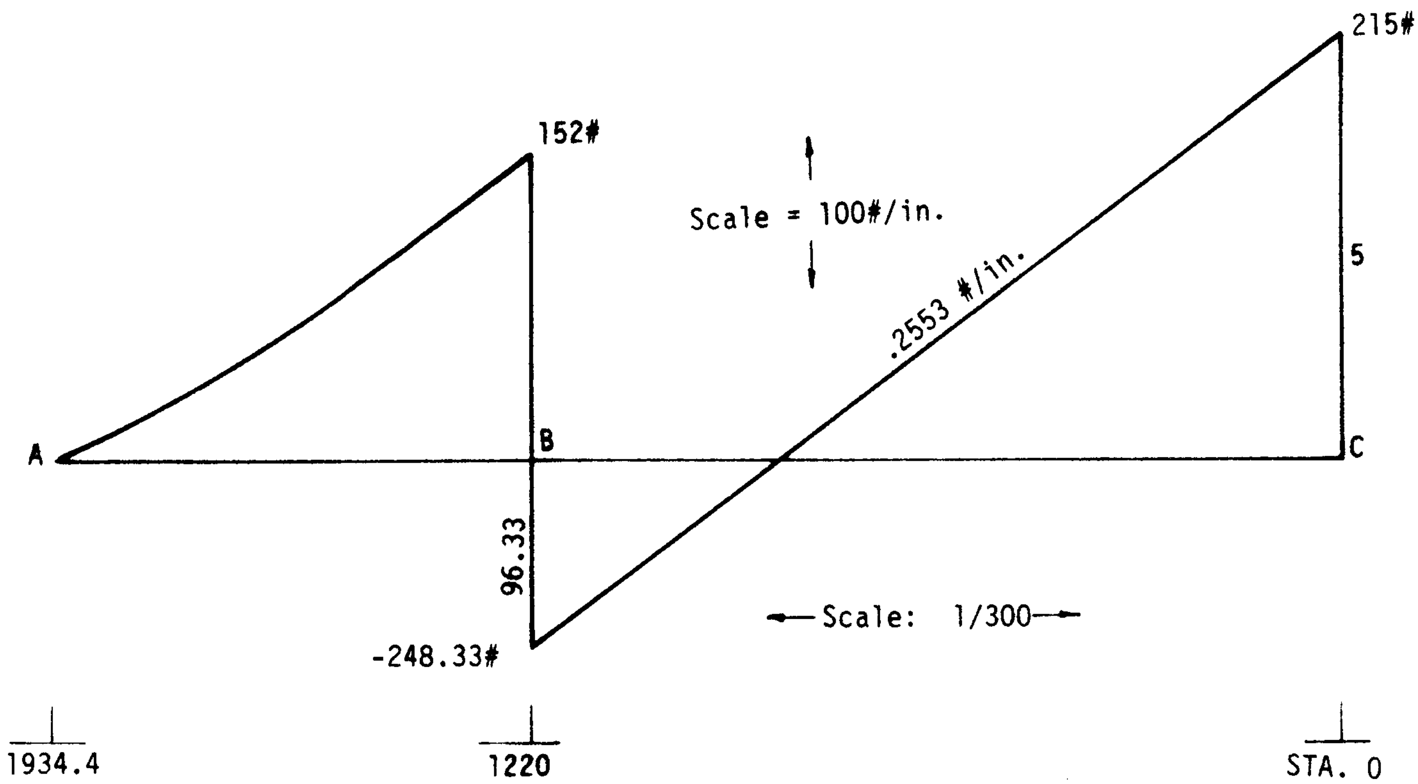
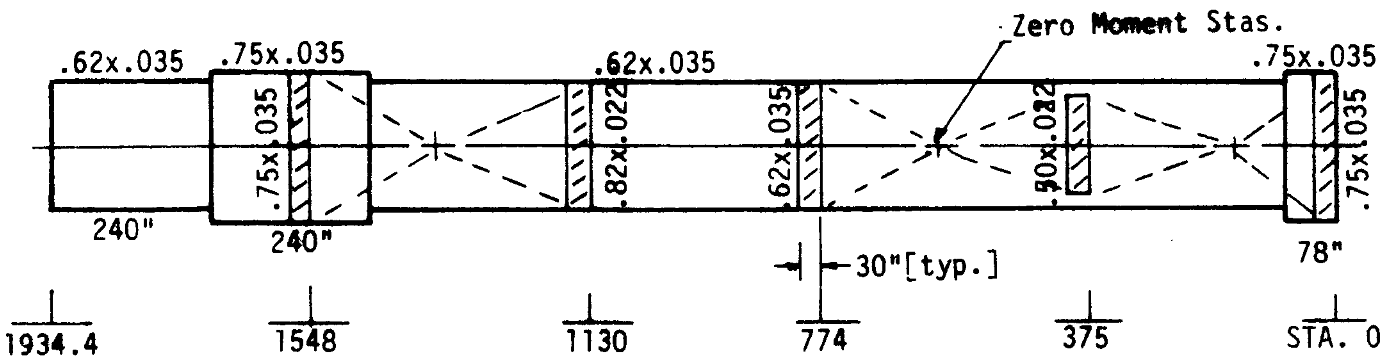


Figure 45. Chord Shear Diagram

Upper Caps



Wing Sta.

Shaded Areas = Calculated Cap Sizes

Solid Lines = Practical Cap Sizes [Caps can't go to zero dia. @ zero moment stas]

Lower Caps

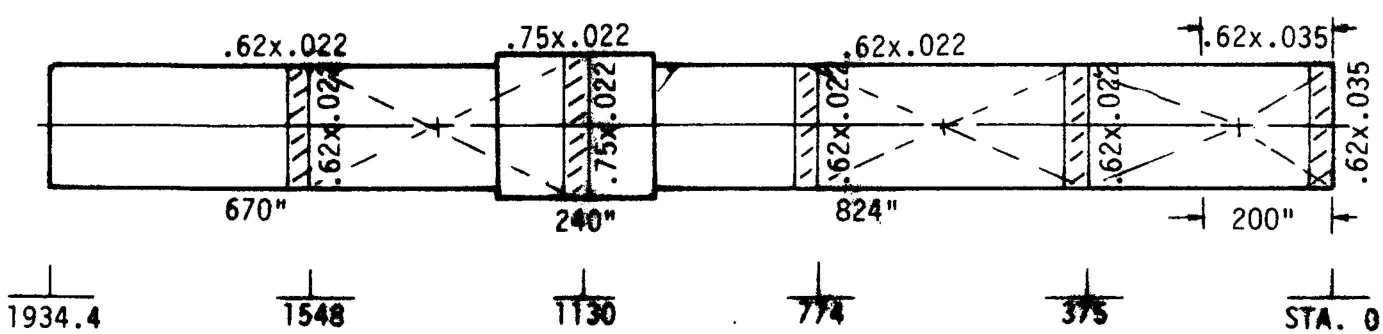


Figure 46. Summary of Spar Cap Sizes [Scale: Diameter, full; length 1/300]

TABLE 16. SPAR WEIGHT SUMMARY

ITEM	WEIGHT
UPPER CAPS	8.12 LBS.
LOWER CAPS	5.39
VERTICALS	2.38
LIFT TRUSS DIAGONALS	5.38
CHORDWISE MEMBERS	2.38
DRAG TRUSS DIAGONALS	5.38
WIRE ATTACH STR. [EST]	1.50
TOTAL	= 30.53# For 1 Truss

Wt. of both trusses, incl. 15% for joints & misc.:

= 1.15 [2 x 30.53]

= 70.22#

Total weight of spars for both wings = 2 x 70.22 = 140.44 lbs.

TABLE 17. WING WEIGHT SUMMARY [BOTH WING PANELS]

ITEM	WT.-LBS	WT. FRACTION [OF WING]
SPAR TRUSSES	140.44	.2050
RIBS	159.90	.2335
L.E.&T.E.	124.10	.1812
AILERONS	27.00	.0394
SPOILERS & STRUCT.	24.66	.0360
LIFT, LDG. & DRAG WIRES	10.00	.0147
FABRIC & DOPE	129.20	.1886
FIXED SOLAR PANEL	69.62	.1016
TOTAL	684.92	1.0000

A sample calculation for the points shown in this figure will be presented in a moment. First, the following assumptions which went into these calculations should be noted:

Sizing Algorithms

Variations of Aspect Ratio. The intent of the preceding analysis of three different bracing schemes for one aircraft configuration was to provide comparable baselines for examination of the effects of changes in design parameters on structural weight. This was done by choosing several different values of each parameter and recalculating wing weight based on its change. Trends could then be examined and generalized expressions could be developed.

The first parameter to be investigated will be aspect ratio (AR). The dominant effect of aspect ratio changes will be on wing spar weight,but other items of wing structure may be affected, too. The basic approach will be to apply a given load at the geometrical a.c.* of constant-chord wings of varying aspect ratio and determine the upper spar cap tube size required to handle the resulting column load in each. Spar weight will be closely proportional to spar cap area.

Bending moment for an aspect ratio = 10 wing could be set to correspond to a column load capability of 1.00 inch O.D. x 0.049 inch wall composite tube 30 inches long. From this moment, a wing loading could be chosen assuming total reference wing area is 1000 square feet and the load derived therefrom applied to each wing. Next, a spar cap tube could be designed that will handle the moment thus developed, with minimum margin of safety. Required tube area can then be plotted against aspect ratio. The weight of the spar for each aspect ratio will be some multiple of the aspect ratio = 10 weight, the multiplying factor being represented by the plotted curve in Figure 47.

* aerodynamic center

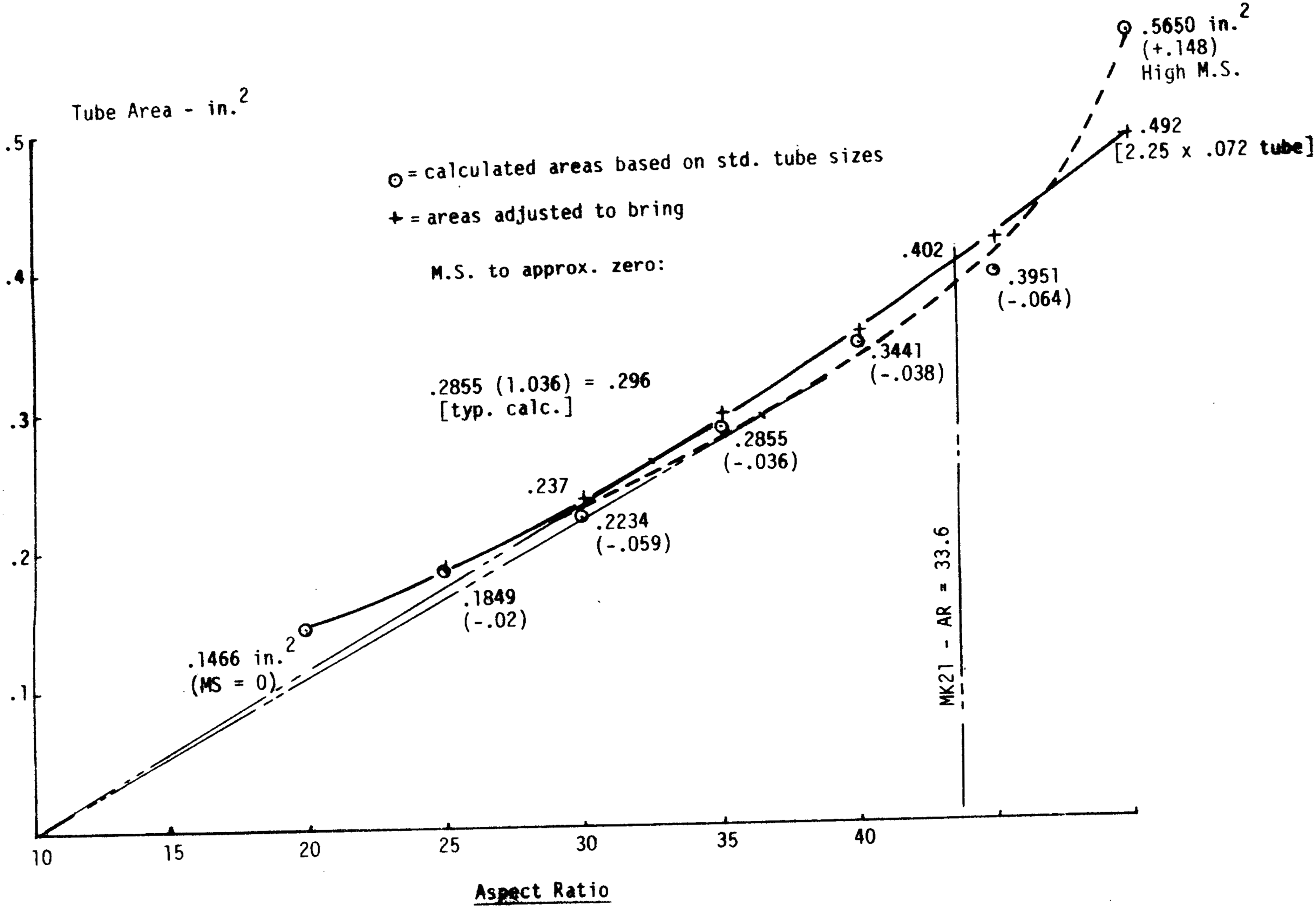
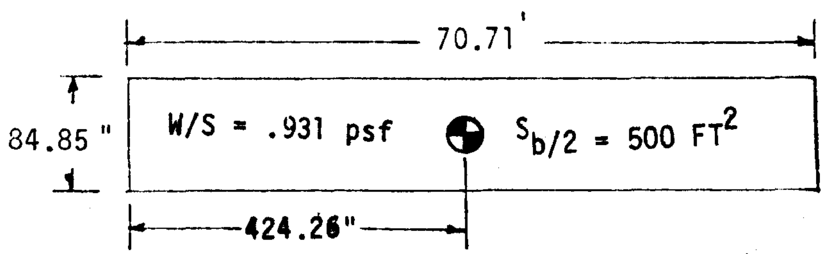


Figure 47. Plot of Spar Cap Tube Area Vs. Aspect Ratio

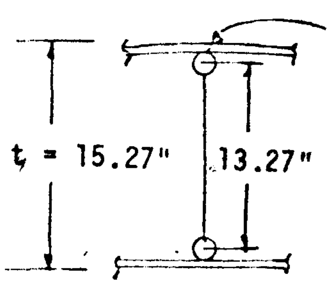
- Wings all have 18 percent thickness-to-chord ratios and the spar is set @ t/c_{MAX};
- Spar cap tubes are all 1/4" below flush with the wing surface (to allow for 1/4" rib caps);
- Column length of tubes is 30 inches;
- Only lift loads on the wing are considered;
- Wing area is 1000 sq. ft. in all wings; and
- Tube end fixity (c) = 1.5.

Given a sample wing geometry as below, the column load may be calculated, a tube size determined and its resultant margin of safety estimated.

AR 20



$$M = .931 \times 500 \times 424.26 = 197{,}493 \text{ in. lbs.}$$



1.5 x .049 Tubes

$$t_{MAX} = .18 \times 84.85 = 15.27"$$
$$h = 15.27 - 1.5 - .5 = 13.27"$$

$$\text{Column Load} = \frac{197493}{13.77} = 14{,}883\#$$

150 x .049 Tube: $A = .2234 \text{ in}^2$ $\rho = .5133$

$$L'/\rho = \frac{24.49}{.5133} = 47.71 \text{ [short column]}$$

$$F_C = 80{,}000 \left[1 - .3027 \left(\frac{47.71}{59.73}\right)^{1.5}\right]$$

$$F_C = 62713 \text{ psi} \quad P_{Allowable} = 62713 \times .2234 = 14010\#$$

$$M.S. = \frac{14010}{14883} - 1 = -.059$$

Aspect ratios from 10 to 45 were considered. Two points are plotted for each of the aspect ratios chosen. The first assumes the standard tube nearest in size to a margin of safety of zero. In every case, the tubes chosen have slightly negative margins of safety (for AR = 20, the margin of safety is -5.9%). These points fall on or close to the dotted line. If tube area is adjusted to bring the margin of safety to approximately zero, then the points fall on the solid line. Two points are of interest, one on each curve. The first occurs around aspect ratio 20 on the zero margin of safety line and corresponds to the point of diminishing returns where tube size goes up faster than aspect ratio. The second is the corresponding point on "nearest real tube size" line at aspect ratio 27.

Assumptions were also made to estimate the effect of aspect ratio on the weight of wing components:

- All ribs are assumed to be made of spruce with 1/4 inch square members. The weight of a rib at any aspect ratio, then, will be proportional only to wing chord;
- Leading edge material for all aspect ratios will be made of the thinnest plywood available;
- Metal trailing edges come in standard sizes with weight a function of trailing edge length;
- Fabric covering is a function only of wetted area which remains constant for all wings considered.

In effect, the second and third assumptions link wing component weight to wingspan by the relation below:

$$AR = \frac{b^2}{S_{REF}}$$

$$b = \sqrt{AR \cdot S_{REF}}$$

If S_{REF} is constant (last assumption), then

$$b \sim \sqrt{AR}$$

and weight of any component will be

$$\text{Weight at Desired AR} = (\text{Weight Calculated at AR} = 33.6) \times \left(\frac{AR\ 33.6}{AR_{DESIRED}}\right)$$

If weights are calculated for entire wings at various aspect ratios, an interesting phenomenon appears. Table 18 presents data to illustrate this point.

TABLE 18. COMPARATIVE WEIGHTS OF TWO WINGS OF DIFFERENT ASPECT RATIO

ITEM	AR=33.6 (MK21 WING)	AR=20 WING	CHANGE
SPAR, INCL. WIRES	150.44#	89.52#	1.681
RIBS	159.90	159.90	1.000
LEADING EDGE	102.40	132.73	0.771
TRAILING EDGE	21.70	16.74	1.296
AILERON RIBS	12.18	15.79	0.771
AILERON SPAR	8.26	8.26	1.000
AILERON T.E.	6.56	4.89	1.342
SPOILERS & STRUCT.	24.66	24.66	1.000
FABRIC & DOPE	129.20	129.20	1.000
SOLAR CELLS	97.20	97.20	1.000
	712.5#	678.89#	1.050

The conclusion to be drawn from this table is that, even though spar weight will vary markedly from aspect ratio 20 to aspect ratio 33.6, total wing weight will increase only 5%. This small change in total wing weight for a 68% change in aspect ratio is due to the lack of dependence of most wing structural components on aspect ratio and the small fraction of spar weight to wing weight to begin with.

STRUCTURAL WEIGHT ESTIMATION

It is one of the objectives of this follow-on report to derive a set of equations for preliminary weight analysis of this class of aircraft. From previous studies it has been determined that this class of aircraft falls somewhere between human powered aircraft (HPA) and light wing loading sailplanes, in terms of structural weight. So, to derive the empirical equations desired, those two areas were used as sources of weight data and weight estimation equations.

The detail level that is expected to be known about a particular aircraft has determined the form and accuracy of the equations presented here. It has been determined that the known factors would be gross weight, wing area and span, tail volume coefficient, airfoil thickness ratio, and flight dynamic pressure. In addition to these factors, methods of construction, types of materials used,and ultimate load factors are also assumed to be known. To help in deriving the equations the following constraints were placed on the aircraft configurations.

	MIN	MAX
1. Aspect ratio	10	35
2. Wing loading	0.5	1.5 lbs/ft^2
3. Gross weight	1000	3000 lbs

The weight estimation equations arrived at are presented here in four groups: the wing, fuselage, tail surfaces and propeller. The equations are expected to produce error no greater than $\pm$ 15 percent for the given restrictions.

The Wing

To arrive at a reasonably accurate wing weight, the wing was divided into six subgroups. Those groups are the spar, leading edge, trailing edge, ribs, covering, and controls.

The spar weight can be derived from Figure 47 as:

$$W_S = 0.12114\ K_1\ (K_2AR)^{0.9}\ K_3\ W_G\ (^n/3)^{0.2n}$$

where

K_1 = 1.0 for a wire braced wing and 1.25 for a cantilever or strut braced wing

K_2 = 0.011

K_3 = 1 + 0.008AR

For the leading edge the weight was found to vary as:

$$W_{L.E.} = 0.0332 \left(\frac{33.6}{AR}\right)^{0.5} \cdot S$$

and the trailing edge weight can be described simply as

$$W_{T.E.} = K_{TE} \times b$$

where

K_{TE} = weight of T.E. material per unit length

The variation of leading edge weight with aspect ratio and wing area is shown in Figure 48. In a fashion similar to the trailing edge, the covering weight can be found by multiplying the per unit weight of the covering material by the wing surface area with a correction factor included for wing thickness. This factor must be included because, for this type of wing the airfoil is quite thick causing a higher requirement for covering than just twice the wing area. So, the resultant equation is:

$$W_C = K_C\ (2S + 1/2\ \frac{t}{c}\ b)$$

where

K_C = weight per unit area of covering

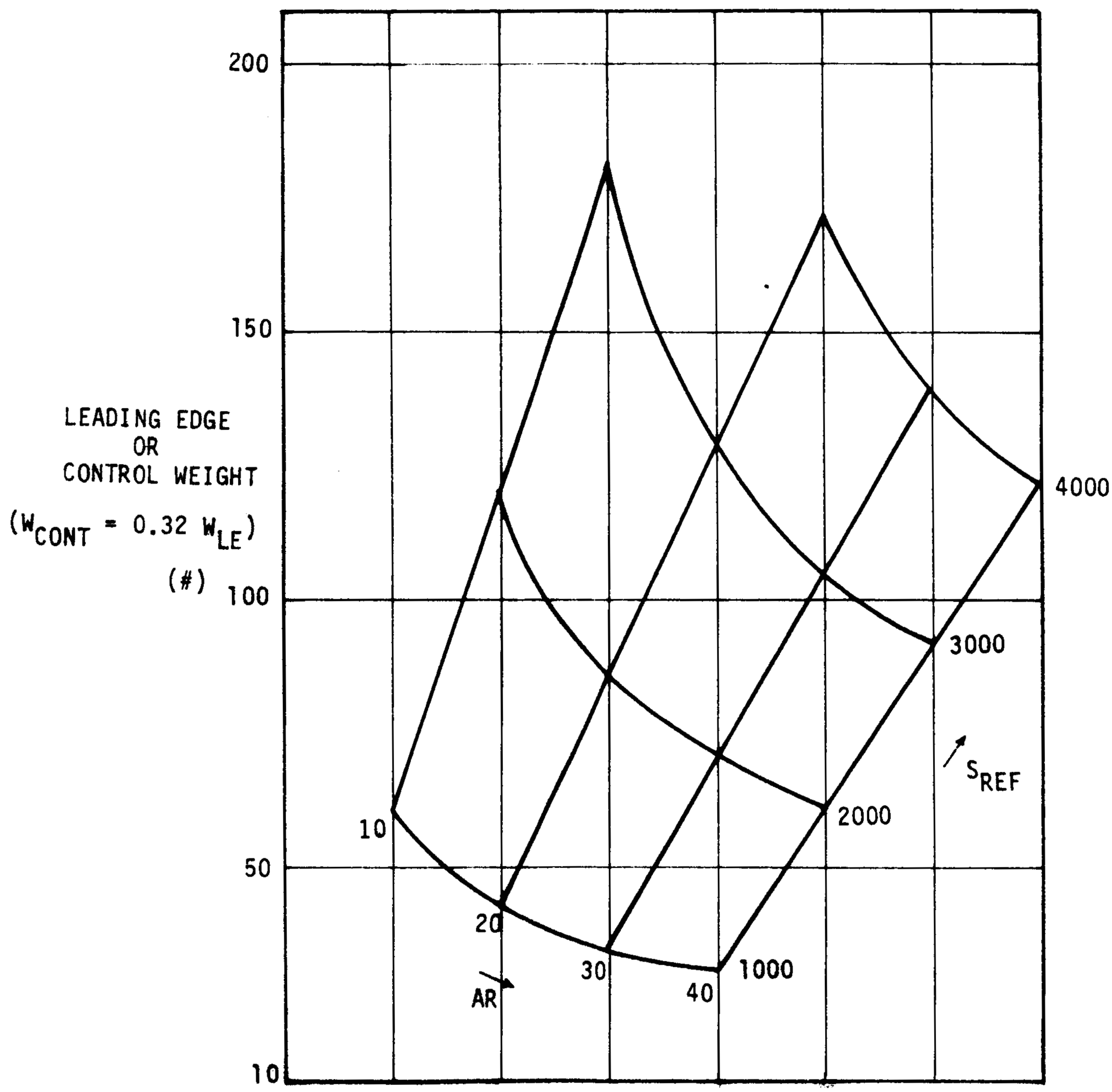


Figure 48. Leading Edge and Control Weights Vs. Aspect Ratio and Wing Area

The rib weight can be given as

$$W_R = K_R \ (1/2S + 1/2 \ (S \ \frac{t}{c}))^{0.6}$$

where

$$K_R = 1.0 \text{ for wood ribs and } 0.75 \text{ for composite ribs}$$

This equation assumes a constant chord wing section. For tapered sections, the result should be multiplied by a factor of 0.9. The final consideration of the wing section is the controls. The control weight has been found to vary as:

$$W_{CONT} = 0.0106 \left(\frac{33.6}{AR}\right)^{0.5} (S)$$

This equation is also plotted in Figure 48 and differs from the leading edge weight by a factor of 0.32. All of the above weight equations, except for the ribs and covering, have been derived from a detailed, parametric, study of wing component weights for varying aspect ratios. This detailed analysis was done as a part of this contractual study. The equations for the covering and ribs are modified equations used for HPA work.

The Fuselage

Under this study, a detailed weight work-up was done for only one fuselage design, a pod and boom type. Given this, the equation derived is for that type only and is based on wing loading and flight dynamic pressure. The resulting equation is:

$$W_{FUS} = \left[\frac{0.2 \ W \times n}{S}^{0.8}\right] \left(\overline{q} \ . \ S\right)^{0.9}$$

The variation of fuselage weight with wing area and dynamic pressure is shown in Figure 49. For this plot, W_G = 1758 lbs and n = 3. HAPP landing gear weight is based on sailplane landing gear and varies with the gross weight as:

$$W_{SK} = \frac{W_G^{\,1.1}}{150}$$

This equation is plotted in Figure 50.

The Tailplanes

Assuming that both the vertical and horizontal tails employ the same construction methods, one weight equation can be given for both surfaces.

That equation is:

$$W_{TP} = 2N\left[\left(\frac{W_G\ n\ K_{TP}}{105}\right)^{0.87}\left(\frac{\ell S_{TP}}{C\ S}\right)^{F_{TP}0.6}\right]$$

where

N = number of tail surfaces (2 vert., 3 vert., 1 horiz., etc.)

K_{TP} = 2/3 for wire bracing or clamped - clamped beam ends

ℓ = tail moment arm

S_{TP} = tailplane surface area

F_{TP} = tail covering factor (1.0 for fabric and dope, 1.2 for mylar)

This equation is a modified version of the ones given in Reference 9 They were modified so the tail volume coefficient would appear in the equation. It should be noted that the above equation includes controls and is graphed in Figure 51 with n = 3, F_{TP} = 1.0, N = 1 and K_{TP} = 1.

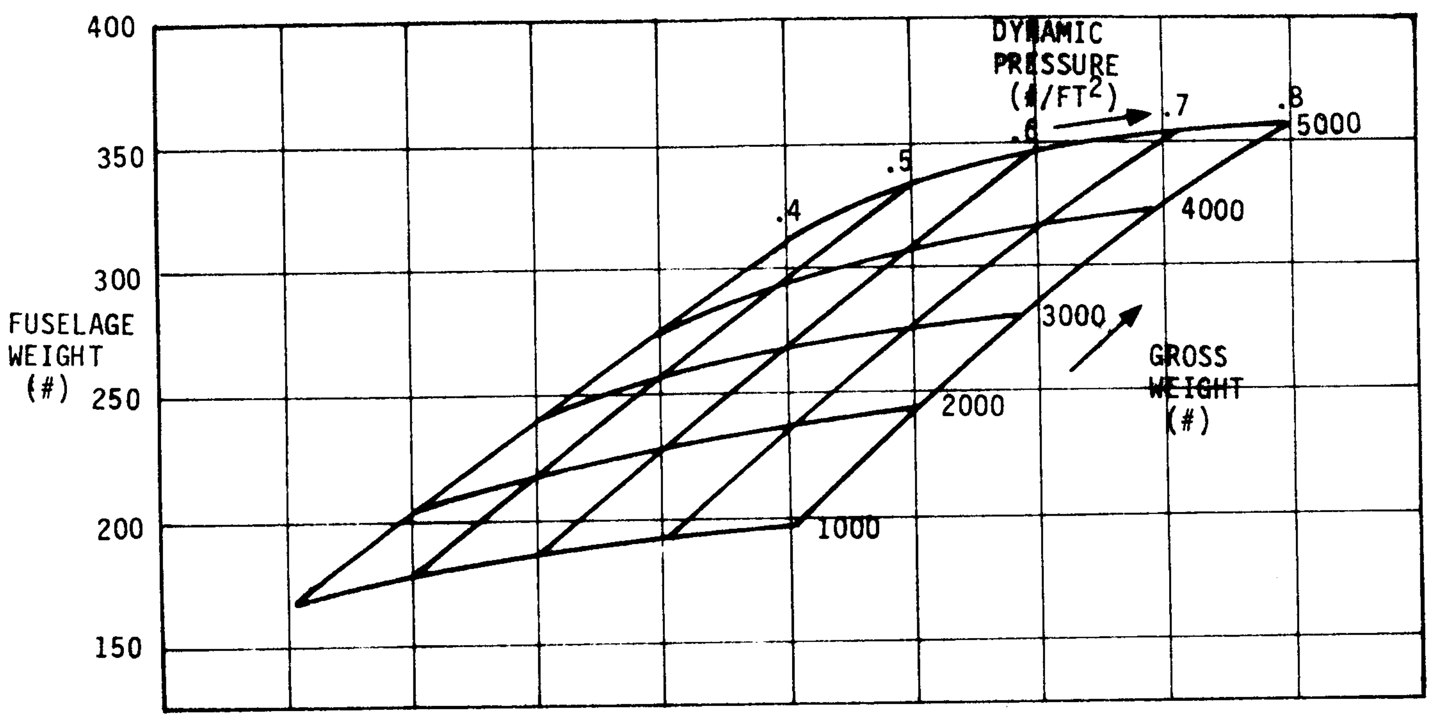


Figure 49. Fuselage Weight Vs. Dynamic Pressure and Wing Area

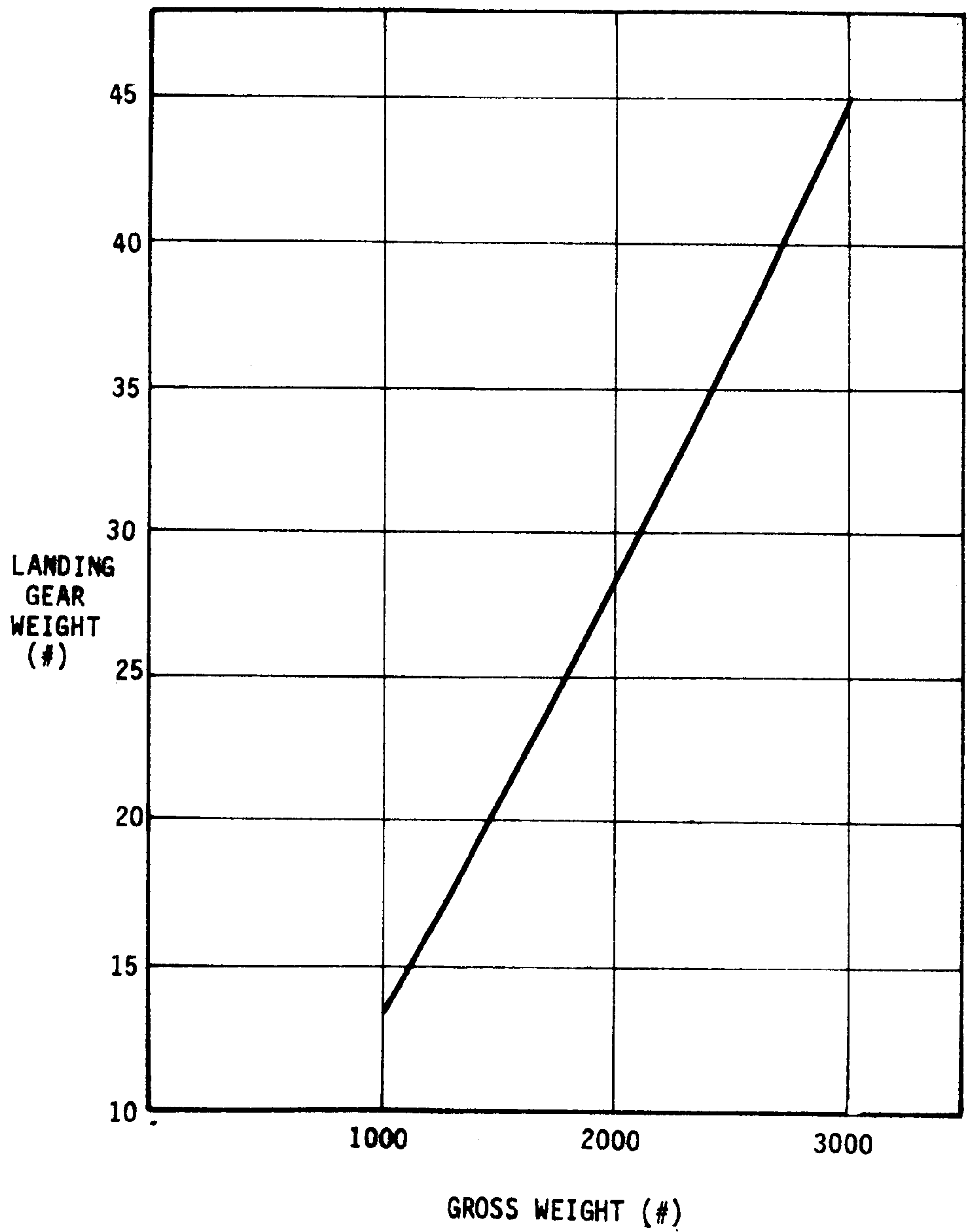


Figure 50. Landing Gear Weight Vs. Gross Weight

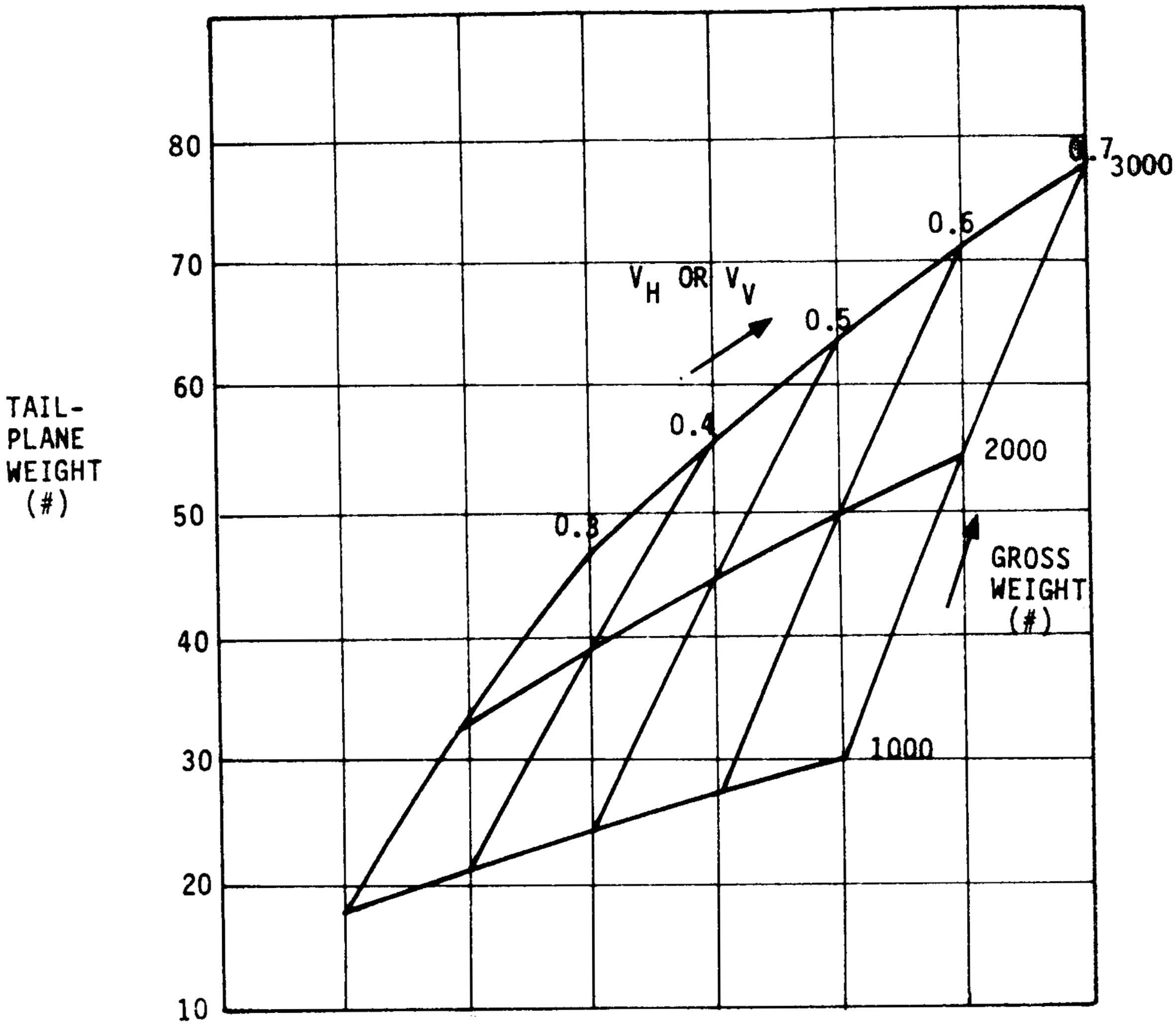


Figure 51. Tail Plane Weight Vs. Gross Weight and Tail Volume Coefficient

The Propeller

Based on the work in Reference 5, the following propeller weight equation was derived. The propeller weight is based on wing loading as follows:

$$W_p = 100\,(W/S)^{0.5}$$

and is plotted in Figure 52.

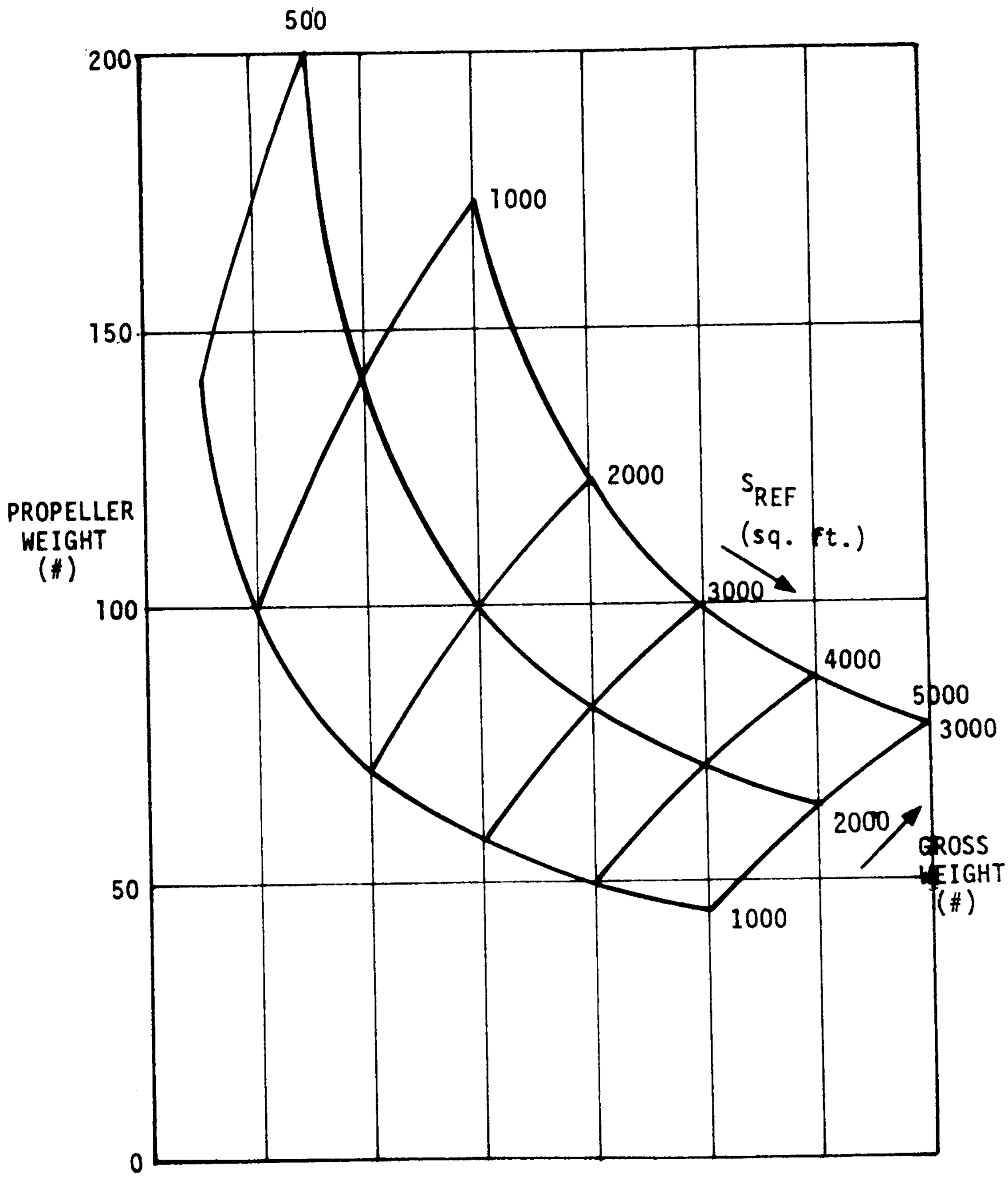


Figure 52. Propeller Weight Vs. Wing Loading

APPENDIX B

WOOD AS AN ENGINEERING MATERIAL IN THE MK21 VEHICLE

The design of the MK-21 vehicle calls for using the most structurally efficient materials available, now or in the near term. Due to the unusually rigid requirements for low vehicle weight, structural efficiency of the MK-21 is based upon strength per unit weight.

Few, if any, materials can match the graphite/epoxy composite on those terms, and for that reason graphite/epoxy, which is very stiff, comprises the primary structural member of the wing, the spar.

For other structures, however, the loads are so low and/or the requirement that they be flexible enough to bend to given shapes so great as to rule out graphite/epoxy. Such structures are the wing and tail ribs, the wing leading edge and fairing strips and formers on the pylon and pod, all substantial contributors to the overall weight.

These structures are made of wood, because to make them of anything else would be to impose unnecessary weight penalties and, very likely, unnecessary penalties in manufacturing cost.

The wing leading edge is a case in point. As configured in this study the leading edge comprises a D-tube of .016 inch thick birch plywood and 1/4 inch square spruce corner strips. This structure, which is 322.2 feet in length, weighs 102.4 pounds, or about 5 ounces per foot. If the structure were made of 2024T3 aluminum alloy of the same thickness (which is the thinnest structural aluminum alloy sheet made) it would weigh 365.7 pounds, or 3.57 times as much. It would, in fact, weigh more than twice the weight of the spar, and the loads on the leading edge are essentially nonexistent.

The loads on the wing and tail ribs are also very low. A weight comparison of ribs made of several candidate materials was made in the MK-10 study. This study showed the superiority of a truss made of spruce strips and plywood gussets, much in the manner of ribs used in light training and pleasure aircraft of an earlier vintage.

The accompanying table shows the comparative structural efficiences of spruce, birch plywood, 2024T3 (clad) sheet and graphite/epoxy. Observe that spruce beats 2024T3 in all but stiffness (and weighs only 1/7 as much) and that birch plywood beats 2024T3 in both column and shear buckling efficiency - and weighs about 1/4 as much.

The superiority of graphite/epoxy shows clearly in the table. However, the material is simply too stiff for applications requiring flexibility in manufacture - as wing leading edges and ribs, for example.

The above paragraphs are offered because, although the acceptance of new materials by design engineers is sometimes difficult, it is frequently more difficult to draw their attention to the fact that on a case by case basis, some "old" materials have better application than the new ones - and the MK-21 solar HAPP is seen as one of those applications.

TABLE B-1

COMPARATIVE WEIGHTS & STRUCTURAL EFFICIENCIES OF MATERIALS (1)

Material	WT. LBS/in.^{3}	Tension EFF. F_{T_U}/W x 10^{-3}	Shear Column EFF. $\sqrt{EC/W}$ x 10^{-3}	Buckling EFF. $\sqrt[3]{EC/W}$ x 10^{-2}	Stiffness E_C P51 x 10^{-6}
Spruce	.015	626	79	—	1.4
Birch Plywood	.028	307	39	38	1.2
2024T3 Al. Alloy (CLAD)	.100	600	32	22	10.7
(2) Graphite/ Epoxy [uni]	.061	1573	104	56	40.0

(1) Ref. NASA CR-1285 "Potential Structural Materials And Design Concepts For Light Aircraft".

(2) Lockheed California Division Data.

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16. Abstract

This study was conducted to develop sizing algorithms for very lightweight aircraft structures. Three types of bracing schemes were analyzed: Fully cantilevered strut bracing and wire bracing and scaling rules were determined. Wire bracing appears to provide the lightest wing structure for Solar High Altitude Powered Platforms.

This report follows a more comprehensive study of Solar Powered Aircraft, NASA CR 3699, and is meant to provide an addition to the earlier work.

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