MR. GOVER

THE COLLEGE OF AERONAUTICS

CRANFIELD

ADDENDUM TO TYPE RECORD

Avro "Lancaster" Mk. 1 P. A. 474

Type Record No. 1/P1/Lancaster.

Investigation of Boundary Layer Conditions on Wings

M. O. S. Contract No. 6/Aircraft/9807/C. B. 6(a)

Department of Aircraft Design

January, 1956.

THE COLLEGE OF AERONAUTICS

DEPARTMENT OF AIRCRAFT DESIGN

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LIST OF CONTENTS

DESCRIPTION

Introduction

The investigation involves, briefly, the mounting of a wing of 45° sweep-back in a dorsal position on the fuselage of 'Lancaster' P.A.474 such that wing incidence will be variable in flight over a 20° range, i.e. ± 10°.

A 'partial chord' technique for swept wings has been established by the College using a similar wing mounting configuration on an Avro 'Anson' Aircraft and this technique has been applied to the current investigation.

Wing Dimensions

The equivalent wing chord is 130" but, using the above technique, it is only necessary to represent accurately 40% (52") of this. The actual wing chord is 86", the forward 52" representing 40% of the equivalent chord and the remaining 34" consisting of an elliptical fairing. Wing t/c ratio is 10% giving a maximum thickness of 13" at 40% equivalent chord (52" from the leading edge); span is 8 ft. and the total wing area, therefore, is 57.33 ft^2.

Wing Structure

The wing is a composite wood and metal structure, the decision to make it so having been influenced by the necessity to achieve a very high degree of surface finish coupled with a desire to minimise jigging problems. Basically, therefore the wing consists of a conventional metal spar having birch ply bonded to each face of the web, spruce leading and trailing edge beams, close-pitched wooden ribs and a 16 s.w.g. metal skin, the inside face of which also carries a birch ply bonding. This structure was assembled using normal glueing technique and, during assembly, an internal humidity seal was achieved on the wooden parts by spraying with 3 coats of 'phenoglaze' G.300. The leading edge, trailing edge and capping members are mahogany, again 'Phenoglaze' coated, and the leading edge is detachable to allow of experimentation with a variety of nose radii. Two heavy steel joint plates are used to carry the spar boom end loads into the root rib (rib no.1) and the wing spar extension. The root rib itself consists mainly of thick ply-wood side beams connected to the rib boom via ply bonded 16 s.w.g. D.T.D. 710 skin top and bottom. These side beams are in turn connected to the main spar and the joint plates via welded flanges and 14 s.w.g. steel plates above and below the rib datum, which also act as doubling plates for the spar web at the root. The rib is blocked with ply-wood for'd and aft at the wing attachment fitting positions, and light alloy gusset plates are fitted to form a connection between the side beams. Transverse ply-wood shear diaphragms are also provided at these attachment positions. The wing design is such that no actual physical connection is made between the spar boom and the skin. However, in order to stabilise the boom under end loading conditions it is necessary to provide a local attachment of some description. This is achieved by means of 4 BA c's'k h'd bolts tapped into each spar boom flange at a number of rib positions, the skin contour being maintained by means of bakelised fabric distance tubes glued to the rib boom. This local attachment occurs at ribs no. 2,4,7,11 and 17. Pressure plotting holes up to 39% equivalent chord are provided at 3 spanwise positions and 3 additional spanwise pressure plotting stations are also provided on the leading edge only, designed in such a way that an infinite number of percentage positions may be plotted if desired.

Wing Mounting

A series of flight tests were carried out on the aircraft using a pitot mast mounted in a number of different fore and aft positions on the fuselage top. These tests were done in order to determine the airflow pattern along the fuselage and so decide upon the best position for wing mounting. Structural limitations, of course, were also considered and the final decision was made to mount the wing at former No.22. A further series of airflow tests were then carried out to establish a measuring datum in the vicinity of the former.

The wing is held in position via forward and aft attachment fittings - the forward fitting being integral with the incidence changing actuating jack, the actuating jack, in turn, being attached to the port side of the fuselage between formers 19 and 20. A vertical wing spar extension carries female hinge fittings which engage in male fittings mounted on former No.22 in the case of the upper, and on a channel bracket attached to the main floor in the case of the lower. The wing hinge line then falls 2.15" forward of the datum face of former No.22 A torsion box arrangement is incorporated on this spar extension to cater for the offset loads induced at the hinge points.

Fuselage Reinforcing

Side loading due to wing lift, including the appropriate gust factors, forms the major part of the induced wing loading which the fuselage will be required to withstand. These loads are conveyed directly into the main floor via two tubular struts with swaged ends whose attachment at former No.22 is integral with that of the top hinge fitting. End loading in the top skin is diffused round the necessary cut-outs between formers 19 and 20, 21 and 22, 23 and 24, by means of two channel section longerons running fore and aft between formers 18 and 24 and an 18 s.w.g. doubling skin also extending fore and aft between those two formers and, laterally, to stringer No.5, port and starboard. The cut-out positions are reinforced locally by box-section transverse members running between the longerons, coupled with a bearing plate which serves as a clamping base when wing incidence is set and will, in addition, take out the drag and inertia loading from the wing attachment fittings. Channel section torsion members are also installed at the top hinge fitting, running aft between formers 22 and 23.

<u>Instrumentation</u>

Consideration of the range of static and dynamic pressures expected, together with the headroom available has led to the choice of a 50 tube manometer some seven feet in height in which the manomatic fluid will be carbon-tetra-chloride. The manometer is mounted in the rear fuselage aft of former No.22 and carries, on a structure integral with it, a F.24 observation camera, the whole being vibration insulated by means of resilient mountings. A system of 'back lighting' is employed for photographic illumination.

A vane type yawmeter is mounted on the wing tip of the 'Lancaster' in order to give to the pilot an indication of aircraft attitude. This reads via a 'desynn' indicator mounted on the pilot's instrument panel. In addition to this, a differential pressure yawmeter is mounted on the tip of the swept wing and coupled directly to the manometer - giving a positive indication of swept wing attitude to the flight observer in the rear fuselage.

The above is a brief description of the swept wing installation. A detailed treatise is given in the subsequent pages of the type record addendum under the appropriate headings.

LIST OF COLLEGE OF AERONAUTICS MODIFICATIONS TO AIRCRAFT

Title	Drawings	Issued by
Modifications to Fuselage	M.167-1 M.170-2	C. of A.
Installation of Swept Wing	M.170-1	C. of A.
Installation of Actuating Jack	M.167-4	C. of A.
Installation of Wing Tip Yawmeter	M.167-7 A.E.R. 14460	C. of A. and R.A.E.
Alterations to Aircraft Instrument Panel	Embodied on M.167-7	C. of A.
Installation of Swept Wing Instrumentation	AD/51	C. of A.
Modifications to Oxygen System	M.167-9	C. of A.

LIST OF COLLEGE OF AERONAUTICS DRAWINGS ISSUED

rawing No.	Title	Issue	Date
M.160-1	G.A. of 45^{o} Swept Wing	5	19/12/55
M.160-2	Detail of Ribs	4	14/12/55
M.160-3	Arrangement of Main Spar	3	8/12/55
M.160-3A	Detail of Spar Web Plates	2	25/ 8/54
M.160-4	Spar Boom Joint Straps	3	9/12/55
M.160-5	Lower Hinge Bracket (Wing)	4	13/12/55
M.160-6	Upper Hinge Bracket (Wing)	4	13/12/55
M.160-7	G.A. and Details for Pressure Piping	3	14/12/55
M.160-8	Assembly of Boundary Layer Fence	2	10/ 6/55
M.160-9	Yawmeter and Mounting Structure on Swept Wing	1	16/ 6/55
M.160-10	Assembly of Wing Root Structure	1	16/12/55
M.166-1	G.A. of Actuating Jack	2	22/11/54
M.166-2	Details of Actuating Jack	2	22/11/54
M.167-1	Fuselage Reinforcing at Former 22	7	6/ 1/56
M.167-2	Detail of support tubes	2	28/ 2/55
M.167-3	Data Diagram	5	14/12/55
M.167-4	Installation of Actuating Jack	2	13/ 9/55
M.167-5	Upper Hinge Bracket (Fuselage)	3	28/ 9/55
M.167-6	Lower Hinge Bracket (Fuselage)	2	28/ 9/55
M.167-7	Installation of Wing Tip Yawmeter Pole	2	28/ 9/55
M.167-8	Stub End for Wing Tip Yawmeter Pole	2	28/ 9/55
M.167-9	Modifications to Oxygen System	1	12/ 8/55
M.167-10	Adaptor Plate	1	23/ 9/55
M.170-1	G.A. of Swept Wing Installation	4	12/12/55
M.170-2	Arrangement of Wing Mounting Structure and Fuselage Reinforcing	2	15/12/55
M.171-1	Wing Attachment Fitting (For'd)	3	8/ 9/55
M.171-2	Wing Attachment Fitting (Aft)	3	8/ 9/55
M.171-3	Assembly of Clamping Handle	1	26/ 1/55
M.171-4	Detail of Clamping and Lock Nuts	1	26/ 1/55
M.171-5	Detail of Bush	1	26/ 1/55
M.171-6	Special Bolt	2	8/ 9/55
AD/50	Multitube Manometer	1	29/ 8/55
AD/51	Installation of Multitube Manometer	1	14/10/55

SUMMARY OF STRESS ANALYSIS

A. WING SPECIFICATION AND DIMENSIONS

B. WING LOADING CASES

C. DETAIL STRESSING OF WING STRUCTURE

D. DETAIL STRESSING OF FUSELAGE MODIFICATION AND ATTACHMENTS

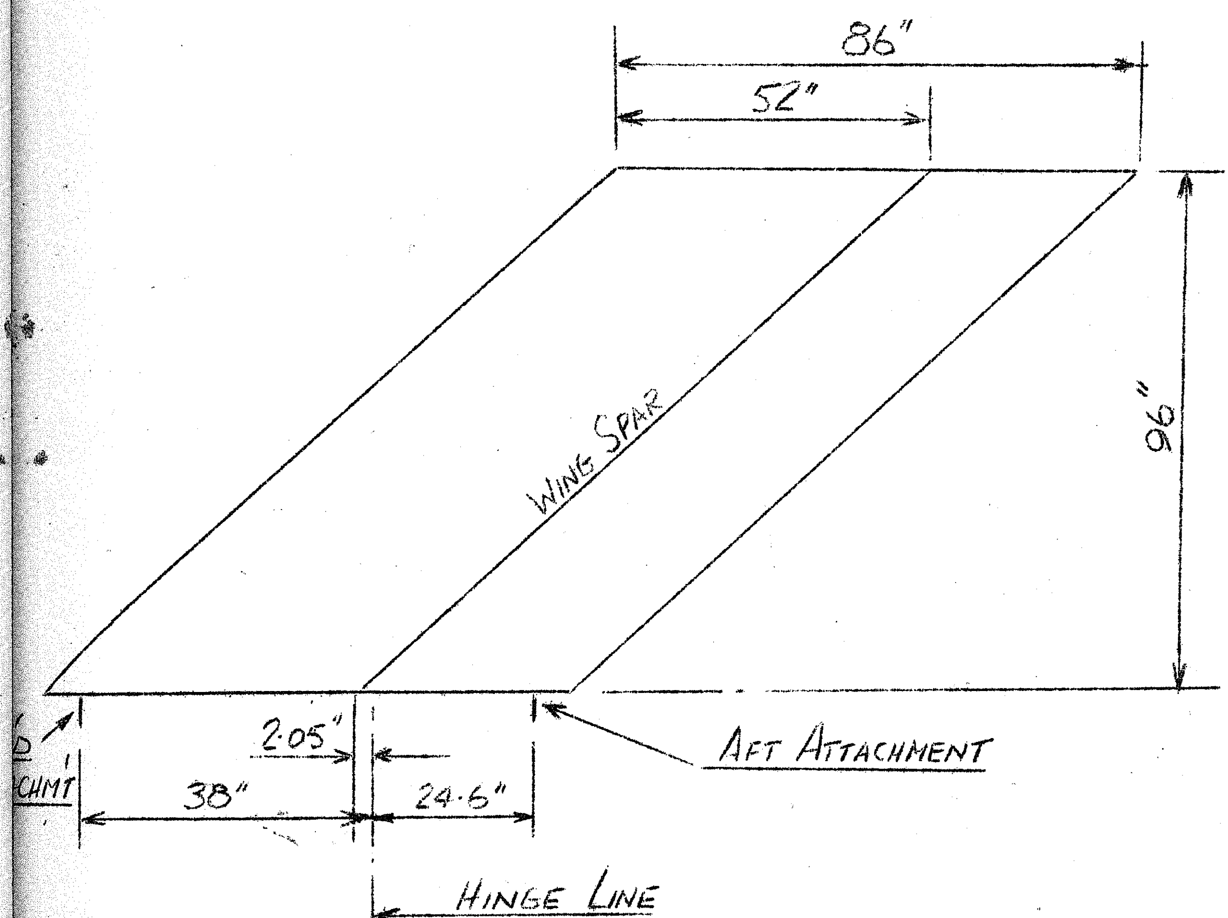
DETAIL STRESSING OF WING TIP YAWMETER

Drgs. No. M-167-7 and M-167-8	No detail stressing done by C. of A. Yawmeter installation is R.A.E. design, see R.A.E. Drg. No. AER.14460.

a - 1 WING SPECIFICATION (GENERAL)

Span	b	=	8'
Chord	c	=	7' 2"
Area	S	=	57.33 ft^2
Ratio	$^t/c$	=	10%
Sweepback		=	45^o
Spar location		:	40% Chord (Equivalent)
Section		:	Constant

a - 2 WING DIMENSIONS



For detailed wing dimensions and mounting geometry see data diagram drg.no. M.167-3.

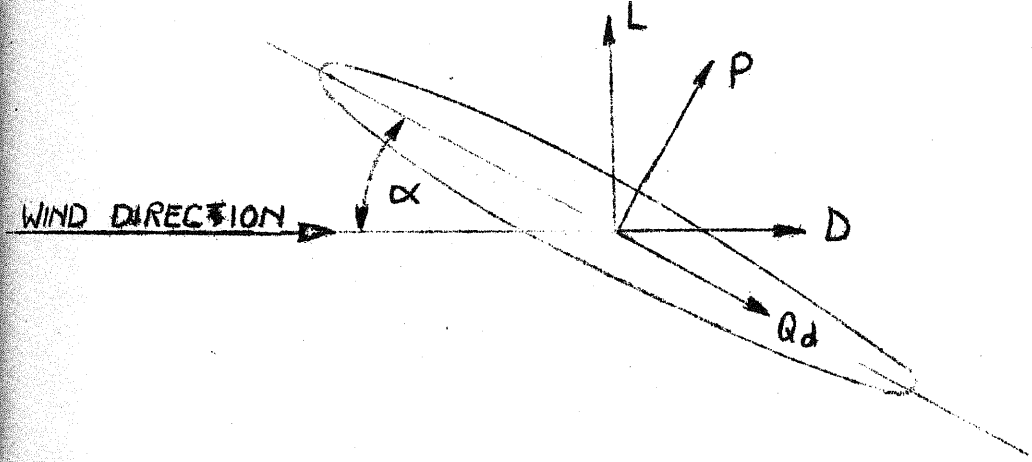
B WING LOADING CASES

b-1 Aerodynamic Wing Loads

The following limitations on forward speed and swept wing incidence will be imposed:-

Wing Incidence	Max. Permissible For'd Speed	
	m.p.h.	knots
10°	230	200
9°	240	208
8°	250	217

A side gust $v = \pm$ 25 ft/sec is assumed to be the worst encountered during flight. The wing incidence at a for'd speed V is effectively increased by an angle v/V, however the resultant airspeed $\sqrt{V^2 + v^2}$ is not substantially different from V in the cases considered. Air loads on the wing for the three cases listed above are, therefore as follows:-



		Case		
		1	2	3
Incidence	Degrees Radians	10 0.1745	9 0.1571	8 0.1396
Speed 'V'	m.p.h. ft/sec	230 337.3	240 352	250 366.7
Side gust 'v'	ft/sec	25	25	25
$v/_V$	Radians	0.0741	0.0710	0.0683
Effective Incidence α	Radians Degrees	0.2486 14.24	0.2281 13.07	0.2079 11.91
$dC_L/_{d\alpha}$ (1)	Per Radian	2.237	2.237	2.237
$C_L = dC_L/_{d\alpha} \cdot \alpha$		0.556	0.510	0.465
$L = \frac{1}{2} C_L \rho S V^2$	lbs	4315	4310	4265
Ratio $L/_D$ (2)		7	8	9.5
Drag D	lbs	616	539	449
L Cos α	lbs	4180	4200	4170
D Sin α	lbs	152	122	93
P = L Cos α + D Sin α	lbs	4332	4322	4263
L Sin α	lbs	1062	975	880
D Cos α	lbs	597	525	440
Q_d = D Cos α - L Sin α	lbs	- 465	- 450	- 440

References:- (1) Lift Curve Slope : Flight Dept. Memo DF/Res/7

(2) Ratio $L/_D$: Experimental investigation of the effect of aspect ratio upon the Aerodynamic characteristics of sweptback wings at low speeds. Fig. X by R. Bumstead Rensselaer Polytechnic Institute (Jan. 1949)

The critical flight case is, therefore, Case 1, with a load normal to the wing chord P = 4332 lbs and a load acting forward along the chord Q_d = 465 lbs.

$\therefore$ Ultimate loads (factor of 1.5) :- P = 6500 lbs

Q_d = 700 lbs

b-2 **Inertia Loads**

Total weight of complete wing assembly = 550 lbs estimated

i) **Fore and Aft Inertia Loads**

Forward inertia loads due to forward and pitching accelerations are very small and can be neglected.

ii) **Downward Inertia Loads**

Max. downward acceleration appears to be 3.17g, giving an ultimate factor of 4.75 quoted in the 'Lancaster' type record.

$\therefore$ Max. downward inertia load $R = (4.75 \times 550)$
$= \underline{2620 \text{ lbs}}$

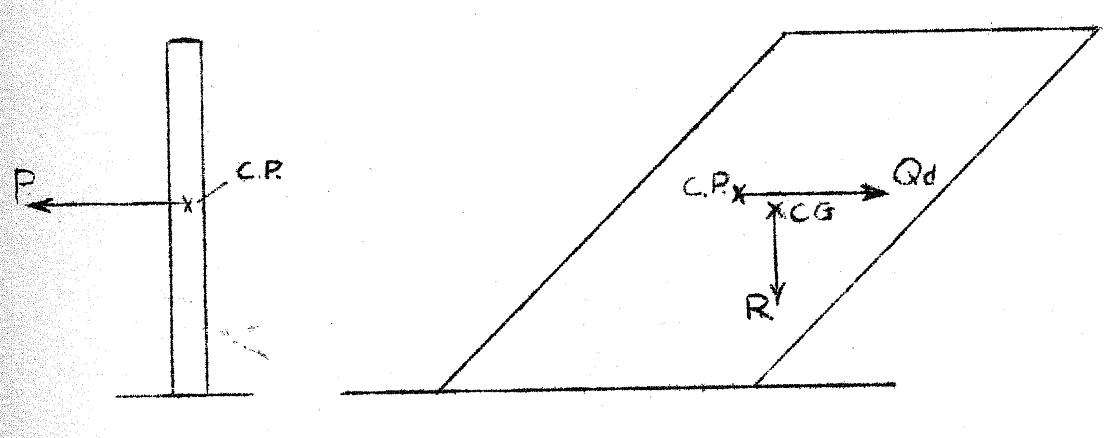
iii) **Inertia Side Loads**

Inertia side load due to gust is always a relief to the aerodynamic load but the magnitude is so small that it may be neglected. Additional side loads due to yawing and and rolling accelerations are also of small magnitude by comparison with the gust side load and therefore they may also be safely neglected.

Installation of the wing, therefore, imposes no restriction on aircraft manoeuvrability other than the speed limitations quoted in section b-1.

b-3 **Critical Loading Cases**

To sum up:- The critical loading cases for the wing and attachments are as follows:-

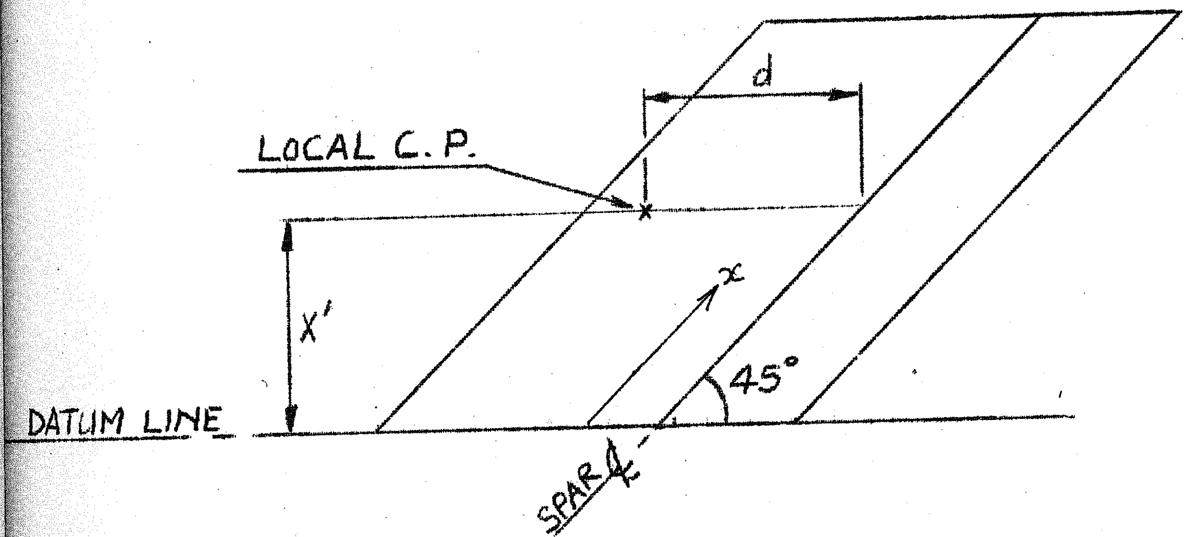


Case	V	α_o	ft/sec 'V'	Factored g	P	Q_d	R
A	230	10^o	25	4.75	6500	-700	2620
B	230	10^o	25	0	6500	-700	0
C	250	0^o	0	4.75	0	138*	2620

* $Q_d = D = \frac{1}{2} C_{D_o} \rho S V^2 \times 1.5 = \underline{138}$ lbs

$C_{D_o} \triangleq .01$ (Ref. 2)

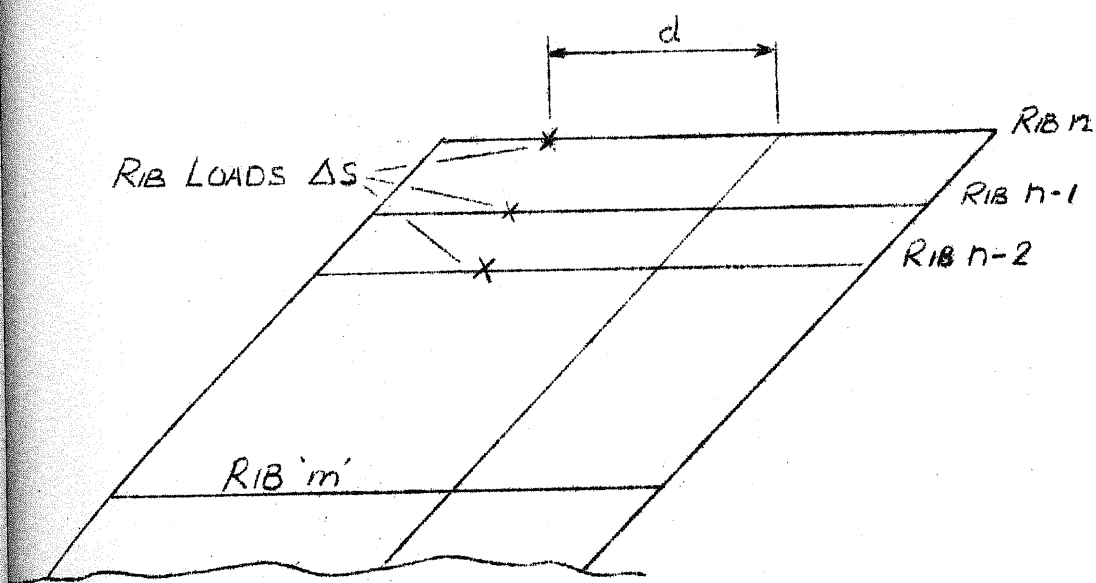
b-4 Shear, Bending and Torque Diagrams at Critical Load



Distribution of load P = 6500 lbs along wing given as follows:-

Station	X' - distance from Wing datum (in.)	d - distance from spar ℄ (in.)	ΔP(lbs)
10	91.2	39.4	385
9	81.6	35.5	560
8	72.0	32.7	613
7	62.4	30.6	650
6	52.8	28.9	683
5	43.2	27.5	705
4	33.6	26.3	721
3	24.0	25.3	726
2	14.4	24.4	728
1	4.8	23.8	729

Assuming a rib is carrying a load Δ S with its C.P. at a distance 'd' from the spar, the load Δ S is transferred into the spar web via the rib web and the torque Δ S.d will be carried by the wing skin.



Then shear force in spar web inb'd of rib 'm' is given

$$\text{by } S_m = \sum_n^m \Delta S$$

$$\text{Torque } T_m = \sum_n^m \Delta S.d$$

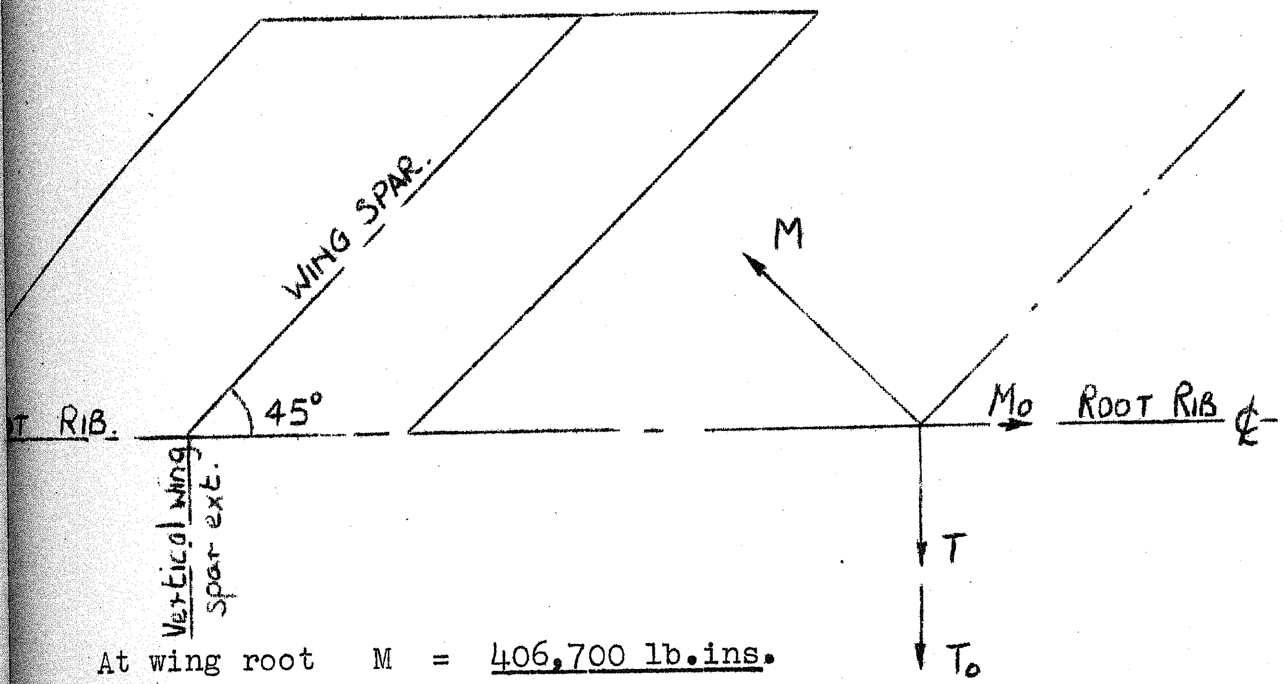
Bending moment along spar at rib 'm' - $M_m = M_{m-1} + S_{m-1} \times p$

(where p = rib pitch measured along spar)

Although the loads given at the stations above are not the rib loads, the shear, torque and bending in the wing can be calculated by exactly the same method. This has been done in the following table and plotted in figures 1,2 and 3 :-

Station	X' (ins)	X (ins)	p (ins)	Δs (lb.)	$S = \sum \Delta s$ (lb.)	$S_{m-1}\,p$ (lb.in)	M (lb.ins)	d (ins)	$d.\Delta s$ (lb ins)	T (lb ins)
10	91.2	128.975	13.576	385	385	0	0	39.4	15,169	15,169
9	81.6	115.399	13.576	560	945	5,227	5,227	35.5	19,880	35,049
8	72.0	101.822	13.576	613	1558	12,829	18,056	32.7	20,045	55,094
7	62.4	88.246	13.576	650	2208	21,151	39,207	30.6	19,890	74,984
6	52.8	74.670	13.576	683	2891	29,976	69,183	28.9	19,739	94,723
5	43.2	61.093	13.576	705	3596	39,248	108,431	27.5	19,388	114,111
4	33.6	47.517	13.576	721	4317	48,819	157,250	26.3	18,962	133,073
3	24.0	33.941	13.576	726	5043	58,608	215,758	25.3	18,368	151,441
2	14.4	20.364	13.576	728	5771	68,464	284,222	24.4	17,763	169,204
1	4.8	6.788	6.788	729	6500	78,347	362,569	23.8	17,350	186,554
Datum	0	0		-	6500	44,122	406,691	-	-	-

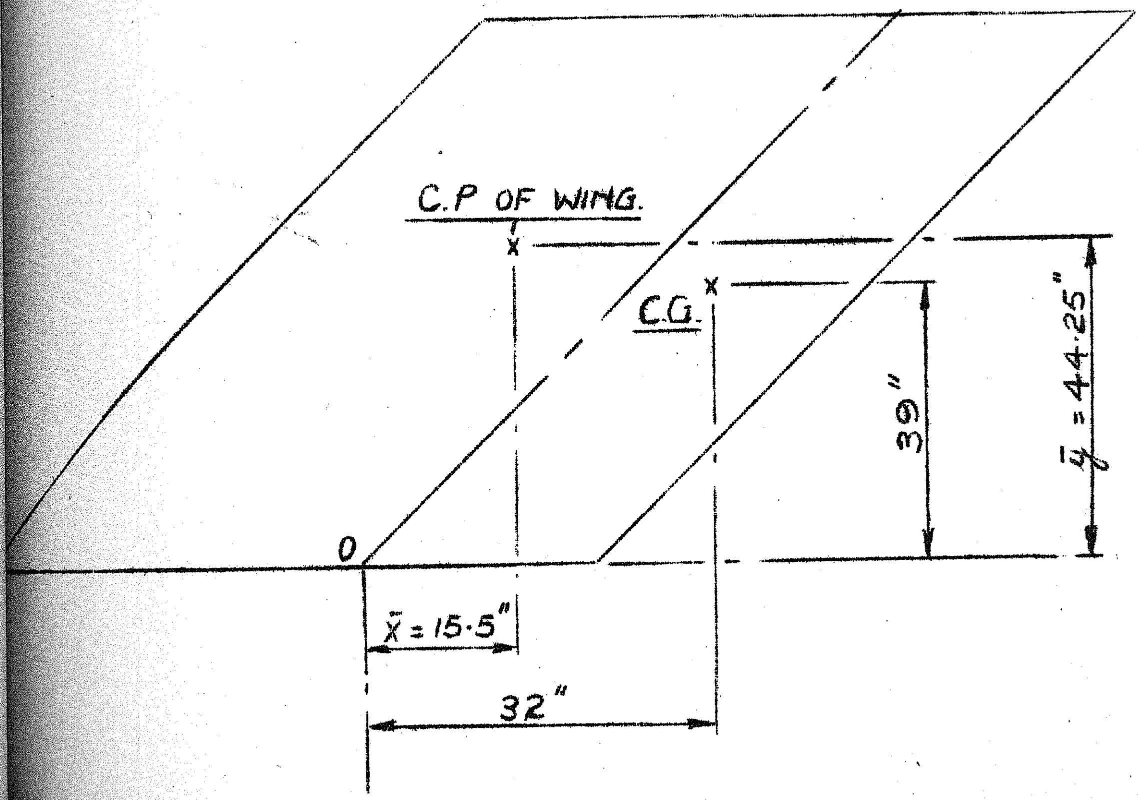
b-5 <u>Bending and Torque at Wing Root and C.P. Position</u>



At wing root $M = \underline{406,700 \text{ lb.ins.}}$
$T = \underline{186,600 \text{ lb.ins.}}$

Balancing moment $M_o = M \cos 45^\circ = \underline{287,600}$ lb.ins.
acting on the vertical wing spar extension (vertical spar).

Balancing torque $T_o = M \cos 45^\circ - T$
$= 287,600 - 186,600$
$= \underline{101,000}$ lb ins
provided by wing attachment fittings.



$\therefore$ Position of C.P. on wing : $\bar{y} = \frac{M_o}{S} = \frac{287,600}{6500}$

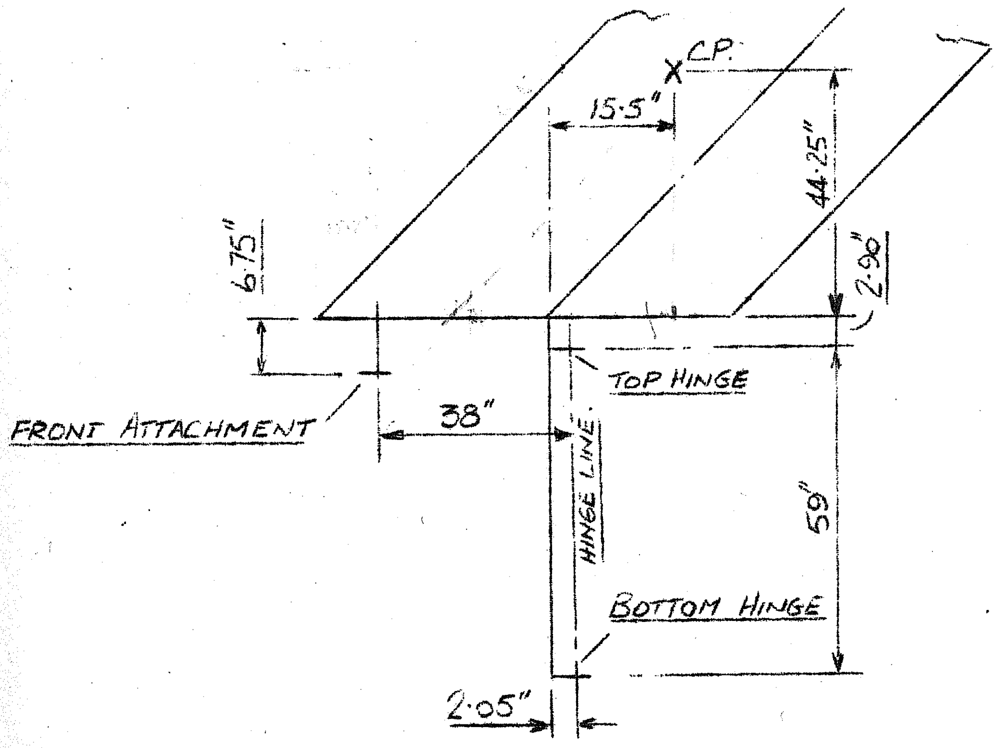
$= \underline{44.25''}$

$\bar{x} = \frac{T_o}{S} = \frac{101,000}{6500}$

$= \underline{15.50''}$

Wing C.G. is assumed to be at a point 39" from the rib 1 datum line and 32" from the vertical spar datum. This is considered to be a conservative estimate.

b-6 <u>Wing Attachment Points - Reactions</u>



52
15·5
67·5
44·25
23·25

Torque about hinge line = 6500 (15.5 - 2.05)
= <u>87,700</u> lb ins

$\therefore$ Reaction at for'd attachment $= \frac{87700}{38} = \underline{2310}$ lb

B.M. at top hinge due to air load = 6500(44.25 + 2.9) - 2310(6.75 - 2.9)

= <u>297,600</u> lb ins

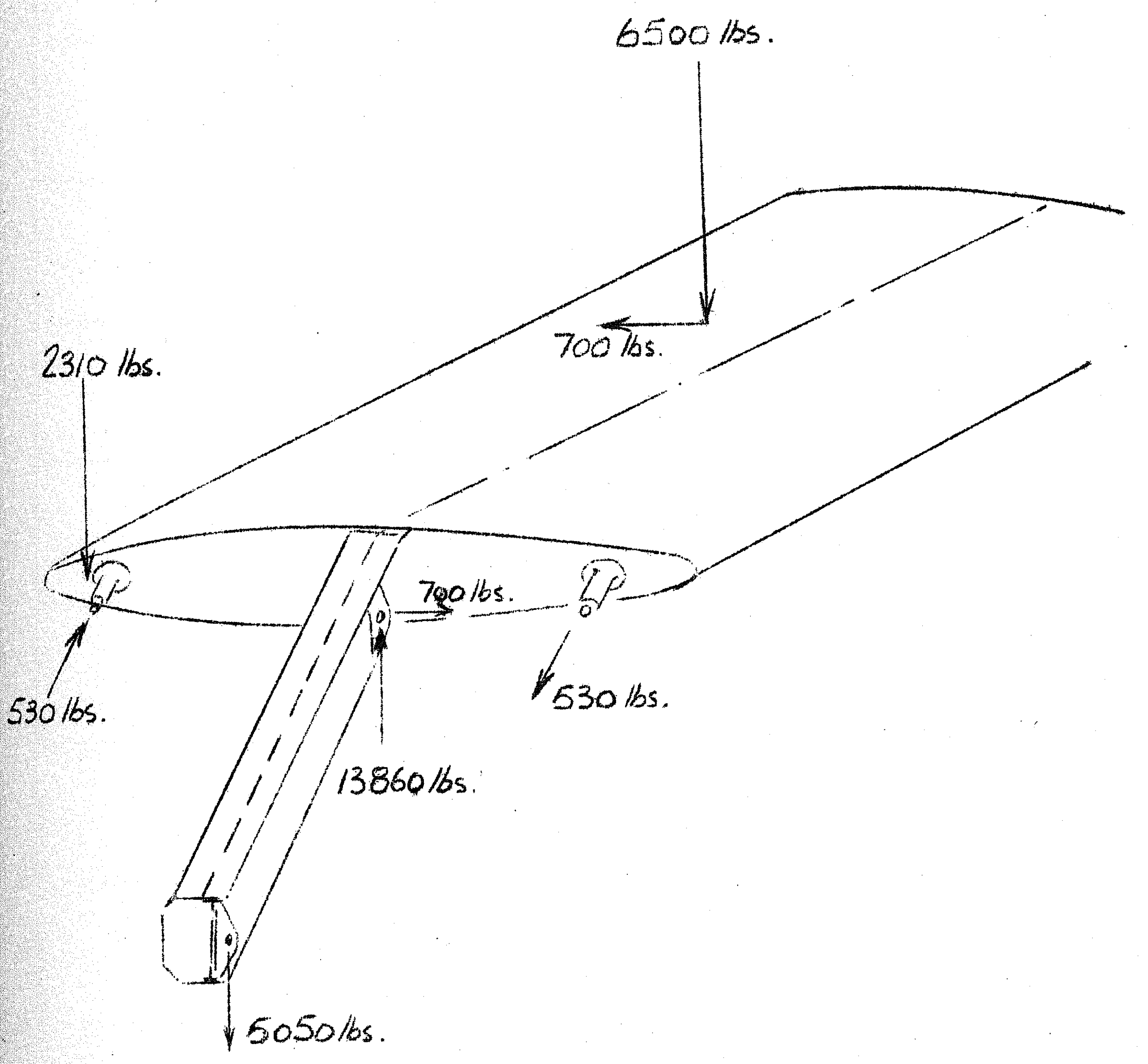
$\therefore$ Balancing reaction at bottom hinge $= \frac{297600}{59}$

= <u>5050</u> lbs

$\therefore$ Reaction at top hinge = (6500 + 2310 + 5050)

= <u>13,860</u> lbs

Case B



For'd and aft load Q_d acting at the C.P. is assumed to be reacted on the for'd and aft attachments via the leading and trailing edge members.

Vertical load on each fitting $= \frac{47.15}{62.6} Q_d = \underline{0.75\ Q_d}$

Load R at C.G. is also assumed to be reacted at the attachments.

$\therefore$ Vertical load on for'd fitting $= \frac{5.35}{62.6} R = \underline{0.085\ R}$

$\therefore$ Load on aft fitting $= \underline{1.085\ R}$

Worst load on aft fitting occurs, therefore, in case C.

When $Q_d = 138$ lb and $R = 2620$ lb, Vertical load = 2950 lb

Similarly, worst load on for'd fitting occurs in case B.

When $Q_d = -700$ lb and $R = 0$, Vertical load = 530 lb.

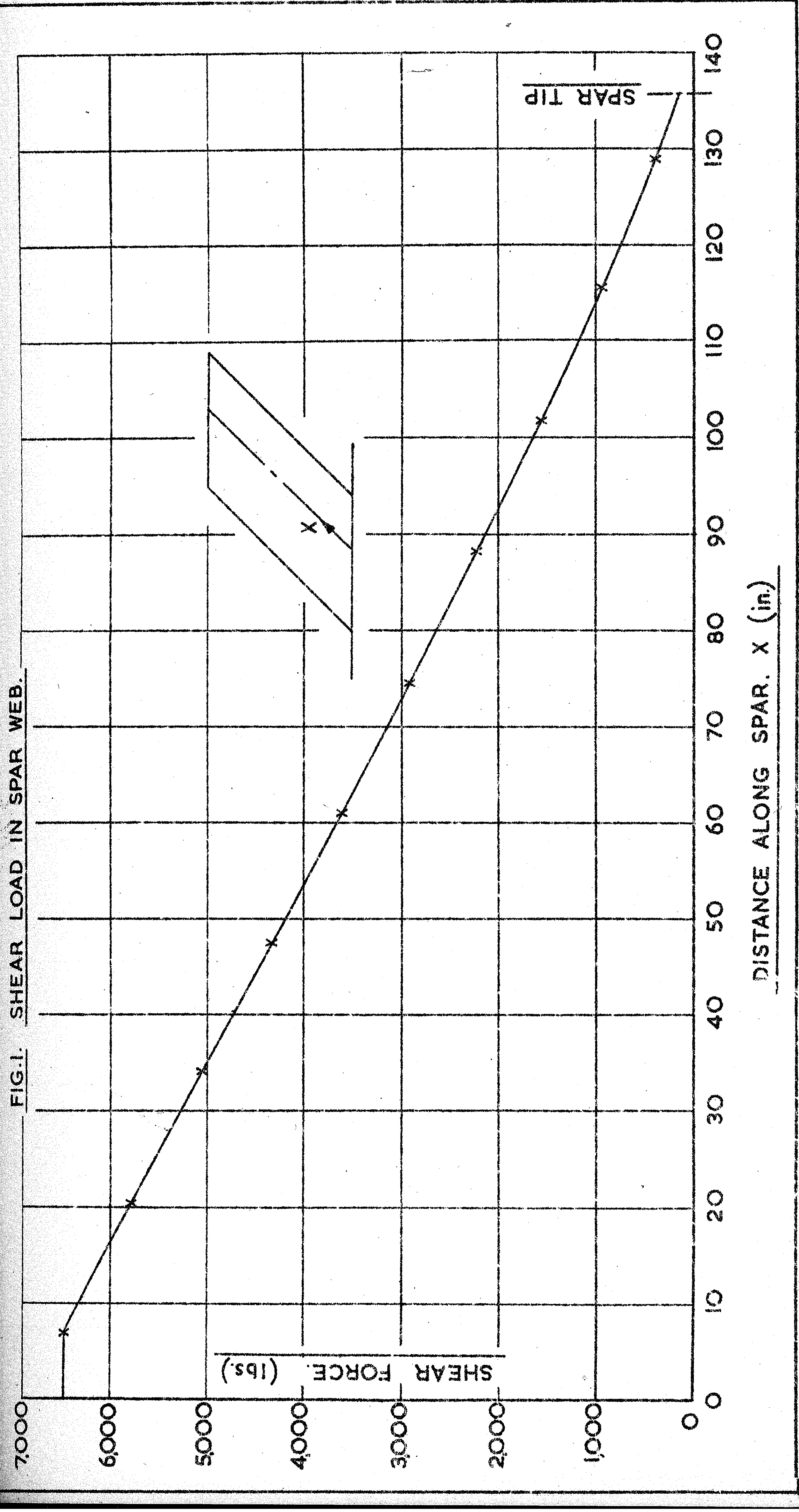
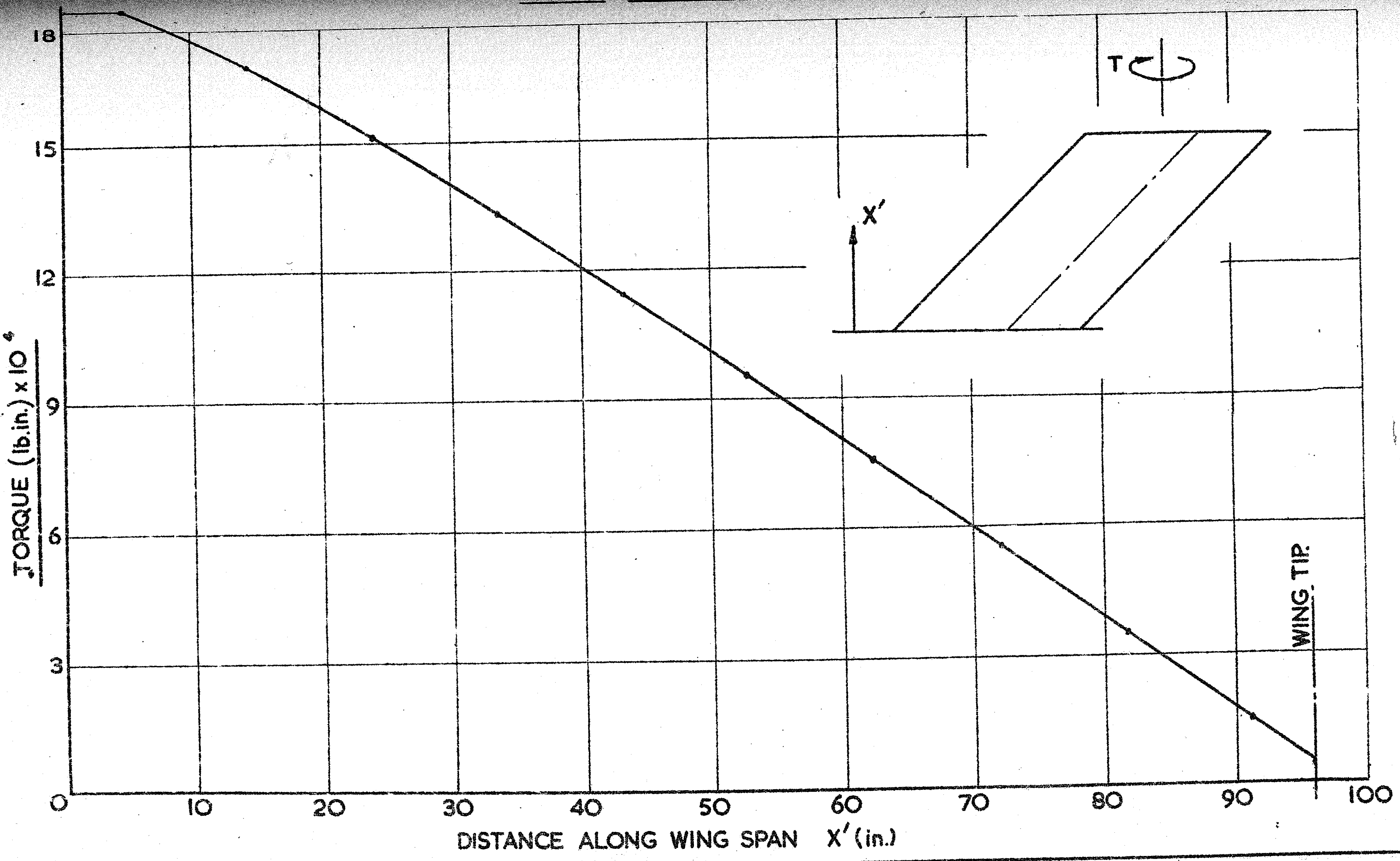


FIG.1. SHEAR LOAD IN SPAR WEB.

FIG. 2. TORQUE CARRIED BY WING SKIN.



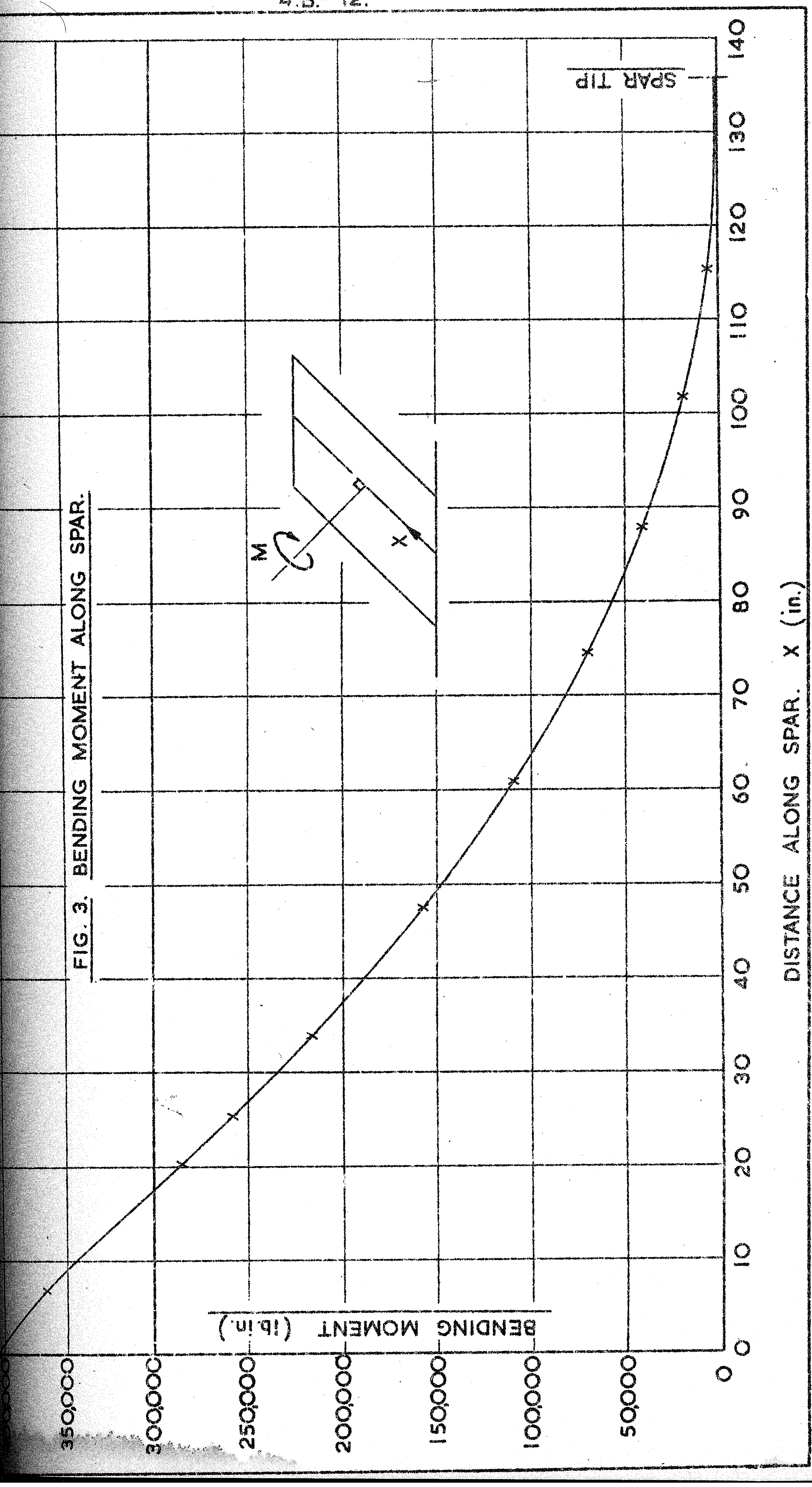


FIG. 3. BENDING MOMENT ALONG SPAR.

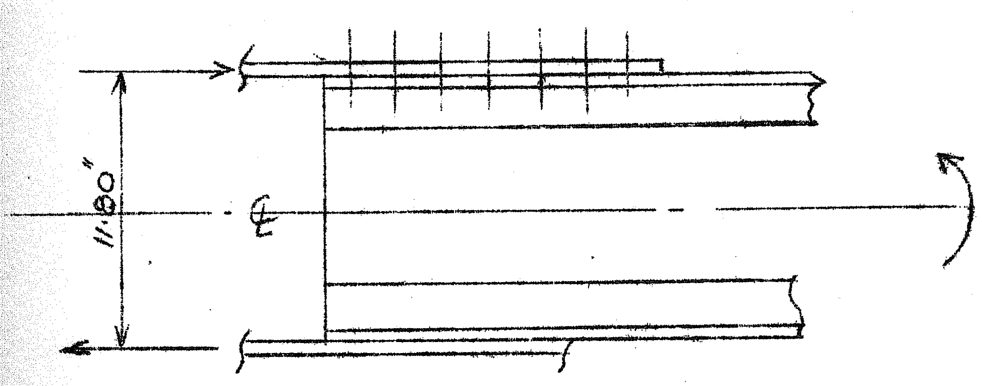
DETAIL STRESSING OF WING STRUCTURE

R.F.

WING SPAR ROOT (Drgs. No. M.160-10 and M.160-3)

Max. Shear Load in spar web = 6,500 lbs.
Max bending moment = 406,700 lbs in

(i) 1. Joint Plate (3 s.w.g. D.T.D. 124 with 10 s.w.g. D.T.D. 124 welded flanges)



Load on each plate = $\frac{406,700}{11.8}$ = 34,500 lbs.

2. Attachment Bolts (5/16" B.S.F. H.T.S.)

14 bolts ∴ load per bolt = $\frac{34,500}{14}$

= 2,470 lbs.

Assuming a concentration factor of 1.2 for uneven distribution ∴ Worst load
= (2470 x 1.2) = 2,960 lbs

Allowable single shear in 3 s.w.g. plate
= 6,210 lbs > 2

3. Bearing of Bolt in Spar Boom

Allowable bearing load - L65 extrusion
= (1700 x 4.33) = 7,360 lbs

Applied Load = 2,960 lbs > 2

(ii) Spar Boom (2" x 2" x .25" L.65 Extrusion)

Depth between centroids = 10.44"

Max. end load = $\frac{406700}{10.44}$ = 39,000 lbs.

Spar web at root is reinforced by 14 s.w.g. S. 3 doubling plates ∴ there are no buckling loads on the boom.

1. Compression

Area of boom section = 0.9375 in^2

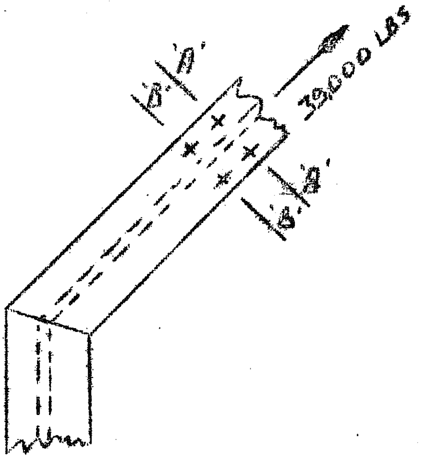
∴ Max. compressive stress = $\frac{39,000}{.9375}$

= 41,600 lbs/in^2

Allowable compressive stress = 55,000 lbs/in^2 1.32

R.F.

2. Tension



Tension across 'A-A' :-

2 - 5/16" dia. holes ∴ Area $= .9375 - (2 \times \frac{5}{16} \times \frac{1}{2})$

$= \underline{0.7812}$ in^2

∴ Tensile stress $= \frac{39000}{.7812} = \underline{50,000}$ lbs/in^2

Allowable tensile stress $= \underline{63,000}$ lbs/in^2 1.26

Tension across 'B-B'

2 - 5/16" dia. and 1 - $\frac{3}{8}$" dia. hole

∴ Area $= .7812 - (\frac{3}{8} \times \frac{1}{4})$

$= \underline{0.6874}$ in^2

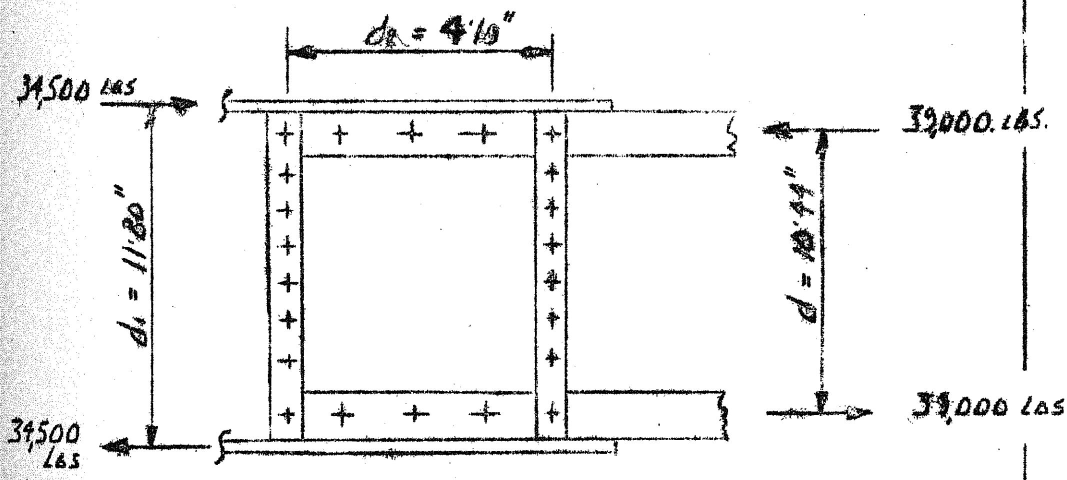
Load in section $= 39,000 - (2 \times 2470)$

$= \underline{34,060}$ lbs

∴ Tensile stress $= \frac{34060}{.6874} = \underline{49,600}$ lbs/in^2 1.27

(iii) Spar Root Web

Shear due to joint plate offset :-



Bending moment at root - M $= 406,700$ lbs.in

Spar boom end load - P $= 39,000$ lbs $(\frac{M}{d})$

Joint plate end load - P_1 $= 34,500$ lbs $(\frac{M}{d_1})$

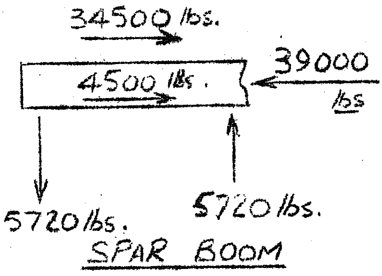
R.F.

$\therefore \quad (P - P_1) = \underline{4,500}$ lbs

Stiffener distance $d_2 = 4.1"$

$\therefore$ Shear flow $= \dfrac{4500}{4.1}$

$= \underline{1,100}$ lbs/in

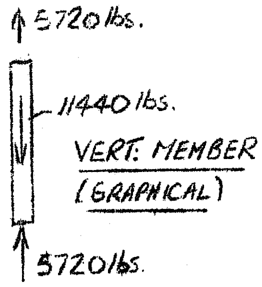


Moment on each boom due to offset $= 34,500 \left(\dfrac{11.8 - 10.44}{2} \right)$

$= \underline{23,450}$ lb.ins.

$\therefore$ Load on each stiffener - L $= \dfrac{23,450}{4.1}$

$= \underline{5,720}$ lb.



$\therefore$ Total load through web = (2x5720) = $\underline{11,440}$ lbs

1. Boom to Web Joint :-

3 - $\frac{3}{8}$" dia. bolts

Allowable single shear in 14 s.w.g. plate = $\underline{8,750}$ lbs

Allowable bearing = $\underline{2,600}$ lbs

$\therefore$ Allowable bearing load on joint = (3 x 2600)

= $\underline{7,800}$ lbs

Applied load = $\underline{4,500}$ lbs 1.73

2. Boom to Stiffener Joint :-

Weakest stiffener - 18 s.w.g. D.T.D. 124 Channel Section with 10 s.w.g. D.T.D.124 radius strip

2 - $\frac{1}{4}$" dia. bolts at joint.

Allowable single shear per bolt = $\underline{3,980}$ lbs

$\therefore$ Total allowable shear = (2 x 3980) = $\underline{7,960}$ lbs

Applied load = $\underline{5,720}$ lbs 1.39

3. Stiffener to Spar Web Joint :-

11 - 2 B.A. Bolts

Allowable single shear per bolt = $\underline{2,180}$ lbs

$\therefore$ Total allowable shear = (11x2180) = $\underline{24,000}$ lbs

Applied load = $\underline{11,440}$ lbs >2

4. Stiffener Strength :-

Area of Section = (2.2 x .048) = $\underline{.106}$ in^2

Area of Radius Plate = $\underline{.09}$ in^2

$\therefore$ Total Area = $\underline{.196}$ in^2

Max. compressive stress on section $= \dfrac{5,720}{.196}$

= $\underline{29,200}$ lb/in^2

Allowable stress = $\underline{56,000}$ lb/in^2 1.92

5. Shear Web (14 s.w.g. S. 3)

Shear flow = 1100 lbs/in

Shear Stress = $\frac{1100}{.08}$ = 13,800 lbs/in^2

This figure is comparatively small and the root web, therefore, is satisfactory.

(iv) Wing Spar at Rib 2

1. Spar Web (18 s.w.g. D.T.D.610 B)

Shear load at Rib 2 = 6,400 lbs (from Fig. 1)

Shear stress in web $-q$ = $\frac{6400}{10.44 \times .048}$

= 12,770 lbs/in^2

The spar web is bonded on both sides with 3/64" B.S.S. 6.V.3 plywood in order to provide a glue joint to the rib. A shear test has been carried out on a similar composite panel giving results as follows :-

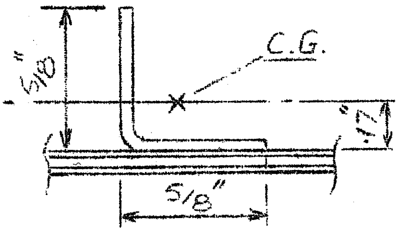
Buckling stress - q_b = 9,150 lbs/in^2

Failing Stress = 18,500 lbs/in^2 > 2

(See Section 6A Report No. T.S.M. 146)

Ratio $\frac{q}{q_b}$ = 1.40, hence, the panel will not buckle under proof loading condition and will, in fact, buckle only very slightly at ultimate load.

2. Web Stiffener ($\frac{5}{8}$" x $\frac{5}{8}$" x 18 s.w.g. D.T.D.610B)



Stiffener cross-sectional area - A_s = .058 in^2

Stiffener moment of inertia - I_s = .00215 in^4

Diagonal tension factor k = $0.434 \left(r + \frac{1}{3} r^3\right)$

(Using the incomplete diagonal tension theory N.A.C.A. T.N. 2661)

Where $r = \frac{q - q_b}{q + q_b} = \frac{3,620}{21,920} = .1607$

Effective stiffener area $A_{se} = \frac{A_s}{1 + \frac{e^2}{p^2}}$

Where e = 0.17" + 0.048" (plywood) + 0.024"

= .242" = distance from centroid to median plane of web

R.F.

$\therefore \quad e^2 = \underline{0.05856}$

$p^2 = \frac{.00215}{.058} = \underline{\bar{0}.0371}$

$\therefore \quad A_{se} = \underline{2.0225}\ in^2$

$\frac{A_{se}}{bt} = \underline{0.11}$

$\therefore \quad \frac{fs}{q} = \underline{0.13}$

aa f_s = average compressive stress over length of stiffener = (0.13 x 12770) = $\underline{1,660}$ lbs/in^2
Allowable stress = $\underline{36000}$ lbs/in^2 | >2

bb $\bar{f}_s$ = average compressive stress over cross-section of stiffener = $f_s \frac{A_{se}}{A_s}$
= $\underline{644}$ lbs/in^2

Effective slenderness ratio = $\frac{d/2}{p}$ = $\underline{27.2}$

Ratio $\frac{e}{c}$ = 1.11

$\therefore$ Allowable column buckling stress
= $\underline{23,000}$ lbs/in^2 | >2

cc Load in stiffener = (644 x .058) = $\underline{38}$ lbs.

$\therefore$ Riveting between stiffener and boom O.K.

dd Max. stress in stiffener at mid-section :-

$$\frac{f_{s\ max}}{f_s} = 1.49$$

$\therefore \quad f_{s\ max.} = 1.49 \times 1660 = \underline{2480}$ lbs/in^2

Allowable compressive stress (crippling)

$f_c = 0.45\ p \cdot \frac{E_t}{E} \cdot f_{ult} = \underline{4280}$ lbs/in^2 | 1.73

3. Spar Boom

Bending moment at rib 2 - M = $\underline{350,500}$ lbs.ins.
(from Fig. 2)

Depth between centroids - d = $\underline{10.44''}$

Area of boom section - A_f = $\underline{.9375}$ in^2

$\therefore$ Stress due to moment - $f_1 = \frac{350500}{10.44 \times .9375}$
= $\underline{35,800}$ lbs/in^2

Angle of diagonal tension is approx. 45^o

hence Tan α = Cot α = 1.0

Compressive stress due to horizontal component of diagonal tension - $f_2 = \dfrac{kq\ \text{Cot}\alpha}{\frac{2A_f}{dt} + 0.5(1-k)}$

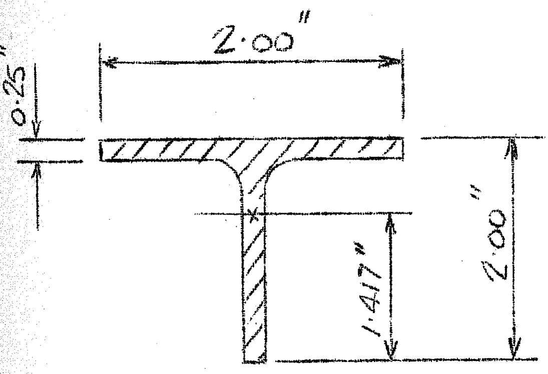
= $\underline{212}$ lbs/in^2

R.F.

Max. B.M. in flange due to vertical component of diagonal tension - $M_f = \frac{1}{12} . k . C_3 . q . t . b^2 . \operatorname{Tan}\alpha$

$(C_3 \simeq 1)$

$= \underline{64.5}$ lb.ins.



Moment of Inertia of spar boom - I_f

$= \underline{.347}$ in^4

Centroid - extreme fibre - y $= \underline{1.417"}$

Then max. compressive stress due to M_f :- f_3

$= \frac{M_f . y}{I_f} = \underline{263}$ lbs/in^2

Stresses, therefore, due to diagonal tension are very small.

∴ Total compressive stress $f = f_1 + f_2 + f_3$

$= \underline{36,275}$ lbs/in^2

Allowable compressive Stress $= \underline{54,000}$ lbs/in^2 1.49

4. <u>Spar to Web Riveting</u> ($\frac{1}{8}$" dia.L.69 Snaphead)

Due to diagonal tension the load per inch run on the joint increases from Q to Q(1 + 0.414 k)

∴ Shear $= \frac{6400}{10.44} \left[1 + (0.414 \times 0.07)\right]$

$= \underline{630}$ lbs/inch

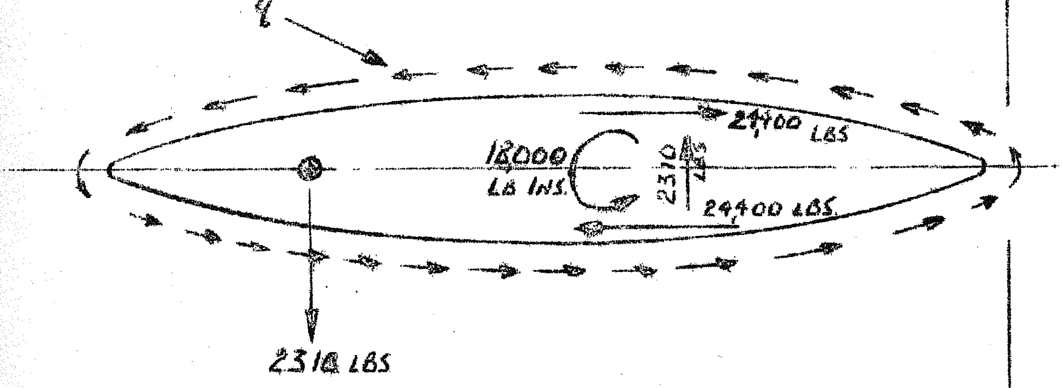
Rivet Pitch $= \frac{1}{2}$"

∴ Joint Strength = (2 x 370) = $\underline{740}$ lbs/inch 1.17

R.F.

ROOT RIB (Rib No. 1) (Drgs. Nos. M.160-10 and M.160-2)

(i) Loading



Shear flow due to torque :- T = 186,600 lb.ins.

$$q = \frac{T}{2A} = 106.8 \quad \text{say} \quad 107 \text{ lbs/in.}$$

$$(A = 874 \text{ in}^2)$$

Load in each joint plate flange = $\dfrac{M \cos 45^\circ}{d_1}$

= 24,400 lbs.

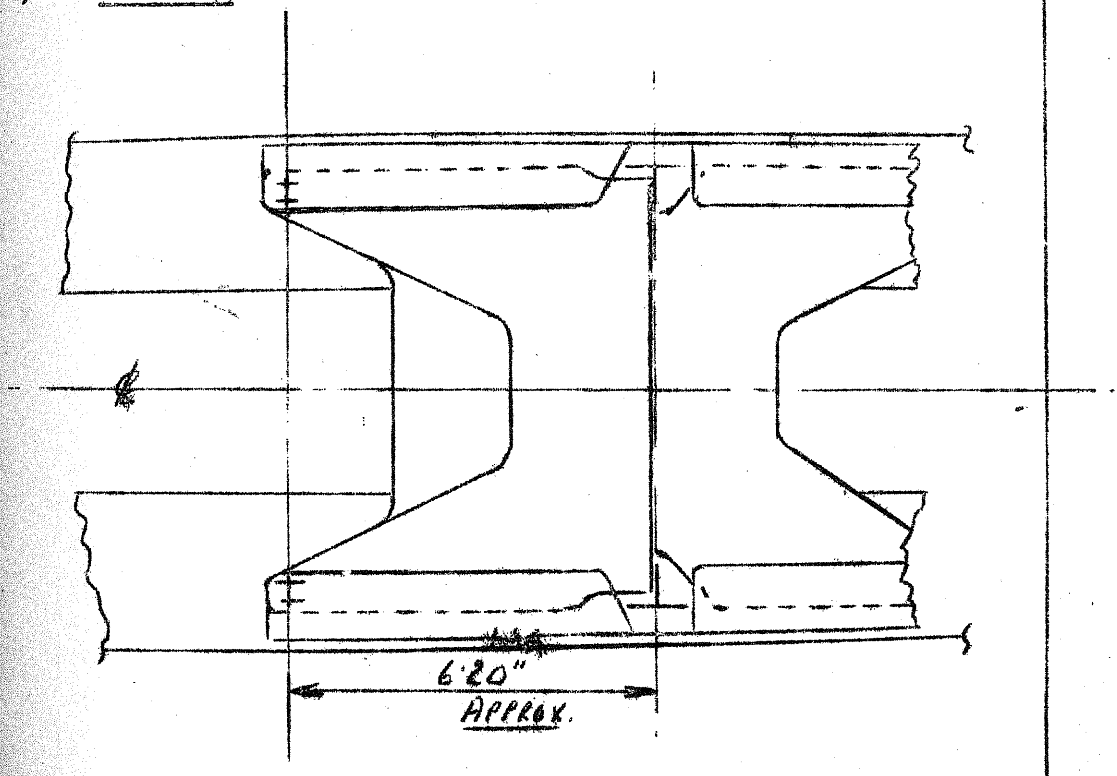
Torque due to forward fitting load
= (2310 x 35.95)= 83,000 lb.ins.

Torque due to hinge offset = (8810 x 2.05)
= 18,000 lb.ins

Total moment on rib = 287,600 - 186,600 - 83,000 - 18,000
= 0

Resultant force is also zero, hence the rib is in complete balance.

(ii) Bending



R.F.

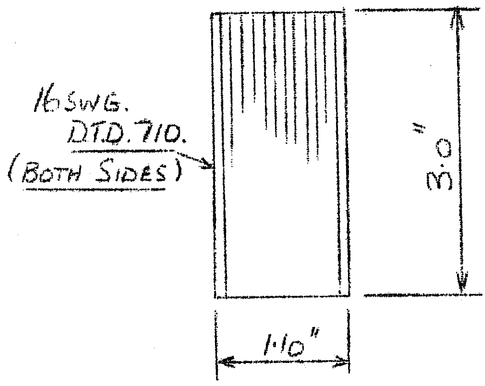
Max. rib side beam load occurs at a section across the first bolt hole on the joint plate flange approx. 6.2" forward of the spar web.

B.M. on section
$= 2310\ (35.95 - 6.2) + \left[106.8 \times 2(528 - 76)\right]$
$= \underline{155,300}$ lb.ins.

Depth between side beam centroids = 9"

$\therefore$ Side beam load $= \frac{155,300}{9} = \underline{17,300}$ lbs.

Each side beam section consists of a plywood block 1.10" thick and 3" deep with 16 s.w.g. alum. alloy D.T.D. 710 bonded on each side.



Plywood Area - A_1 = $\underline{3.3}$ in^2

Area of alum. alloy sheet - A_2= $(3 \times .064 \times 2)$
= $\underline{.384}$ in^2

Young's Modulus - E_1 (Plywood)= 1.0×10^6 lbs/in^2

Young's Modulus - E_2 (Al. alloy sheet)
= 10.0×10^6 lbs/in^2

Assuming the strain ϵ to be identical in both plywood and sheet

than $f_1 = \epsilon E_1$ (Plywood)
$f_2 = \epsilon E_2$ (Sheet)

$\therefore \quad P = A_1 f_1 + A_2 f_2 = \epsilon (A_1 E_1 + A_2 E_2)$

$\therefore \quad \epsilon = \frac{P}{A_1 E_1 + A_2 E_2} = 2420 \times 10^{-6}$

$\therefore \quad f_1 = \underline{2,420}$ lbs/in^2

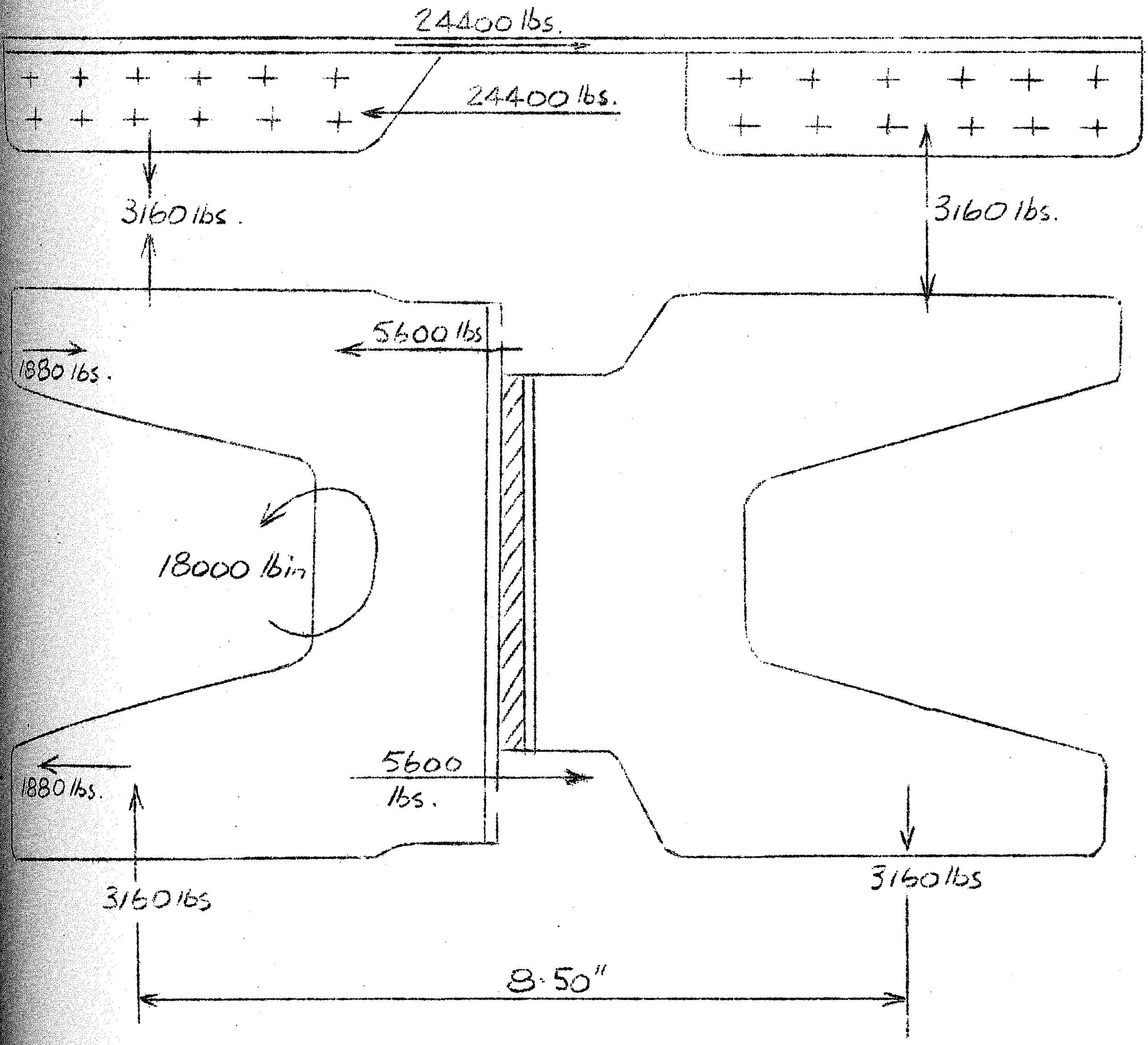
Allowable crushing stress = $\underline{3,570}$ lbs/in^2 1.47

$f_2 = \underline{24,200}$ lbs/in^2

·2% Proof Stress = $\underline{36,000}$ lbs/in^2 1.49

R.F.

(iii) Spar - Rib Joint Plate



Load at each flange (bolt group Crs.)

$$= \frac{24,400 \times 1.10}{8.5} = \underline{3,160} \text{ lbs.}$$

Couples transmitted by doubling plates from top and bottom joint plate flanges into rib beams = (3160 x 8.5) lb.ins. each side

Flange bolt Crs = 9.6"

$$\therefore \text{ Load due to couples} = \frac{2 \times 3160 \times 8.5}{9.6}$$

$$= \underline{5,600} \text{ lbs.}$$

Total load transmitted by flange bolts

$$= (24,400 + 5,600) = \underline{30,000} \text{ lbs.}$$

R.F.

B.M. from aft part of rib at spar web
$= (2 \times 106.8 \times 346) \quad = \underline{73,900}$ lb.ins.

$\therefore$ Load in aft flange $= \dfrac{73,900}{9.6}$

$= \underline{7,700}$ lbs.

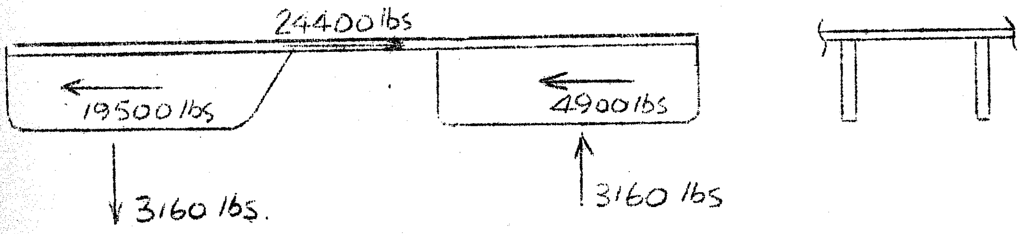
Load in forward flange $= (30,000 - 7,700)$
$= \underline{22,300}$ lbs

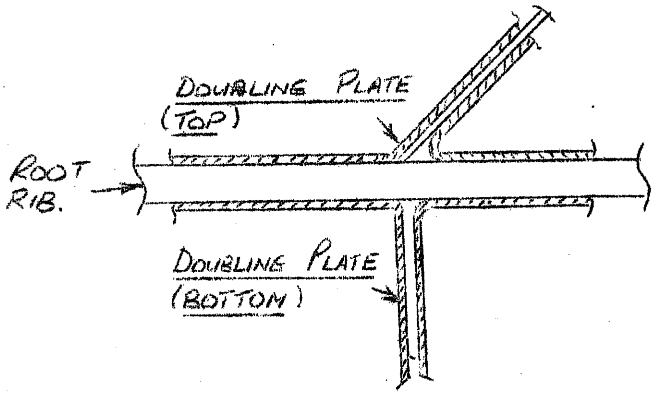
Torque from the vertical spar torsion box is also reacted by the forward flange bolts on one side of the doubling plate.

Torque T = (13,860 - 5050) 2.05
$= \underline{18,000}$ lb.ins.

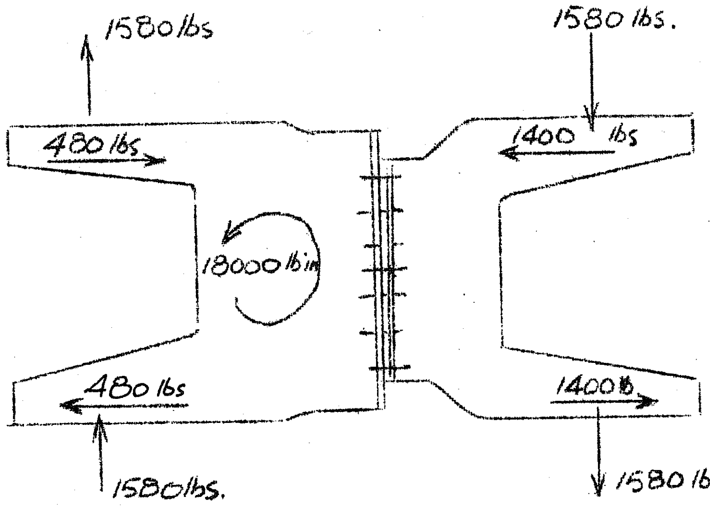
$\therefore$ Load on bolts due to torque
$= \dfrac{18,000}{9.6} \quad = \underline{1,880}$ lbs.

<u>Detail of loading on forward and aft joint plate flanges</u> :-



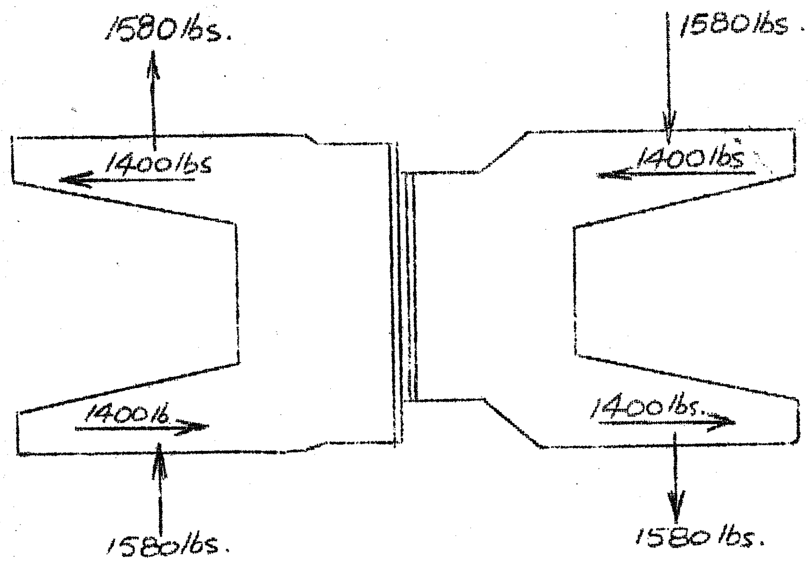


<u>Doubling Plate (Bottom)</u> :-



R.F.

Doubling Plate (Top) :-



1. Flange Bolts (5/16" dia. B.S.F. H.T.S.)

12 off each flange (forward and aft)

Shear

Max. single shear per bolt $= \frac{1}{12}\left(\frac{19500}{2} + 1400\right)$

$=$ 930 lbs.

Allowable single shear (16 s.w.g. al.alloy sheet)

$=$ 5,650 lbs > 2

Bearing

Allowable bearing (16 s.w.g. al. alloy sheet)

$= (448 \times 3.9)$ $=$ 1,740 lbs. 1.88

2. Bottom Doubling Plate Bolts (2 B.A. H.T.S.)
(across spar web)

Shear through doubling plate $=$ 3,160 lbs

Shear from rib to spar web $=$ 2,310 lbs.

$\therefore$ Total shear $=$ 5,470 lbs.

By inspection bolts are adequate > 2

3. Flange Welding

Max. load per flange $=$ 9,750 lbs.

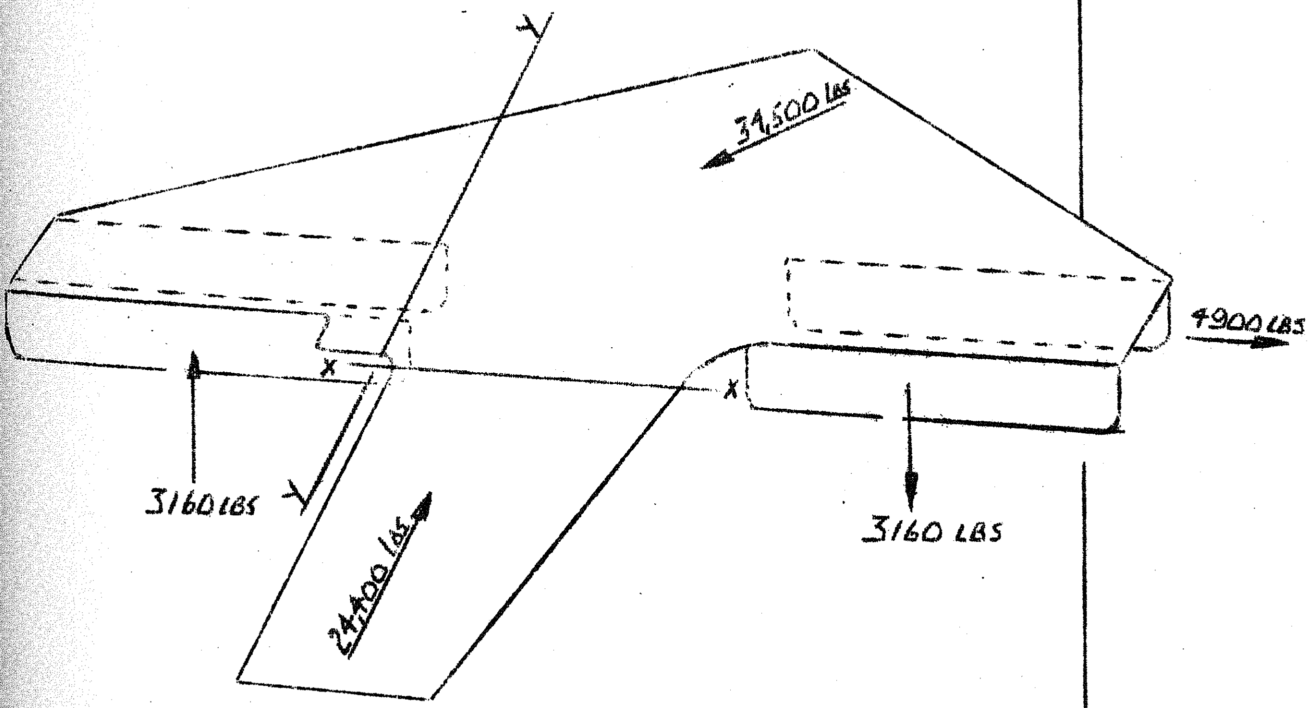
$\therefore$ Load per inch of weld (6" run)

$=$ 1,625 lbs.

Allowable load per inch $=$ 3,000 lbs. 1.85

R.F.

4. Joint Plate



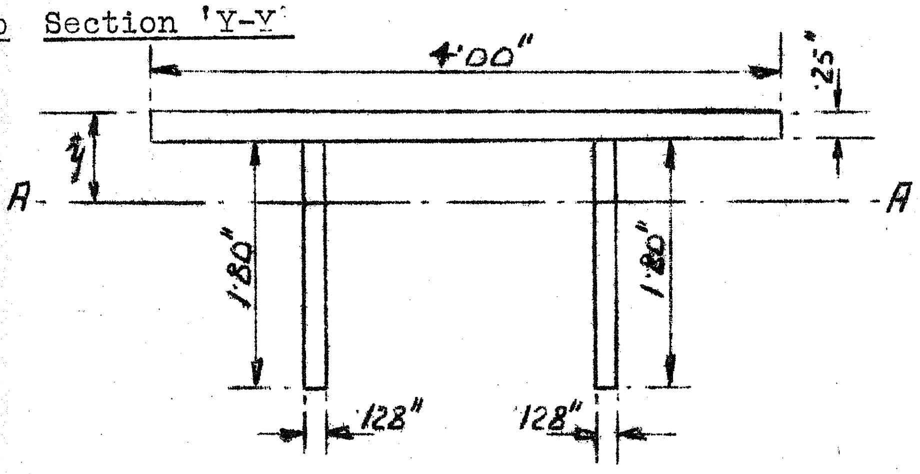
aa Section 'X-X'

Section Area $= (0.25 \times 4) = 1.0\ in^2$

Compressive stress $= \frac{24,400}{1} = \underline{24,400}\ lbs/in^2$

Allowable compressive stress $= \underline{56,000}\ lbs/in^2$ > 2

bb Section 'Y-Y'



Compressive load $= \underline{19,500}$ lbs.

$\therefore$ B.M. $= (19,500 \times 1.1) - (3,160 \times 2.06)$

$= \underline{14,940}$ lb.ins.

Area of section - A $= (4 \times .25) + (2 \times 1.8 \times .128)$

$= \underline{1.461}\ in^2$

Distance of centroidal axis from face of plate

$\bar{y} = \frac{1}{1.461}\left[(1 \times .125) + (.461 \times 1.15)\right]$

$= \underline{.448"}$

$I_{AA} = \underline{.5627}\ in^4$

R.F.

Direct stress $= \frac{19,500}{1.461} = \underline{13,400}$ lbs/in^2

Bending stress (at face of joint plate)

$= \frac{14940 \times .448}{.5627} = \underline{11,900}$ lbs/in^2

Bending stress (at end of flange)

$= \frac{14940 \times 1.602}{.5627} = \underline{42,600}$ lbs/in^2

Section is not symmetrical hence there is induced bending moment about 'B-B' axis (normal to A-A).

I_{BB}, however, is large therefore the bending stress will be so small that it may be neglected.

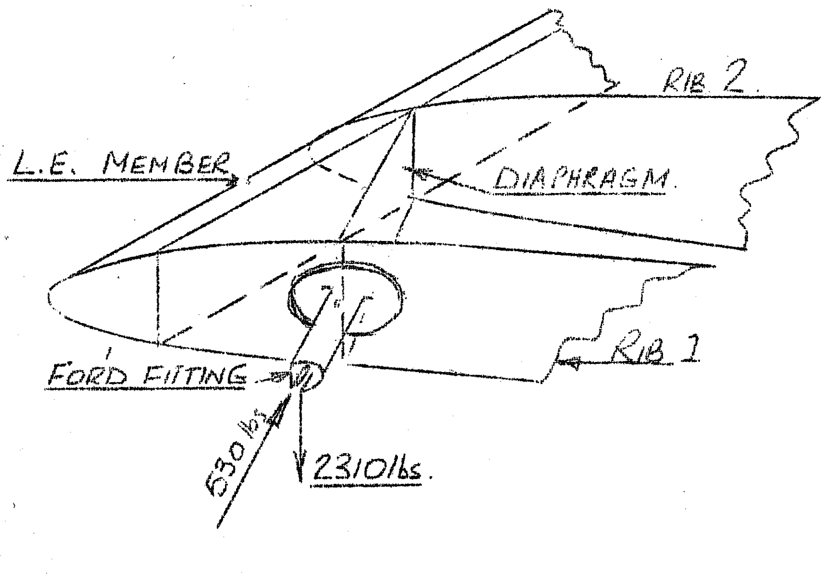
∴ Max. Compressive Stress (at tip of flange section) = (42,600 + 13,400)

$= \underline{56,000}$ lbs/in^2

Allowable compressive stress $= \underline{56,000}$ lbs/in^2 1.0

(iv) Wing Attachment Fitting (Forward)

(Drg. No. M-171-1)



The worst loading condition on the fitting is 2310 lbs. laterally and 530 lbs. vertically.

The vertical load 530 lbs. will be diffused into the leading edge member via the fitting and the diaphragm. The induced stress therefore is negligible.

1. Attachment Fitting Fork End

By inspection the fork end is adequate under the applied loads.

R.F.

2. <u>Bending at Fitting Root End</u> —

The applied load of 2310 lbs. at the fork end produces a moment

$= (2310 \times 6.75) \qquad = \underline{15,600}$ lb.ins.

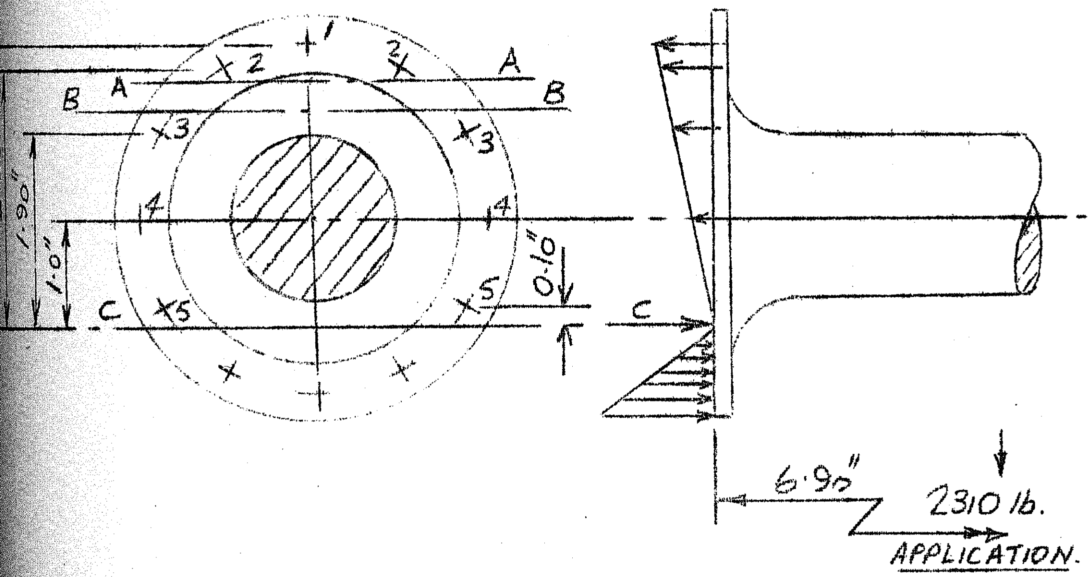
at the fitting root end.

$\therefore$ Bending Stress at root end

$$= f = \frac{15600 \times 4}{\bar{y} \times (0.875)^3} = \underline{29,600} \text{ lb/in}^2$$

Allowable bending stress for S11 $= \underline{108,000}$ lb/in^2 > 2

3. <u>Base Flange of Fitting in Bending</u>



B.M. about fitting Base :-

$M = (2310 \times 6.90") \qquad = \underline{15,950}$ lb.ins.

The B.M. is assumed to be taken by tension in the bolts on one side, & compression against the base on the other of an estimated line 'C-C' (See above Fig.). The base is assumed to be rigid, the loads in the bolts therefore being proportional to their respective distances from 'C-C', as tabulated below :-

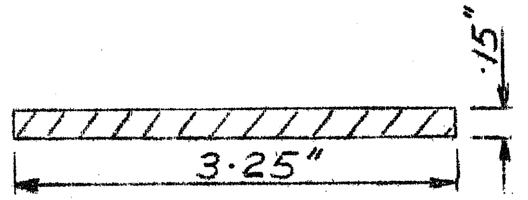
Bolt	No. off	d (ins)	P = kd	P. d.	P (lbs)
5	2	0.10	0.1 k	2 x .01 k	52
4	2	1.00	1.0 k	2 x 1.0 k	516
3	2	1.90	1.9 k	2 x 3.61 k	980
2	2	2.60	2.6 k	2 x 6.76 k	1340
1	1	2.85	2.85 k	8.12 k	1470

Total Moment about 'C-C' $= \Sigma Pd = 30.88$ k.

$$\therefore \quad k = \frac{15950}{30.88} = \underline{516}$$

R.F.

aa <u>Bending across Section 'A-A'</u>
(1.36" from fitting ℄)



BM at Section 'A-A'

$$M = (1470 \times 0.45) + (2 \times 1340 \times 0.2)$$
$$= \underline{1207} \text{ lb.ins.}$$

Sect. Modulus $Z = \left(\frac{1}{6} \times 3.25 \times 0.15^2\right)$
$$= \underline{.0122} \text{ in}^3$$

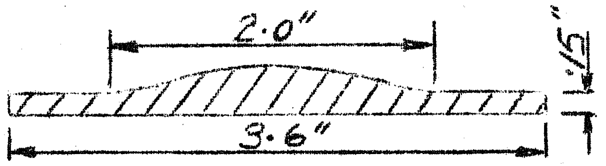
with a form factor K = 1.416 for rectangular section and a 0.2% proof stress of 97000 lb/in^2 for S96 the allowable B.M.

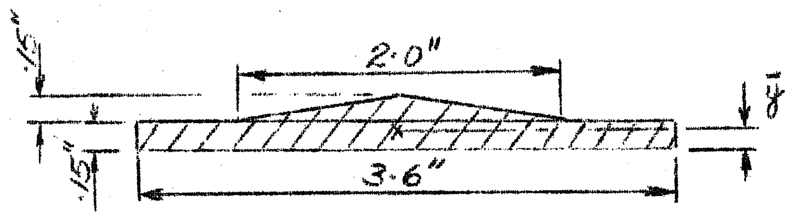
$$M_{cr} = (0.0122 \times 1.416 \times 97000)$$
$$= \underline{1675} \text{ lb.ins.}$$ 1.39

bb <u>Bending across Section 'B-B'</u>
(1.11" from fitting ℄)

$$M = (1470 \times 0.7) + (2 \times 1340 \times 0.45)$$
$$= \underline{2235} \text{ lb.in.}$$

The actual section can be idealised as a rectangle plus a triangle.





Cross sectional area $= (3.60 \times .15) + \frac{1}{2}(1.0 \times .15)$
$$= \underline{0.69 \text{ in}^2}$$

Position of C.G. from base
$$\bar{y} = \frac{1}{0.69}\left[(0.54 \times 0.075) + (.15 \times .2)\right]$$
$$= \underline{0.102} \text{ in.}$$

$$I = \left(\frac{1}{12} \times 3.6 \times .15^3\right) + \left[.54(0.027)^2\right] + \left(\frac{1}{36} \times 2.0 \times .15^3\right) + \left[.15(0.098)^2\right]$$
$$= \underline{0.003035} \text{ in}^4$$

Max. sect. modulus $= \frac{0.003035}{.102} = \underline{0.0298} \text{ in}^3$

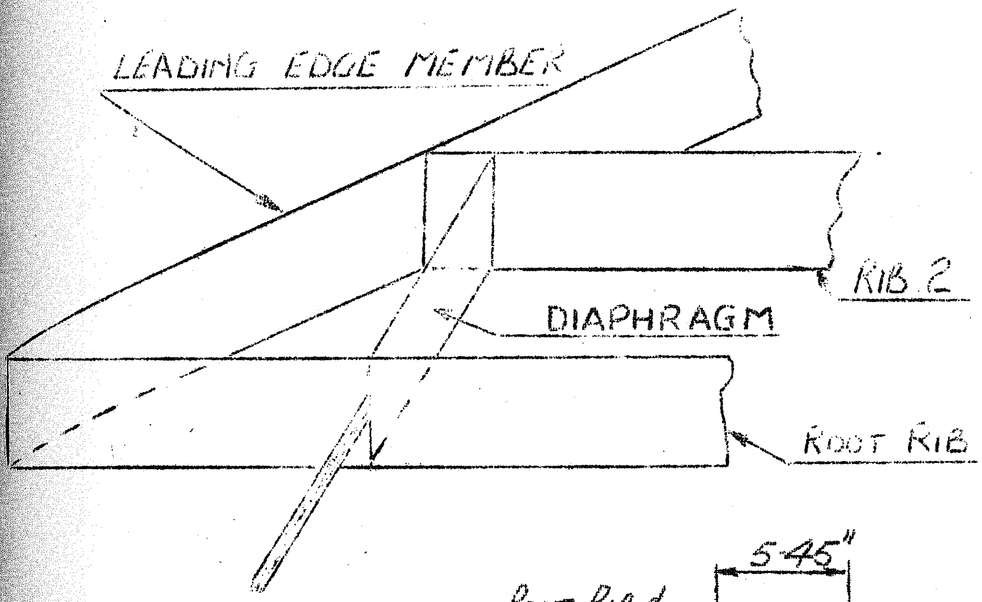
Assuming a form factor of K = 0.8 for the section.
(R.Ae.S. 01.06.03)

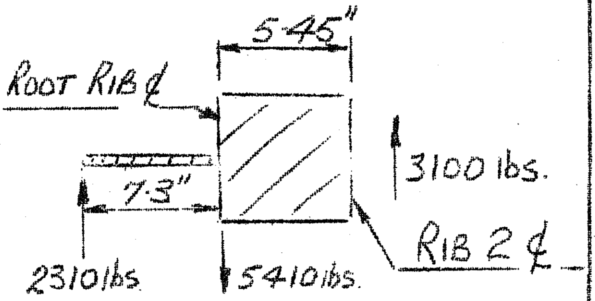
Allowable BM $= (97000 \times 0.8 \times 0.0298)$
$$= \underline{2310} \text{ lb in}$$ 1.035

R.F.

4. Attachment of Diaphragm to forward Fitting

(Drg. No. M.160-1)





The fitting is assumed to be simply supported on Ribs 1 and 2 via the diaphragm.

Load on Rib 2 $= \dfrac{(2310 \times 7.30")}{5.45"} = \underline{3100}$ lbs.

Load on Rib 1 ℄ $= (2310 + 3100) = \underline{5410}$ lbs.

The glue joint between the diaphragm and Rib 2, and Rib 1 has to carry a load of 3100 lbs. over a length of 7.50".

$\therefore$ Load/inch $= \dfrac{3100}{7.50} = \underline{413}$ lb/in

Allowable load/inch for glue joint $= \underline{600}$ lbs 1.45

Section of the diaphragm at Rib 1 carries a BM of $(3100 \times 4.6") = \underline{14,300}$ lb.ins.

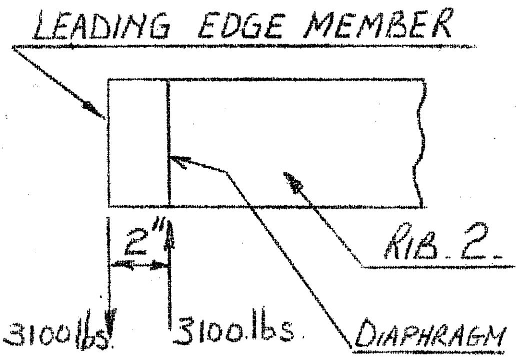
Diaphragm dimensions at that section are 9" x 0.50"

$\therefore$ Bending Stress $= \dfrac{(14300 \times 6)}{(9^2 \times 0.50)} = \underline{2,120}$ lb/in^2

Allowable bending stress (ANC-18)$= \underline{3,500}$ lb/in^2 1.62

5. Loads at Rib 2

At Rib 2, the shear of 3,100 lbs. from the diaphragm is taken out at the leading edge member. The torque of $(3100 \times 2) = 6200$ lb.in. is balanced by a shear flow round the rib, which is very small - approx. 4 lb/in.



The maximum B.M. at Rib 2 is therefore less than <u>6200</u> lb.in.

R.F.

Depth of Rib at diaphragm joint = 7.50"

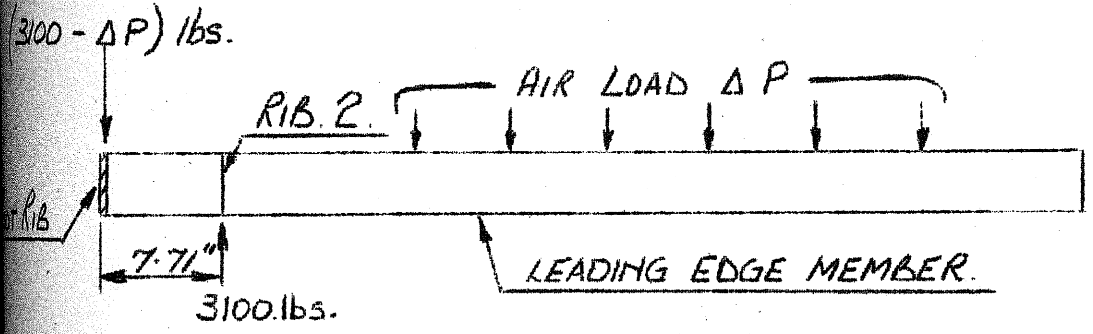
$\therefore$ Load in each rib boom = $\frac{6200}{7.50}$ = $\underline{826}$ lbs.

The boom is 0.5" x 0.5" section spruce

$\therefore$ Applied Stress = $\frac{826}{0.5 \times 0.5}$ = $\underline{3,310}$ lb/in^2

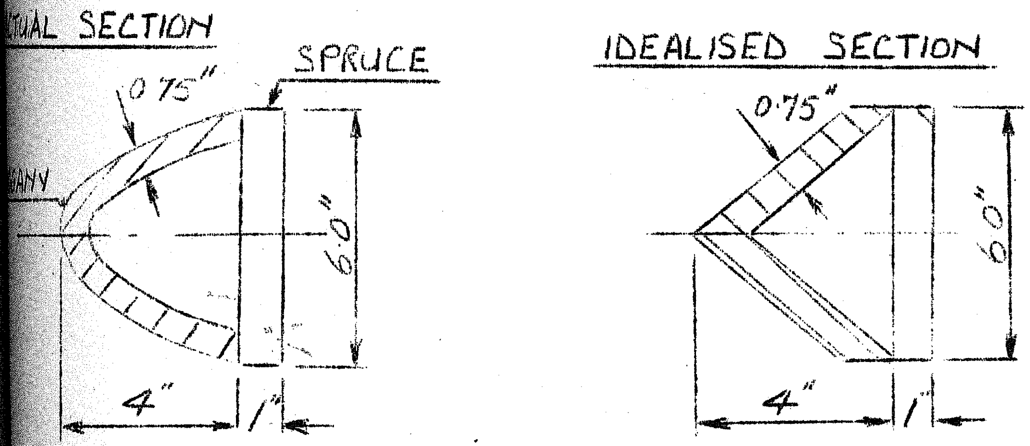
Allowable compressive stress = $\underline{4,400}$ lb/in^2 1.33

6. Bending of Leading Edge Member



The load of 3,100 lbs. in the L.E. member is balanced by air loads ΔP along its length, and a load $(3100 - \Delta P)$ at Rib 1. The loads ΔP can be obtained from wing span air load distribution and L.E. geometry, but it is assumed that the load on Rib 1 is 3100 lbs. and that the maximum B.M. on the L.E. member occurs at Rib 2 which is $\sqrt{2} \times 5.45" = 7.71"$ from Rib 1. Hence the max. B.M. $\doteqdot 3100 \times 7.71 = \underline{23,900}$ lb.in.

The L.E. member comprises a 6" x 1" spruce beam glued to a partially hollow semi-elliptical mahogany nose section the latter being approximately 0.75" thick (see sketch below).



Assuming an idealised section, as above, having uniform properties of spruce.

Section Modulus $Z = \frac{1}{6} \times 6^2 \times 1.75$
$= \underline{10.5}$ in^3

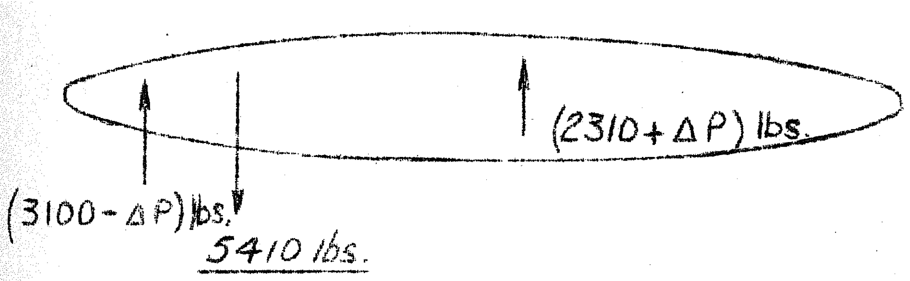
$\therefore$ Bending Stress = $\frac{23900}{10.5}$ = $\underline{2280}$ lb/in^2

Allowable compressive stress (spruce)
= $\underline{4400}$ lb/in^2 1.93

7. Effect of L.E. Member Air Load ΔP on Spar and Rib 1.

The above treatment of the L.E. member actually relieves the shear and bending on the spar, but as the air loads ΔP carried by the L.E. member are very small this effect is neglected.

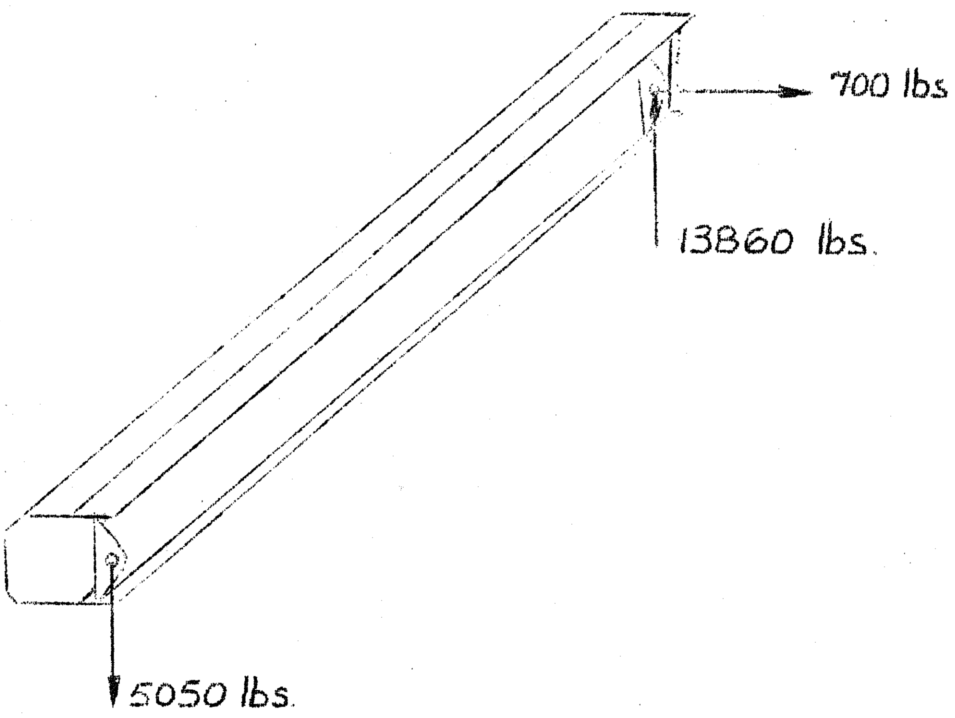
Rib 1 is subjected to a load of 5410 lbs. by the forward fitting and $(3100 - \Delta P)$ lbs. by the L.E. member. The shear load between the fitting and the wing spar is therefore $5410 - (3100 - \Delta P) = (2310 + \Delta P)$. Again, as ΔP is very small its effect can be neglected.



The load $(3100 - \Delta P)$ from the L.E. member actually relieves bending in the rib booms.

Generally, therefore, the effect of the loads ΔP on the spar and Rib 1 is favourable.

5 VERTICAL SPAR EXTENSION (Drg. No. M.160-3 Sheet 2)



(i) Lower Hinge Bracket (Wing) (Drg. No. M-160-5)

The bracket is attached to the spar via 4 - $\frac{3}{8}$" dia. H.T.S. Bolts (boom) and 8 - 5/16" dia. H.T.S. Bolts (web). The $\frac{3}{8}$" bolts are at 9.125" centres.

There is a moment at the web face of 5050 lbs x 2.05" (product of shear load and offset) which is assumed to be taken by the $\frac{3}{8}$" dia. bolts only. Tensile load in bolts therefore is :

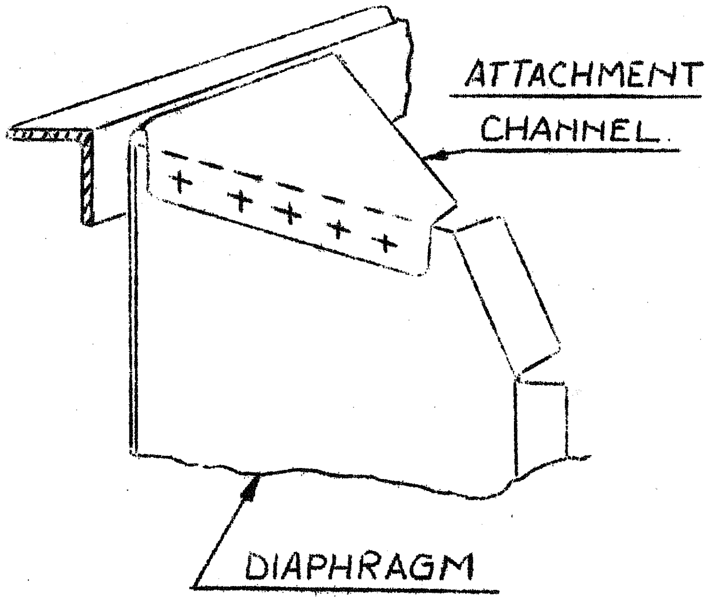
$$\frac{5050 \times 2.05}{9.125} = \underline{1,135 \text{ lbs.}}$$ > 2

The bolts are obviously O.K. at this load.

(ii) Diaphragm on Lower Hinge Bracket (Drg. M-160-3)

The shear load of 5050 lbs. is taken by the spar web, and the moment 10,350 lbs.ins. by the torsion box via the diaphragm. The diaphragm is 14 s.w.g. D.T.D.610, manufactured in two pieces and attached to the spar boom and web via the hinge bolts, and to the torsion box via $\frac{1}{8}$" dia. rivets.

R.F.



Bending moment is maximum at the diaphragm root. Assuming this load is taken entirely by the attachment channel, (see sketch) load in each flange of channel $= \frac{10350}{11.625} = \underline{890}$ lbs.

cross section area of flange $= 3'' \times 0.08''$

$\therefore$ Nom. Stress $= \frac{890}{3 \times 0.08} = \underline{3,700}\ \text{lb/in}^2$

Local Instability stress $f_I = KE\ (\frac{t}{b})^2$

$K = 0.58$

$\therefore\ f_I = \left[0.58 \times 10^7 \times (\frac{.08}{3})^2\right] = \underline{4,120}\ \text{lb/in}^2$ 1.15

iii) Torsion Box between hinges

Torque between hinges $= 2.05'' \times 5050$ lbs.
$= \underline{10,350}$ lb.ins.

Included area of torsion box $= \underline{43}\ \text{in}^2$

$\therefore$ Shear flow due to torque $= \frac{10,350}{2 \times 43}$
$= \underline{120}$ lb/in.

Rivets attaching box skin to diaphragm - $\frac{1}{8}''$ dia. at 1.00" max. pitch are obviously OK at this loading.
Torsion box skin is 18 s.w.g. DTD.610

$\therefore$ Applied shear stress $= \frac{120}{0.048} = \underline{2500}\ \text{lb/in}^2$

Maximum unsupported panel size is 9" x 5.25" the buckling stress of which is $\underline{5200}\ \text{lbs/in}^2$ (R.Ae.S. 02.03.05)

The panels, therefore, will not buckle. O.K.

R.F.

(iv) Spar web between Hinges

Shear flow in web due to 5050 lbs. applied

$$= \frac{5050}{10.44''} = \underline{483.0} \text{ lbs/in}$$

(10.44" = distance between boom centroids).

Total shear flow in web is therefore

$$483 + 120 = \underline{603} \text{ lb/in}$$

Web is 18 s.w.g. D.T.D.610.

$$\therefore \quad \text{Shear stress in web } q = \frac{603}{0.048}$$

$$= \underline{12,560} \text{ lb/in}^2$$

Maximum panel dimensions on spar web = 4.25" x 8.38" the buckling stress of which = $\underline{8800}$ lb/in^2 = qb

(R.Ae.S. 02.03.04)

Ratio $q/_{qb}$ = 1.43

The spar web will buckle under ultimate load but not under proof load.

Shear stress for failure in the web

$$> 0.445 \quad x \quad 56,000 = \underline{25,000} \text{ lb/in}^2$$ > 2

(02.03.13)

Since the spar booms and stiffener pitches are identical to those on the actual wing spar, and carry less load, the calculations for diagonal instability also hold for the spar extension but have an increased R.F. OK

(v) Torsion box between upper hinge and Rib 1

The diaphragm forward of the hinge transmits into the box a torque of 13,860 x 2.05 = $\underline{28,400}$ lb.in.

∴ Net torque from this diaphragm to Rib 1

= 28,400 - 10,350 = $\underline{18,050}$ lb.in.

∴ Shear flow in this portion of the box

$$= \frac{18,050}{2 \times 43} = \underline{210} \text{ lb/in}$$

Attachment to Rib 1 is made via 2 B.A. bolts and welded angle at box skin, and via Joint Plates etc. at the spar. Pitch of 2 B.A. bolts is approx. 2". Bolts obviously O.K. >2

Diaphragm to skin attachment carries a shear flow of 210 + 120 = 330 lb/in OK

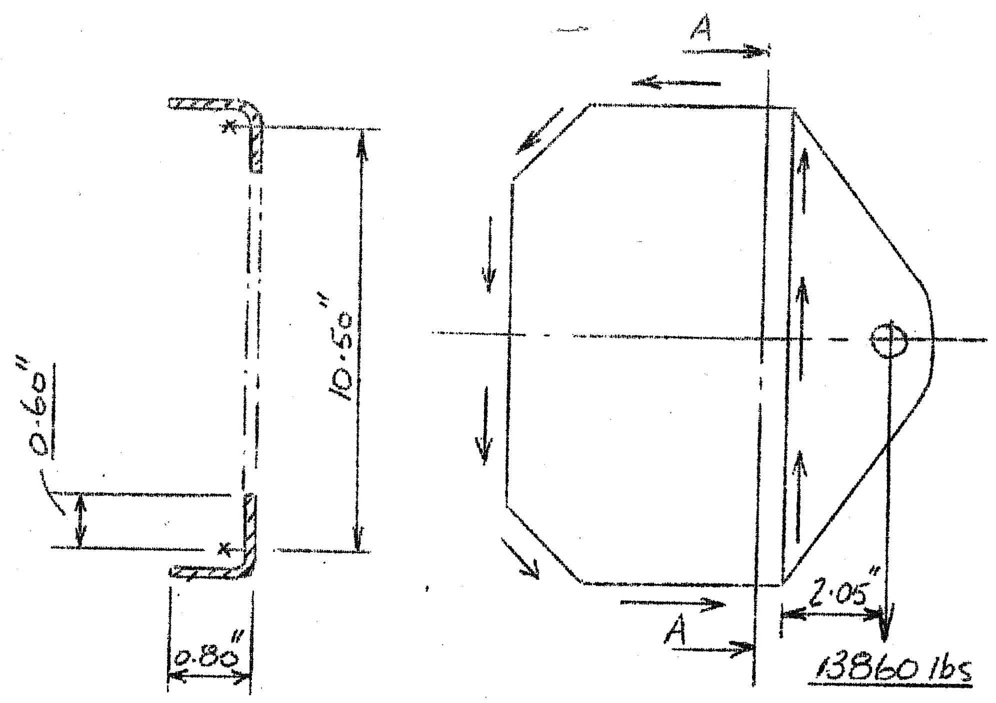
Shear Stress in the box = $\frac{210}{0.048}$ = $\underline{4380}$ lb/in^2 which is very small. The panels will not buckle.

(vi) Diaphragm forward of the Upper Hinge Bracket

Max. bending moment on the diaphragm is $\underline{28,400}$ lb.in

Assuming an effective boom section resisting this moment of 2 x 0.8" x 0.6" 14 s.w.g. D.T.D. 610 angles, their centroids being 10.5" apart.

R.F.



$\therefore$ Load on each angle $= \dfrac{28,400}{10.5} = \underline{2,700}$ lbs.

and the stress $= \dfrac{2,700}{1.40 \times 0.8} = \underline{24,100}$ lb/in^2

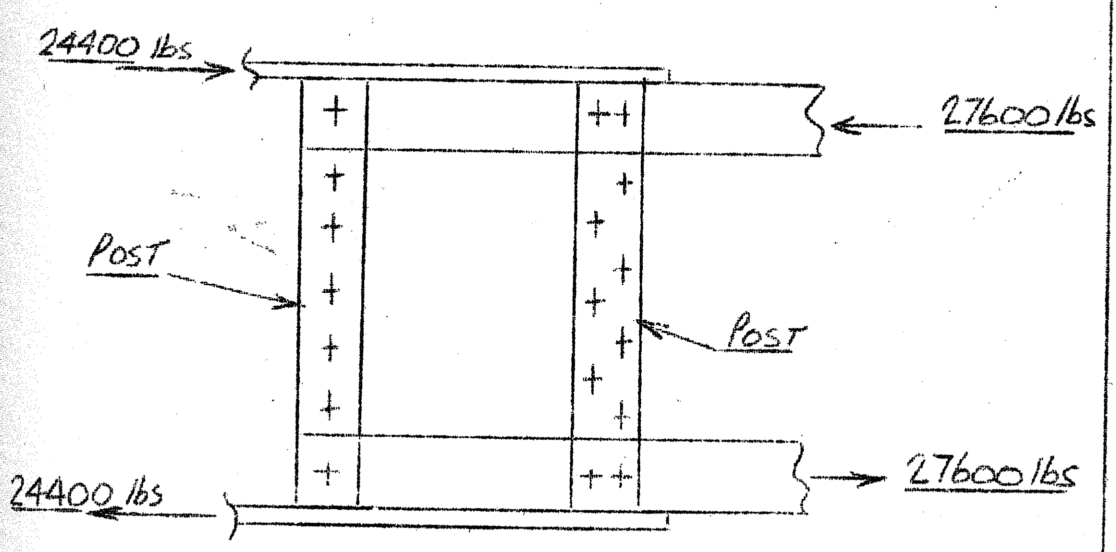
Allowable stress $= \underline{36,000}$ lb/in^2 1.5

The diaphragm flange is bolted to the Spar Joint Plate, no local instability will therefore occur.

(vii) Spar Extension at Rib 1

B.M. at Rib 1 = 287,600 lb.in.

Distance between spar boom centroids = 10.44"
and Joint Plate centroids 11.80"



R.F.

Load in spar boom = $\frac{287,600}{10.44}$ = $\underline{27,600}$ lbs.

Load in joint plate = $\frac{287,600}{11.80}$ = $\underline{24,400}$ lbs.

The difference, 3200 lbs. is balanced by the spar web. Moment on the spar boom due to the offset is balanced by equal and opposite loads on two posts, one at Rib 1 and one at the web joint. Their centres are 6.25" apart.

The moment at each spar boom =

$24400 \left(\frac{11.80 - 10.44}{2}\right)$ = $\underline{16,600}$ lb.ins.

$\therefore$ Load on each end of the posts

= $\frac{16,600}{6.25}$ = $\underline{2,660}$ lbs.

Shear flow in the panel due to the offset

= $\frac{3200}{6.25}$ = $\left(\frac{2 \times 2660}{10.44}\right)$ = $\underline{512}$ lb/in

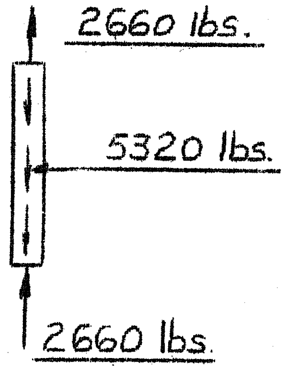
1. Post at Rib 1 (1" x 10 s.w.g.D.T.D.124)

aa $\frac{3}{8}$" dia. bolt in spar boom takes $\underline{2660}$ lbs.
Allowable load in single shear
= $\underline{8750}$ lbs. >2

bb Max. Stress in post-cross section area
= (1" x 0.128") = $\underline{0.128}$ in^2

$\therefore$ Max. f = $\frac{2660}{.128}$ = $\underline{20,800}$ lb/in^2

Allowable stress = $\underline{56,000}$ lb/in^2 >2



cc 2 BA Bolts in Web
Total load taken by the web
= (2 x 2660) = $\underline{5,320}$ lbs.
Bolts are at approx. 1.10" pitch and are obviously O.K.

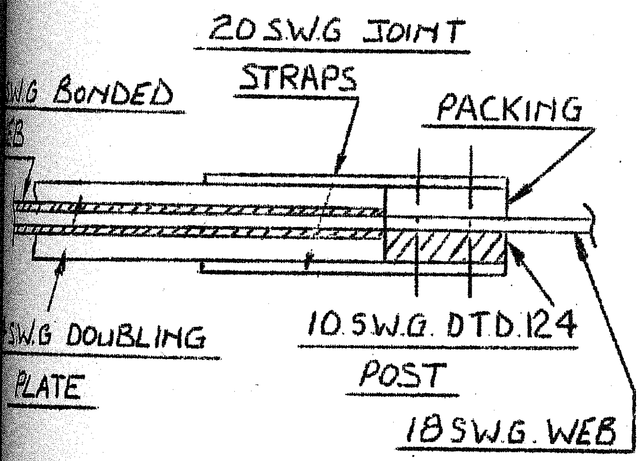
2. Post at Spar Web Joint

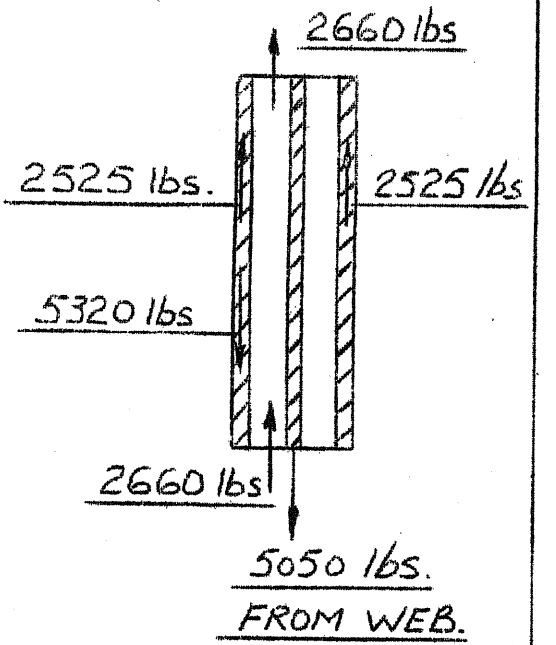
aa Rivets attaching post to web.
The loads of 5050 lbs. from the web, and 5320 lbs. from the post, are acting in opposite directions.

$\therefore$ Load at one shear face = (5320 - 2525)
= $\underline{2795}$ lbs.

This load is taken by 19 - 5/32" dia. rivets for which the allowable single shear strength in 20 s.w.g. DTD.610 is (19 x 500)
= $\underline{9500}$ lbs. >2

R.F.





<u>bb</u> <u>Attachment to Spar Boom</u> is made via 6 - 5/32" dia. rivets, single shear strength of which = (6 x 575) = <u>3,450</u> lbs.
Applied load = <u>2,660</u> lbs. 1.3

<u>cc</u> <u>Web Joint Strap</u> (20 s.w.g. D.T.D.610)

Shear due to load of 2795 lbs. in the strap is approx. 10,000 lb/in^2 which is acceptable.

3. <u>Shear in Spar Web between Upper Hinge and Rib 1</u>

Shear flow due to offset fitting of spar boom = <u>512</u> lb/in

Shear load in web between hinge and Rib 1 = (13860 - 5050) = <u>8,810</u> lbs.

Shear flow due to this load

$$= \frac{8810}{10.44} = \underline{843}\ \text{lbs/in}$$

Shear flow due to torque at same section = <u>210</u> lbs/in

∴ Total shear flow in web

$$= (512 + 843 + 210) = \underline{1565}\ \text{lb/in}$$

of this, 621 lbs/in is taken by the spar web across Rib 1 into the wing spar web; the remaining 944 lb/in is taken by the 14 s.w.g. D.T.D.124 doubling plate.

Shear stress in L.A. web

$$= \frac{621}{.048} = \underline{13,000}\ \text{lb/in}^2$$

and Shear stress in doubling plate

$$= \frac{944}{.08} = \underline{11,750}\ \text{lb/in}^2$$

The above stresses are acceptable and buckling will not occur.

Attachment of spar web to boom is made via $\frac{3}{8}$" dia. bolts at 1" pitch which is ample to take out the shear flow.

R.F.

(viii) Upper Hinge Bracket (Drg. M-160-6)
(Bolts)

Loads on the hinge bracket are :-
Shear of 13860 lbs. offset at 2.05" from the spar web; and a normal load of 700 lbs.
Attachment is made via 4 x $\frac{3}{8}$" dia. bolts and 8 x 5/16" dia. bolts in the boom and web respectively.

By inspection the bolt strength is adequate. OK

RF

-4 WING SKIN AND RIBS (Drgs. No. M-160-1 and M-160-2)

(i) Shear in Wing Skin

From Fig. 3, the maximum torque carried by the skin

$$T = 186,600 \text{ lb.in.}$$

Enclosed area of wing skin

$$A = 874 \text{ in}^2$$

∴ Shear flow in wing skin $= \dfrac{186600}{2 \times 874} = 107$ lbs/in

aa Joint at Rib 1

Allowing strength of glue joint to Rib 1 $= 600$ lb/in

Applied shear flow $= 107$ lb/in >2

bb Wing Skin Buckling

Wing skin is 16 s.w.g. D.T.D.710 reduxed to 3/64" birch ply.

Mean shear stress in L.A. neglecting any effect from the birch ply

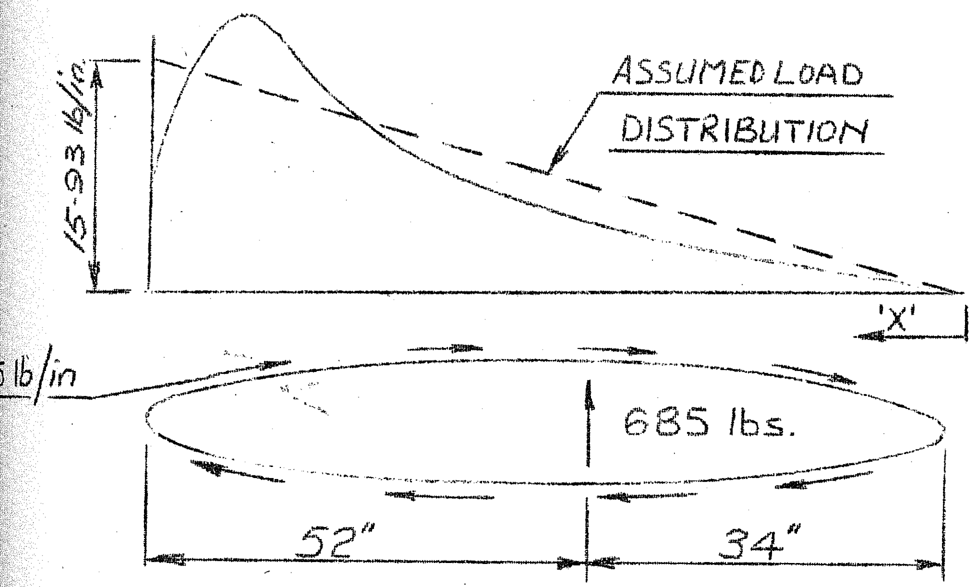
$$= \frac{107}{.064} = 1,670 \text{ lb/in}^2$$

A typical skin panel is 6" x 86".
The radius of curvature is so large that the panel is assumed flat for this calculation

Buckling Stress (R.Ae.S. 02.03.05) $= 5,500$ lb/in^2 > 2

The panel will not buckle under the applied load.

ii) Wing Rib (Drg. No. M-160-2)



Each rib has a distributed air load which is reacted by the wing spar web, and a torque which is balanced by a shear flow round the wing skin.

RF

From the spanwise load distribution table (Section B para. b.4) the maximum air load over a span of 96" is approx. 730 lbs. Since the rib pitch is 6", the maximum air load per rib

$$= \left(\frac{6}{96} \times 730\right) = \underline{457} \text{ lbs.}$$

A triangular chordwise load distribution curve is assumed, factored 1.5 to obtain a safe overall factor for the rib. The C.P. of the air load is therefore $(1/3 \times 86") = 28.63"$ from the L.E. or 23.37" from the spar datum, which approximates favourably with that stated in Section B para. b.4.

Torque due to air load

$$T = \left(23.37" \times 457 \times 1.5\right) = \underline{16{,}000} \text{ lb.in.}$$

This torque is balanced by a shear flow round the rib

$$q = \frac{T}{2A} = \frac{16000}{(2 \times 876)} = \underline{9.15} \text{ lb/in}$$

From this loading the vertical shear and BM diagram can be obtained (Fig.4)

aa <u>Shear in Rib Web</u>

From fig. 4 maximum shear occurs at the spar web ℄

Max. shear $= \underline{460}$ lbs

Rib web depth $= \underline{12"}$

$$\therefore \text{ Shear flow } = \frac{460}{12} = \underline{38.3} \text{ lb/in}$$

This load is very small, the rib web - spar web joint is therefore adequate.

Rib web is 3/32" ply.

$$\therefore \text{ Shear stress in web } = \frac{38.3}{.0938} = \underline{407} \text{ lbs/in}^2$$

which again is very small and obviously OK.

bb <u>Rib Boom Bending</u>

Maximum BM in Rib at spar web ℄ (Fig.4) $= \underline{7{,}600}$ lb.in.

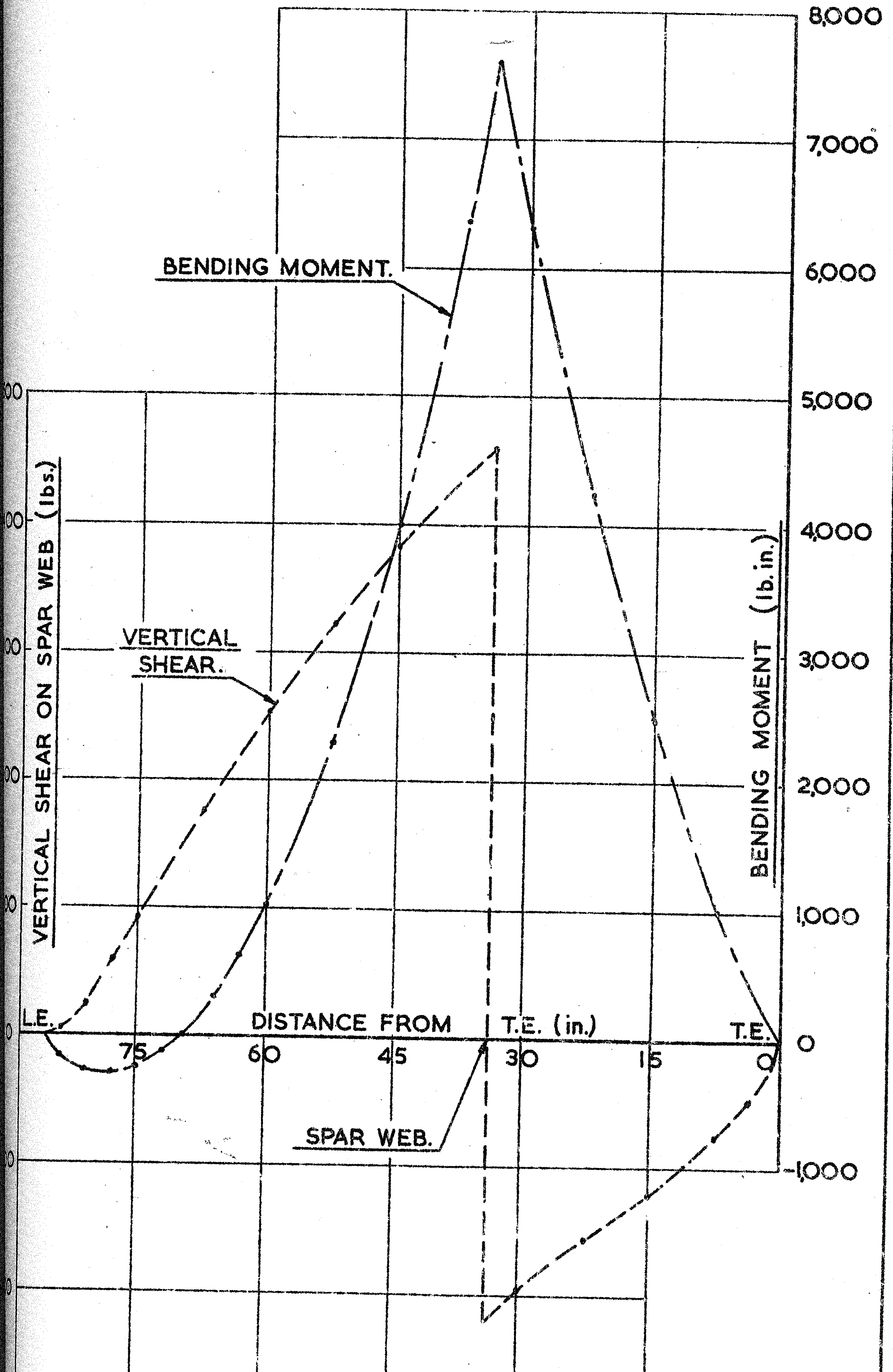
Depth between rib centroids $= \underline{12"}$

$$\therefore \text{ Rib boom end load } = \frac{7600}{12} = \underline{633} \text{ lbs.}$$

Applied stress in boom

$$= \frac{633}{(.5 \times .5)} = \underline{2530} \text{ lb/in}^2$$

Allowable compressive stress $\underline{4400}$ lb/in^2 1.73

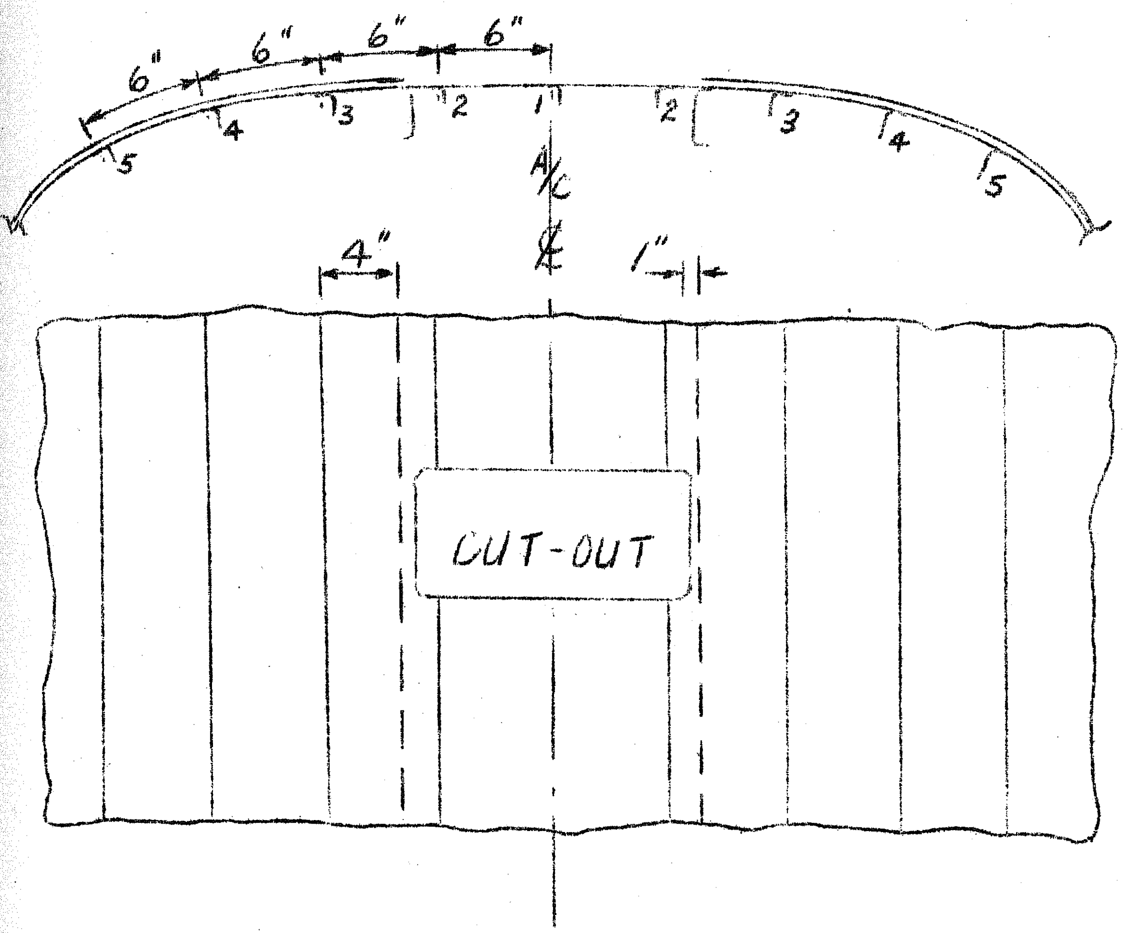


SHEAR & BENDING DIAGRAM FOR WING RIB.

FIG. 4.

D DETAIL STRESSING OF FUSELAGE MODIFICATIONS AND ATTACHMENTS

d-1 Fuselage Cut-outs (Drg. No. M.170-2)



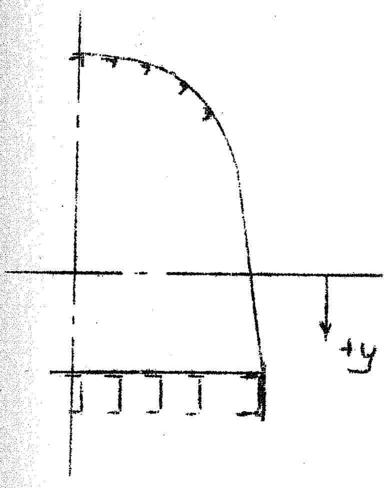
The original fuselage section is of 22 swg (.028") D.T.D. 390 skin with stringers at 6.00" pitch. The stringers are 12 swg (.104") angle sections of 0.22 in^2 area. 3 cut-outs are made on the top of the fuselage to accommodate the spar and the for'd and aft fittings. A doubling skin of 18 swg (.048") D.T.D. 610 is riveted in position over the cut-outs extending fore and aft between formers 18 and 24 and laterally to stringer No.5 port and starboard. Dimensionally, the doubling skin extends 22.00" for'd and 10.50" aft of the for'd and aft cut-out edges respectively. In addition, a 3.00" x 0.75" x 18 swg channel section longeron is positioned on each side of the cut-outs running parallel to the fuselage axis, again between formers 18 and 24. Built up box section transverse members are also included at each cut-out position, running between the longeron faces.

i) Bending of Fuselage

A bending calculation of the original fuselage section (without cut-out or reinforcement) has been carried out using a bending moment of 1520 ton ins. This is the maximum quoted in the "Lancaster" type record between formers 18 and 24. Results are summarised in the following table:-

Station	f_{bm}	A_e	y	$y\ A_e$	y^2A_e	$f = \frac{My}{I}$	$P = f\ A_e$
1	-2900	0.148	-49.2	-7.29	359	-15,370	-2,270
2	-2900	0.298	-46.8	-13.94	653	-14,600	-4,350
3	-4380	0.311	-45.6	-14.18	646	-14,220	-4,420
4	-5860	0.324	-43.3	-14.03	607	-13,500	-4,370
5	-5860	0.330	-39.5	-13.03	515	-12,320	-4,070
6	-5860	0.337	-34.9	-11.77	411	-10,900	-3,670
7	-3920	0.323	-29.5	-9.53	281	-9,200	-2,970
8	-1280	0.295	-23.7	-6.99	166	-7,400	-2,180
9	-1280	0.303	-17.3	-5.24	91	-5,400	-1,640
10	-1280	0.314	-11.9	-3.74	44	-3,710	-1,170
11	-	0.358	-5.9	-2.11	12	-1,840	-660
12	-	0.388	0	0	0	0	0
13	-	0.361	5.0	1.81	9	1,560	560
14	-	0.413	10.1	4.17	42	3,150	1,300
Floor	-	2.028	15.5	31.43	487	4,830	9,810
Floor Members	-	3.679	17.5	64.38	1,127	5,460	20,100
Σ		10.21		-0.06	5,450		0

= 0



Notation:-

f_{bm} = Mean buckling stress at station (lb/in^2)

A_e = Effective stringer area

$= A_s + \frac{1}{3}bt + \frac{2}{3}bt\,\frac{f_{bm}}{f}$

y = Distance from neutral axis (in.)

I = Moment of inertia of effective section

$= 2\sum A_e y^2 = \underline{10,900}$ in^4

f = Stress due to bending

$= \frac{My}{I} = \underline{312}\,y$ (lb/in^2)

P = Stringer end load (lb.)

$= fA_e$

Panel buckling stress:-

Panel	b (in.)	t (in.)	R (in.)	K	f_b (lb/in^2)
1 - 2	6.0	0.028	47.6	13.3	2,900
2 - 3	"	"	"	"	"
3 - 4	"	"	22.5	26.9	5,860
4 - 5	"	"	"	"	"
5 - 6	"	"	"	"	"
6 - 7	"	"	"	"	"
7 - 8	"	"	∞	5.85	1,280
8 - 9	"	"	"	"	"
9 - 10	"	"	"	"	"
10 - 11	"	"	"	"	"
11 - 12	"	"	"	"	"
12 - 13	"	"	"	"	"
13 - 14	4.1	"	"	"	"
14-Floor	5.6	"	"	"	"

ii) Edge Loads at Cut-out

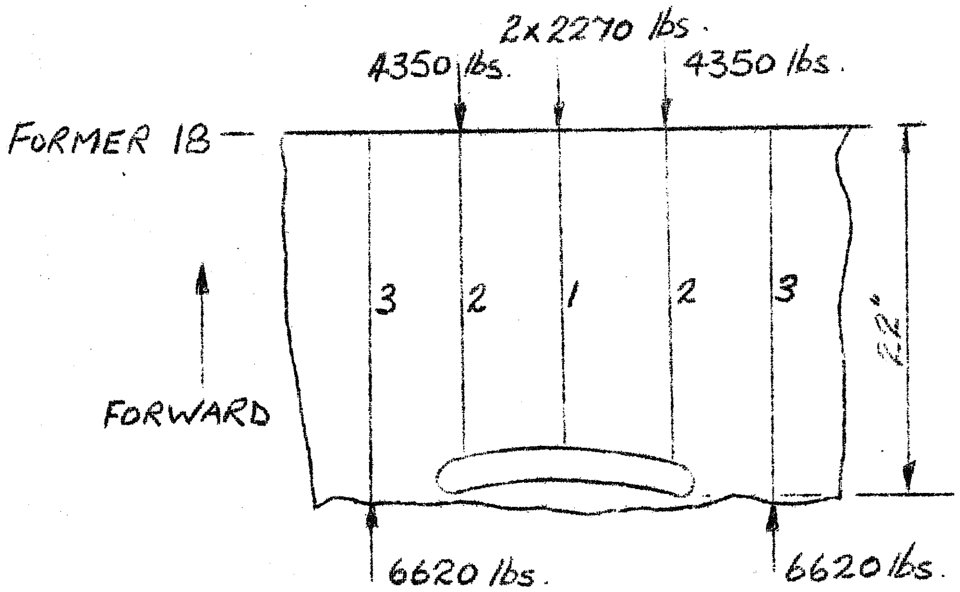
All the area of the doubling plate from the edge of the cut-out to stringer No.5 is assumed to be concentrated at the position of stringer No.3, but corrected in the ratio of the square of the distance to the neutral axis in order to give a moment of inertia equal to that of the original plate.

The total load (6620 lb) from discontinued stringers 1 and 2 is assumed to be carried by this equivalent area and is distributed among the bays of the doubling plate in proportion to it.

Bay	Edge of Cut-out Str.3	Str.3-Str.4	Str.4-Str.5
Area = bt(in^2)	0.24	0.288	0.288
y(from N.A.) (in.)	46.2	44.5	41.4
y^2 (in^2)	2134	1980	1714
(At Str.3 Y_3 = 45.6 in y_3^2 = 2079 in^2)			
y^2/y_3^2	1.026	0.953	0.825
$a_e = bt(y^2/y_3^2)(in^2)$	0.246	0.274	0.238
$A_e = \sum a_e$ = 0.758 in^2			
a_e/A_e	0.324	0.362	0.314
= $6620(a_e/A_e)$	2180	2430	2110
$f = P/_{bt}(lb/in^2)$	9080	8440	7320
b (in.)	4.0	6.0	6.0
t (in.)	0.048	0.048	0.048
R (in.)	47.6	22.5	22.5
K(R.Ae.S. 02.01.10)	7.0	18.5	18.5
f_b (lb/in^2)	10080	11840	11840
f_b/f	1.11	1.40	1.62

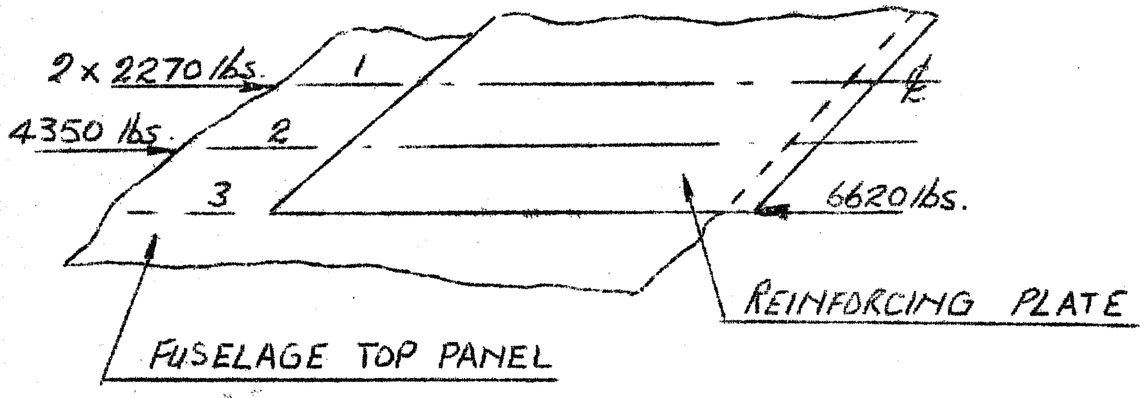
From the above it is evident that the doubling plate will not buckle under the loading conditions.

iii) Diffusion of Edge Load into Fuselage (For'd)



The edge load of 6620 lbs is assumed to diffuse back into stringers No. 1 and 2 over the length of the doubling plate extending from the cut-out to former No.18 as indicated in the above diagram.

1) Riveting



Rivet pitch at stringers 1 and 3 = 0.50"

Rivet pitch at stringer 2 = 1.00"

Total No. of rivets to transmit the edge load on one side of the doubling plate back into the fuselage top panels will consist of the sum of all the rivets along stringers 2 and 3 and half the rivets along stringer 1.

No. of rivets = 88

R.F.

∴ Load per rivet = $\frac{6620}{88}$

= 76 lbs

Assuming a stress concentration factor of 4 around the corner of the cut-out.

∴ Max. rivet load = (4 x 76) = 304 lbs

Strength of 5/32" dia. L.69 rivets in single shear = 520 lbs (in 18 swg D.T.D. 610) 1.71

2) Shear Stress on Fuselage Panel

Shear flow in panel 2-3 (2 rivets per inch) = (76 x 2)

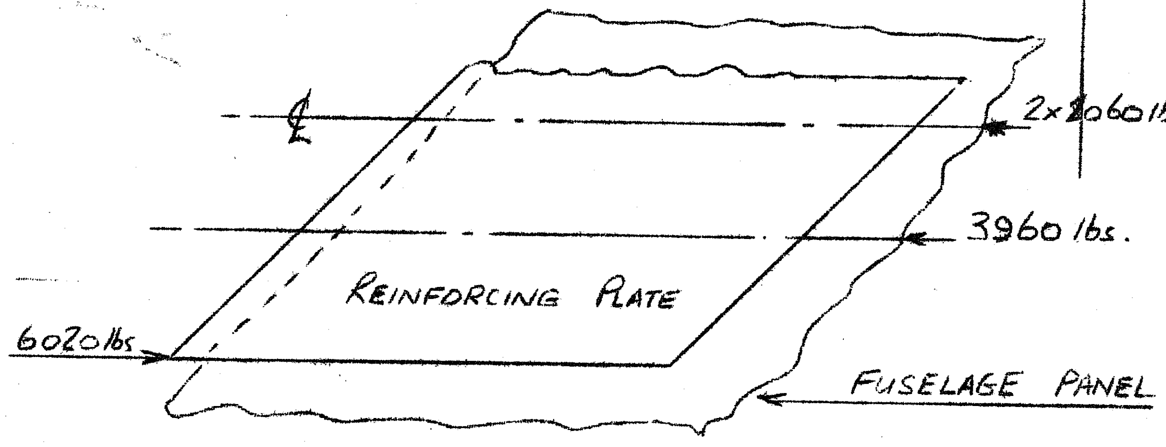
= 152 lb/in

$$\text{Shear stress} = \frac{152}{.028} = 5430 \text{ lb/in}^2$$

Panel width is reduced to 4" by the addition of the longeron, and the doubling plate provides an appreciable stabilising effect. Buckling of the panel, therefore, is most unlikely under the loading conditions.

iv) Diffusion of Edge Load into Fuselage (Aft)

The edge load is assumed to diffuse back in a similar manner to the previous calculation but, in this case, diffusion takes place between the aft cut-out and former No.24, a distance of approx. 10.5" as opposed to approx. 22" in the for'd case. The max. bending moment, however, over the region of the aft cut-out is quoted in the "Lancaster" type record as 1380 tons ins. compared with the for'd cut-out condition of 1520 tons ins. The edge load and the loads in the stringer can therefore be reduced accordingly by approx. 9%.



R.F.

Rivet pitches along the stringers are the same as before but we may assume that the longeron rivets at 1.00" pitch are also effective.

Total No. of rivets = 53

Load per rivet $= \frac{6020}{53} = \underline{114}$ lbs

Again assuming a stress concentration factor of 4:-

Max. rivet load = $(4 \times 114) = \underline{456}$ lbs

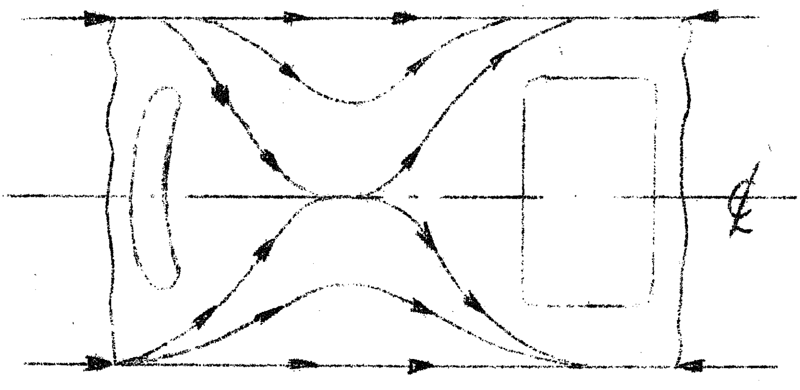
Allowable load = $\underline{520}$ lbs. 1.14

Shear flow in panel 2-3 = $(114 \times 2) = \underline{228}$ lbs

$\therefore$ Shear stress $= \frac{228}{.028} = \underline{8140}$ lb/in^2

Again, the panel is unlikely to buckle under these conditions.

v) <u>Diffusion of Edge Load Between Cut-outs</u>

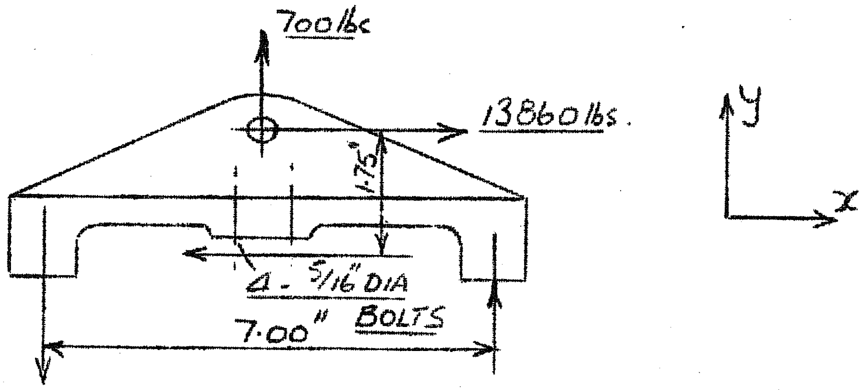


A calculation has been carried out to ascertain the correct distribution of load between the cut-outs and it is found that, owing to the short length available for diffusion coupled with the excess of stiffness of the edge members over the stringers, only part of the load is carried to the centre line - the remainder going straight through. The condition between the cut-outs, therefore, is not critical compared with the ends and no detail is necessary.

d-2 Fuselage Former No. 22 — R.F.

Loads and Stresses Due to Vertical Spar Extension

i) Upper Hinge Bracket (Fuselage) (Drg.No. M 167-5)



Loads on bracket:- 13,860 lb in direction 'X'

700 lb in direction 'Y'

Tension in the bolts due to the 700 lb load is very small and may be neglected.

Side load of 13,860 lb. is assumed to be taken out by the 4 - 5/16" dia. bolts at the bracket centre and transmitted directly into the support tubes.

Load per bolt $= \dfrac{13,860}{4} = \underline{3,460}$ lb

Allowable single shear - 5/16" B.S.F. H.T.S. = 6,200 lb — 1.79

Offset moment = (13,860 x 1.75) = 24,250 lb.ins

This induces tension and compression on the $\frac{3}{8}$" dia. former attachment bolts at 7" c'rs.

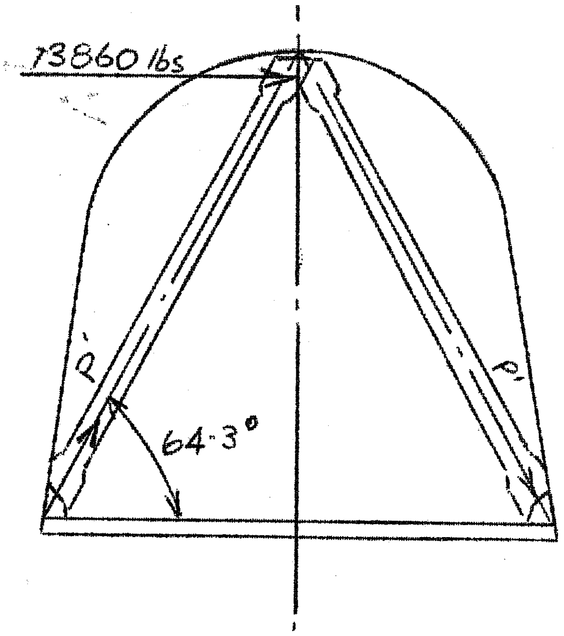
$\therefore$ Load per side $= \dfrac{24,250}{7} = \underline{3470}$ lb.

Load per bolt = 1,490 lb

Allowable tensile load - $\frac{3}{8}$" B.S.F. H.T.S. = 9,350 lb — > 2

No calculations are necessary on the bracket itself - this is obviously adequate.

ii) Support Tubes at Former 22 (Drg. No. M.167-1)



R.F.

Load in tubes $P' = \frac{13,860}{2\cos 64.3^\circ} = \underline{16,000}$ lb

1) Compression

$2\frac{5}{8}$" Dia. x 17 swg T.45, $\ell = \underline{68"}$

assume struts to be pin-ended.

$A = \underline{.451}$ in^2 $\quad k = \underline{.9081"}$ $\quad c = \underline{.628"}$

$\frac{e}{c} = \underline{.15}$ $\quad \frac{\ell}{k} = \underline{75}$

Allowable stress $f = \underline{42,000}$ lb/in^2

$\therefore$ Allowable load $= \underline{19,000}$ lb

Applied load $= \underline{16,000}$ lb 1.19

2) Instability of Swaged Ends

Due to lack of information on the probable behaviour of the swaged ends, a compression test to destruction was carried out on an identical strut simulating the actual end conditions (see section 6, report No. T.S.M. 145). Conclusions resulting from the test indicate that there is no permanent deformation at a load of 7.5 tons (16,800 lb.) and final failure (due to instability) does not occur until a load of 10.5 tons (23,500 lb.) is reached. 1.47

) Floor Attachment Bracket (Drg. No. M 167-1/5)

1) Support tube attachment - 7-5/16" dia. bolts

Average load per bolt $= \frac{16,000}{7} = \underline{2,290}$ lb

Assuming a factor of 1.2 for uneven load distribution

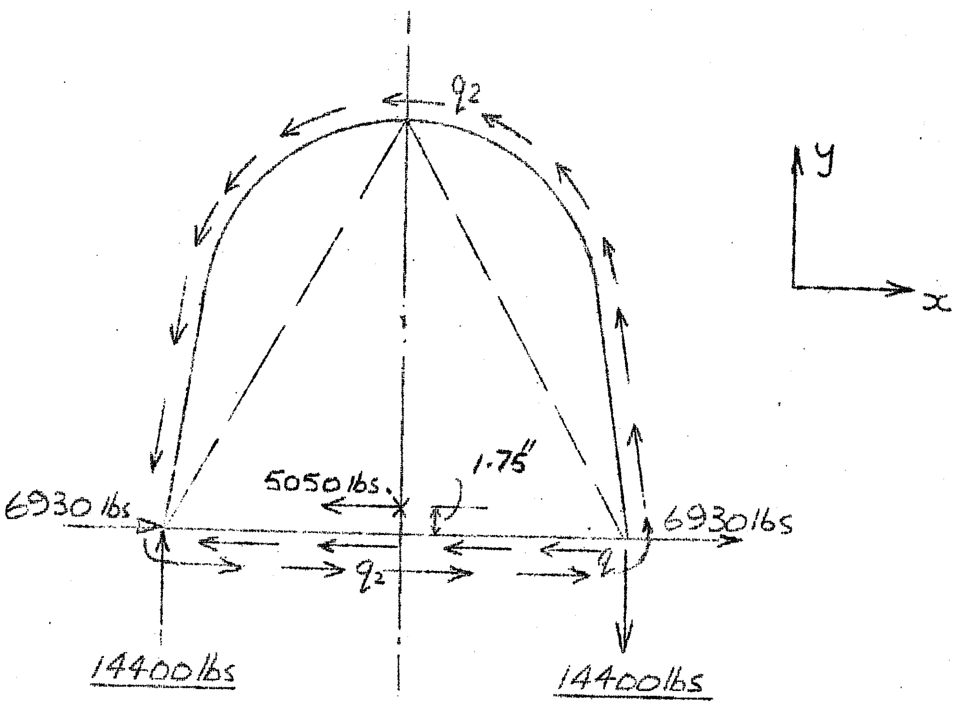
Max. load = (1.2 x 2290) = $\underline{2,470}$ lb

Single shear strength = $\underline{6,200}$ lb > 2

Allowable bearing load (.224" S.3) = $\underline{5,800}$ lb. > 2

By inspection the bolts attaching the bracket to the fuselage (10 - 5/16" dia., 2 - $\frac{1}{4}$" dia.) are obviously satisfactory.

iv) Fuselage Former No. 22



We have previously assumed, for stressing purposes, that the total side load of 13,860 lb. is taken by the support tubes. In actual fact, a percentage of this load is transferred directly into the former itself and a detailed treatise will subsequently be given. For the time being we may continue on the original assumption.

1) Loading conditions at the corners will be 6,930 lb. and 14,400 lb. respectively in directions 'X' and 'Y' (see sketch) and, in addition, a side load of 5,050 lb. from the vertical spar will be applied at the former ℄ , 1.75" above the floor. Assuming that all bending is taken by the "pi" section longerons concentrated at the corners, then the shear flow between the corners is constant.

Resultant load in direction 'X' = (13,860 - 5050)
= 8810 lb

Shear flow along floor - $q_1 = \frac{8810}{65} = \underline{136}$ lb/in

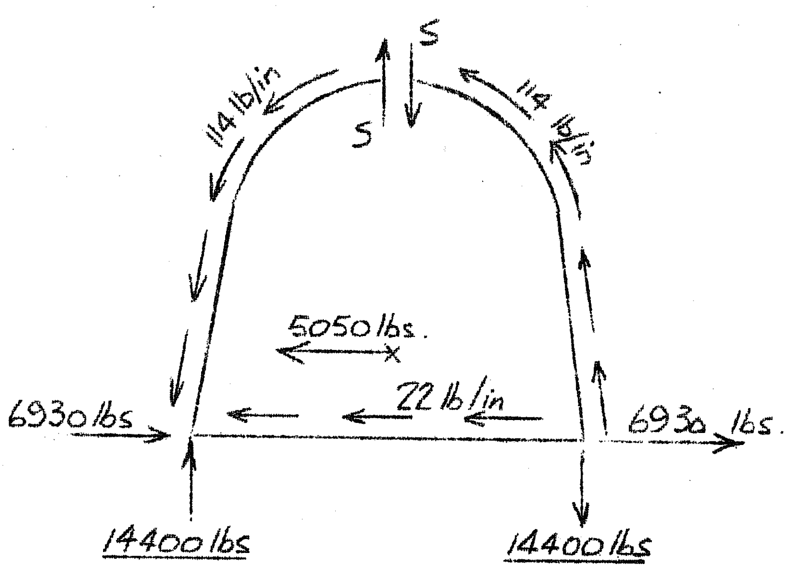
Floor member ℄ is 2" below floor skin

∴ Torque about floor skin - T = (14,400 x 60.3) - (5,050 x 1.75) - (13,860 x 2) = 830,940 lb.in.

∴ Shear flow around former - $q_2 = \frac{T}{2A} =$ 114 lb/in.

where A = Enclosed area = 3,650 in^2

R.F.



Consider the former cut at the top ℄ .

Since the former is symmetrical and the loading assymmetrical the only unknown force at the cut is a shear 'S'.

Let the statically determinate bending moment on the frame be M_o and let the bending moment due to S = 1 lb be m.

Total bending moment $M = M_o + S.m$

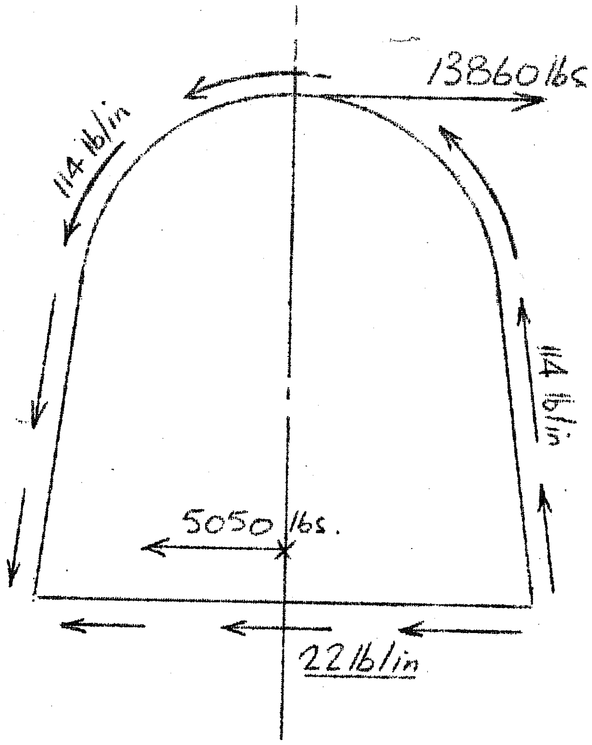
Strain energy $U = \int \frac{M^2}{2EI}\, ds$

From $\frac{\delta U}{\delta S} = 0$, we have $S = - \dfrac{\int \frac{mM}{EI} . ds}{\int \frac{m^2}{EI} . ds}$

For strain energy bending calculation see table I p. 4D-14

2) Assuming, now, that the total side load is transferred directly into the former, by-passing the support tubes entirely:-

R.F.



Using the same assumption, shear flows in the fuselage skin and floor skin are again 114 lb/in. and 22 lb/in. respectively.

Once more, for strain energy bending calculation see Table I p. 4D-14.

3) The true side load distribution will be as follows:

k% in the support tube

(1 - k)% directly into the former.

If P_o = Load in support tube under total load conditions

then true load in support tube - $P' = kP_o$.

M_1 = Bending in former with total load in support tube

M_2 = Bending in former with total load in former.

$\therefore$ Total bending moment in frame $M = kM_1 + (1 - k)M_2$

Then total strain energy in former and support tube:-

$$U = \int \frac{M^2}{2EI} ds + \frac{P'^2 \ell}{2AE}$$

which should be a minimum hence:-

$$\frac{\delta U}{\delta k} = \int \frac{2M \frac{\delta M}{\delta k}}{2EI} ds + \frac{2P' \frac{\delta P'}{\delta k} \ell}{2AE}$$

$$= \int \frac{M(M_1 - M_2)}{EI} ds + \frac{P' P_o \ell}{AE}$$

R.F.

$$\therefore \int \frac{k(M_1 - M_2)^2 + M_2(M_1 - M_2)}{EI} ds + \frac{kP_o^2 \ell}{AE} = 0$$

hence
$$k = - \frac{\int \frac{M_2(M_1 - M_2)}{EI} ds}{\frac{P_o^2 \ell}{AE} + \int \frac{(M_1 - M_2)^2}{EI} ds}$$

P_o = 16000 lb. , ℓ = 68" , A = .451 in^2 and E = 10^7 lb/in^2

$$\therefore \frac{P_o^2 \ell}{AE} = \underline{3,860} \text{ lb.ins}$$

Integrals:- (For detailed calcs. see Table II p. 4D-15)

$$\int \frac{M_2(M_1 - M_2)}{EI} ds = - \underline{23,099} \text{ lb.ins}$$

$$\int \frac{(M_1 - M_2)^2}{EI} ds = \underline{45,439} \text{ lb.ins}$$

$$\therefore k = - \frac{(-23,099)}{3,860 + 45,439} = \underline{0.4685}$$

or $\underline{46.85\%}$

Hence bending moment in frame

$$M = (0.4685 M_1 + 0.5315 M_2)$$

See Table II p.4D-15 for tabulated values of M

See Fig. 5 p.4D-27 for bending moment diagram.

TABLE I

St'n	ds	$\frac{ds}{I}$	m	$m. \frac{ds}{I}$	$m^2. \frac{ds}{I}$	$(M_o)_1$	$(M_o)_1\ m. \frac{ds}{I}$	$M_1=(M_o)_1+S_1 m$	$(M_o)_2$	$(M_o)_2 m. \frac{ds}{I}$	$M_2=(M_o)_2+S_2 m$
1	4	2.24	2.0	4.48	9	460	2,100	-6,700	-230	-1,000	9,900
2	"	"	6.0	13.44	81	1370	18,400	-20,100	-2,100	-28,000	28,300
3	"	"	10.0	22.40	224	2740	61,400	-33,100	-5,600	-125,000	45,100
4	"	"	14.0	31.36	439	4790	150,200	-45,300	-12,500	-392,000	58,400
5	"	"	17.7	39.65	702	7980	316,400	-55,400	-21,800	-864,000	67,900
6	"	"	20.8	46.59	969	12,800	596,400	-61,700	-35,000	-1,631,000	70,400
7	"	"	23.1	51.74	1195	20,100	1,040,000	-62,600	-49,200	-2,546,000	67,800
8	"	"	24.9	55.78	1389	29,400	1,640,000	-59,800	-64,200	-3,581,000	62,000
9	"	"	26.1	58.46	1526	40,100	2,344,200	-53,400	-79,800	-4,665,000	52,400
10	"	"	27.1	60.70	1645	51,500	3,126,000	-45,500	-95,400	-5,791,000	41,900
11	"	"	27.9	62.50	1744	63,400	3,962,500	-36,500	-110,500	-6,906,000	30,900
12	"	"	28.7	64.29	1845	75,700	4,866,700	-27,100	-125,300	-8,056,000	20,100
13	"	"	29.4	65.86	1936	88,500	5,828,600	-16,800	-139,500	-9,187,000	9,500
14	"	"	30.0	67.20	2016	101,700	6,834,200	-5,700	-153,300	-10,302,000	-1,300
15	"	"	30.5	68.32	2084	115,400	7,884,100	6,200	-166,600	-11,382,000	-12,100
16	"	"	31.0	69.44	2153	129,300	8,978,600	18,300	-180,500	-12,534,000	-23,400
17	"	"	31.4	70.34	2209	143,600	10,100,800	31,200	-193,200	-13,590,000	-34,100
18	"	"	31.7	71.01	2251	158,200	11,233,800	44,700	-206,300	-14,649,000	-45,700
19	"	"	32.0	71.68	2294	173,100	12,407,800	58,500	-219,100	-15,705,000	-57,000
20	"	"	32.2	72.13	2323	188,100	13,567,700	72,800	-231,900	-16,727,000	-68,700
21	"	1.36	28.0	38.04	1065	186,800	7,105,900	86,500	-216,000	-8,217,000	-74,100
22	"	"	24.0	32.61	783	158,800	5,178,500	72,900	-186,400	-6,079,000	-64,800
23	"	"	20.0	27.17	543	130,800	3,553,800	59,200	-156,800	-4,260,000	-55,500
24	"	"	16.0	21.74	348	102,800	2,234,900	45,500	-127,200	-2,765,000	-46,200
25	"	"	12.0	16.30	196	74,800	1,219,200	31,800	-97,600	-1,591,000	-36,900
26	"	"	8.0	10.87	87	46,800	508,700	18,200	-68,000	-739,000	-27,500
27	"	"	4.0	5.43	22	18,800	102,100	4,500	-38,400	-209,000	-18,300
28	2	0.68	1.0	0.68	1	-1,900	-1,300	-5,500	-15,900	-11,000	-11,300
					Σ = 32,079		Σ = 114,861,700			Σ = -162,535,000	

$$S_1 = -\frac{114,861,700}{32,079} = -\underline{3581}$$

$$S_2 = -\frac{(-162,535,000)}{32,079} = \underline{5067}$$

TABLE II

Stn	$\frac{ds}{I}$	M_2	$M_1 - M_2$	$\frac{1}{E} M_2(M_1-M_2)$	$\frac{M_2(M_1-M_2)ds}{EI}$	$\frac{1}{E}(M_1-M_2)^2$	$\frac{1}{EI}(M_1-M_2)^2 ds$	$0.4685M_1$	$0.5315M_2$	M
1	2.24	9,900	-16,600	-16.43	-36.8	27.56	61.7	-3,140	5,260	2,100
2	"	28,300	-48,400	-136.97	-306.8	234.25	524.7	-9,420	15,040	5,600
3	"	45,100	-78,200	-352.68	-790.0	611.52	1369.8	-15,510	23,970	8,500
4	"	58,400	-103,700	-605.61	-1356.6	1075.37	2408.8	-21,220	31,040	9,800
5	"	67,900	-123,300	-837.21	-1875.3	1520.29	3405.4	-25,950	36,090	10,100
6	"	70,400	-132,100	-929.98	-2083.2	1745.04	3908.9	-28,910	37,420	8,500
7	"	67,800	-130,400	-884.11	-1980.4	1700.42	3808.9	-29,330	36,040	6,700
8	"	62,000	-121,800	-755.16	-1691.6	1483.52	3323.1	-28,020	32,950	4,900
9	"	52,400	-105,800	-554.39	-1241.8	1119.36	2507.4	-25,020	27,850	2,800
10	"	41,900	-87,400	-366.21	-820.3	763.88	1711.1	-21,320	22,270	1,000
11	"	30,900	-67,400	-208.27	-466.5	454.27	1017.6	-17,100	16,420	-700
12	"	20,100	-47,200	-94.87	-212.5	222.78	499.0	-12,700	10,680	-2,000
13	"	9,500	-26,300	-24.99	-56.0	69.17	154.9	-7,870	5,050	-2,800
14	"	-1,300	-4,400	+0.57	+1.3	1.94	4.3	-2,670	-690	-3,400
15	"	-12,100	18,300	-22.14	-49.6	33.49	75.0	2,900	-6,430	-3,500
16	"	-23,400	41,700	-97.58	-218.6	173.89	389.5	8,570	-12,440	-3,900
17	"	-34,100	65,300	-222.67	-498.8	426.41	955.2	14,620	-18,120	-3,500
18	"	-45,700	90,400	-413.13	-925.4	817.22	1830.6	20,940	-24,290	-3,400
19	"	-57,000	115,500	-658.35	-1474.7	1334.01	2988.2	27,410	-30,300	-2,900
20	"	-68,700	141,500	-972.11	-2177.5	2002.22	4485.0	34,110	-35,610	-1,500
21	1.36	-74,100	160,600	-1190.05	-1616.9	2579.24	3504.4	40,530	-39,380	+1,100
22	"	-64,800	137,700	-892.30	-1212.4	1896.13	2576.3	34.150	-34,440	-300
23	"	-55,500	114,700	-636.59	-864.9	1315.61	1787.5	27,740	-29,500	-1,800
24	"	-46,200	91,700	-423.65	-575.6	840.89	1142.5	21,320	-24,550	-3,200
25	"	-36,900	68,700	-253.50	-344.4	471.97	641.3	14,900	-19,610	-4,700
26	"	-27,600	45,800	-126.41	-171.8	209.76	285.0	8,530	-14,670	-6,100
27	"	-18,300	22,800	-41.72	-56.7	51.98	70.6	2,110	-9,730	-7,600
28	0.68	-11,300	-5,800	+6.55	+4.5	3.36	2.3	-2,580	-6,010	-8,600
					Σ = -23,099		Σ = 45,439			

R.F.

Former moments of inertia used in strain energy calculations;-

Above floor line (St'ns. 1-20) - 4.00" x 1.5" x 10 swg D.T.D. 390 Channel section.

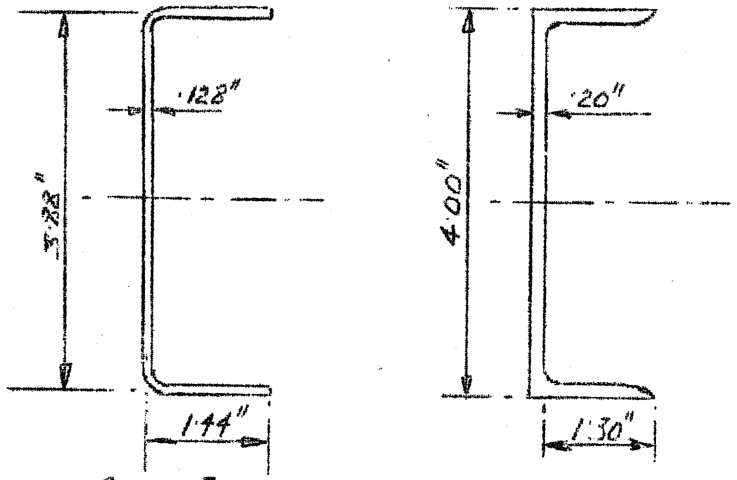
$$\therefore I \approx \left(\frac{1}{12} \times 3.88^3 \times .128\right) + \left(2 \times 1.44 \times .128 \times 1.9^2\right)$$

$$= \underline{1.787} \text{ in}^4$$

Below floor line (St'ns. 21-28) - 4.00" x 1.5" x 0.2" extruded light alloy channel

$$\therefore I = \left(\frac{1}{12} \times 0.2 \times 4.0^3\right) + \left(2 \times 1.3 \times 0.2 \times 1.9^2\right)$$

$$= \underline{2.944} \text{ in}^4$$



ABOVE FLOOR　　　BELOW FLOOR

Max. bending moment on former - $M_{Max.} = \underline{10,100}$ lb.in. at Stn No. 5

$$\therefore \text{ Stress on former due to bending} - f = \frac{My}{I} = \frac{10100 \times 1.9}{1.787}$$

$$= \underline{10,730} \text{ lb/in}^2$$

$$\text{Local instability stress on flanges} - f_c = 0.58E_t\left(\frac{.128}{1.44}\right)^2$$

$$= \underline{28,000} \text{ lb/in}^2$$

> 2

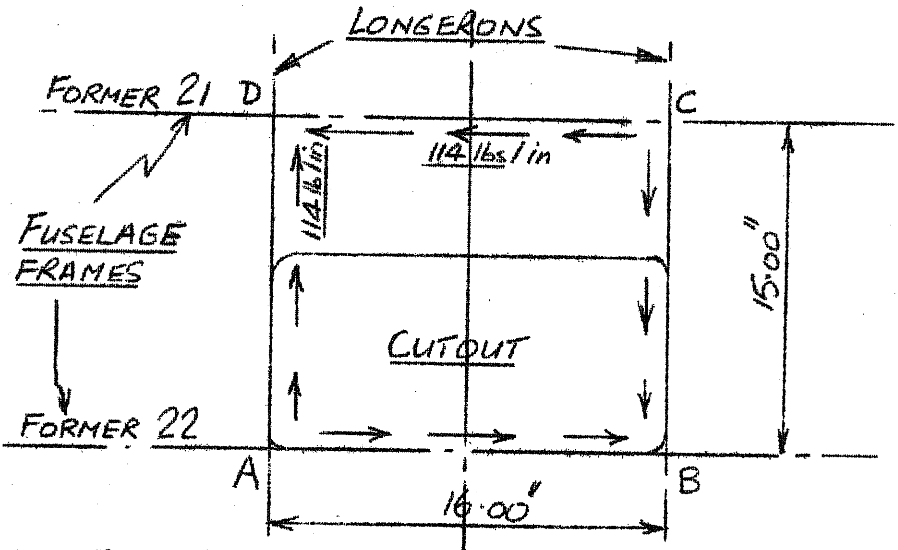
R.F.

The former web is cut locally to allow stringer continuity, however, if we assume an effective skin area the loss is made good. The bending moment will, of course, be a maximum at the top of the former and, in this region the existing 22 swg fuselage skin is further reinforced by the 18 swg doubling plate; this is more than adequate to cover any loss.

The loading conditions at former 22 considered in the calculation are those induced solely by the swept wing. Loads imposed under normal flying conditions have not been considered and will, of course, be additional. Sufficient detailed information for the consideration of these loads is not available in the "Lancaster" type record and, in view of this, it is suggested that a check on this particular component might be carried out by the parent company who, presumably, hold the detail figures.

d-3 <u>Forces at Cut-out due to Shear in the Skin</u>

Shear flow in skin = 114 lb/in (Section d-2, Sub-section iv, paragraph 1.)



The cut-out is effectively bounded by the fuselage frames and longerons, as shown above.

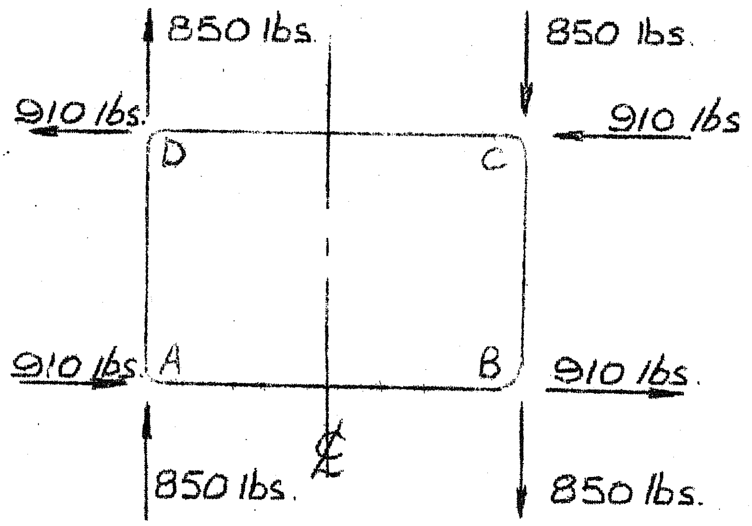
It is necessary to replace the skin removed at the cut-out by a balancing shear force system along the reinforcement of the cut-out edges.

Considering edge AD, the shear flow of 114 lb/in. produces a compression force at D and a tension force at A.

Forces at A and D $= \left(\frac{1}{2} \times 114 \times 15\right) = \underline{850}$ lbs

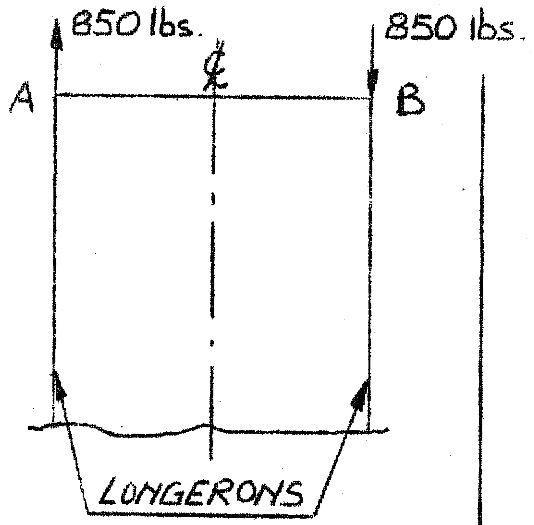
R.F.

Considering the remaining two edges the forces at all corners are as shown below:-



Considering edge AB. At corners A and B there are equal and opposite loads of 850 lbs. hence displacement midway between A and B, due to these loads, is zero.

The adjacent panel can therefore be regarded as fixed at the centre with a load of 850 lbs. applied at the corner of the longeron diffusing into it.



The above conditions are similar at all edges.

i) Stress in Longeron

The longeron is of channel section 3" x 0.75" x 18 swg. D.T.D. 610. Attachment is made via the upper flange and web only, therefore effective area =

$$(3" + 0.75")0.048 = \underline{0.18} \text{ in}^2$$

$$\therefore \text{ Maximum Stress} = \frac{650}{0.18} = \underline{3,600} \text{ lb/in}^2$$

Allowable stress for local instability

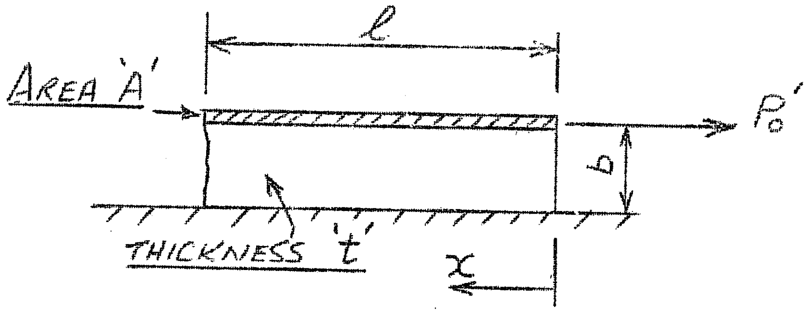
$$= 3.62 \left(\frac{.048}{3}\right)^2 E_t$$

$$= \underline{5,200 \text{ lb/in}^2}$$

1.44

R.F.

ii) Shear Stress in Diffusion Panel



By analysis, the edge load

$$P'' = P_o' \frac{\sinh k(\ell - x)}{\sinh k\ell}$$

$$(P'' = 0 \text{ when } x = \ell)$$

and the shear stress in the panel

$$q = -\frac{P_o' G}{kbAE} \frac{\cosh k(\ell - x)}{\sinh k\ell}$$

$$= -\frac{P_o' k \cosh k(\ell - x)}{t \sinh k\ell}$$

where $k = \sqrt{\frac{tG}{bAE}}$

Maximum shear stress occurs when $x = 0$

$$\text{i.e. } q_{max} = -\frac{P_o' k}{t \tanh k\ell}$$

For these calculations:-

$$t = 0.048'' + 0.028'' = 0.076''$$
$$b = 8.0''$$
$$A = 0.18 \text{ in}^2$$
$$G/_E = 0.30 \qquad \therefore k = 0.126 \text{ in}^{-1}$$

Assuming all the load in the longeron between fuselage bays is diffused to zero $\ell \doteq 16''$

$$\therefore k\ell = 2.01 \qquad \tanh k\ell = 0.96$$

$$\therefore q_{max} = \frac{850 \times 0.126}{0.076 \times 0.96} = \frac{112}{0.076}$$

$$= \underline{1470} \text{ lb/in}^2$$

and maximum shear flow $= q_{max} \cdot t = \underline{112}$ lb/in

Rivets attaching the longeron to the skin are 5/32" Dia. at 1" pitch. Allowable rivet strength = 520 lbs/rivet > 2

R.F.

Shear flow at cut-out corners

$= 112$ lbs/in (due to cut-out) $+ 114$ lb/in.

$= \underline{226}$ lb/in

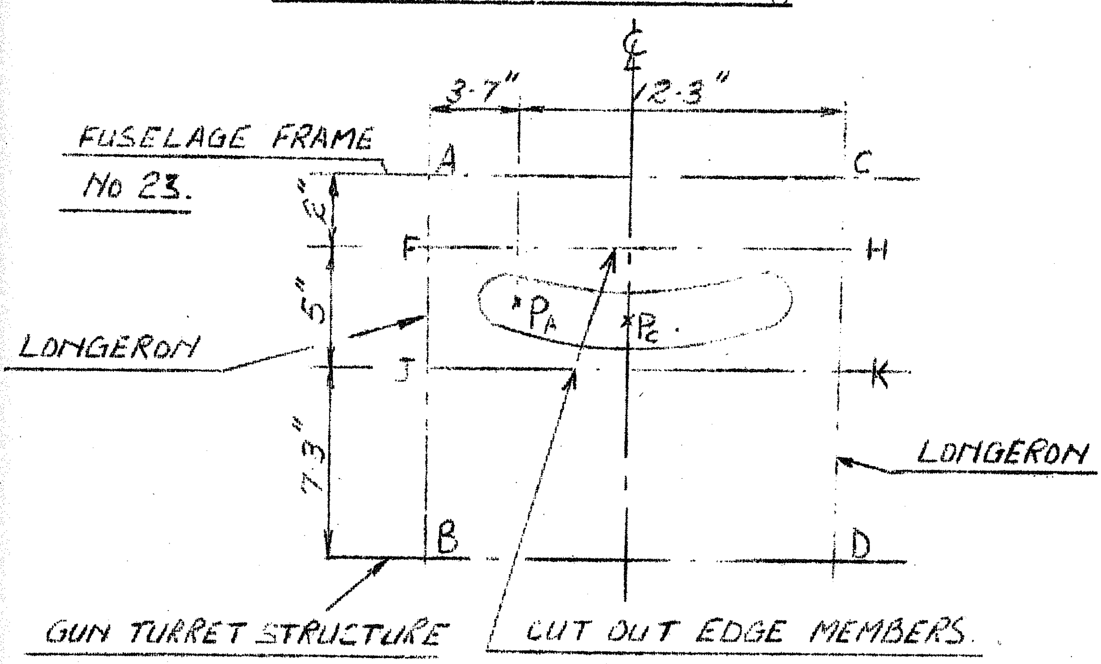
and shear stress $= \dfrac{226}{0.076}$

$= \underline{3000}$ lb/in^2

This stress is very small and is obviously O.K.

The diffusion of the 910 lb. load from the frames, treated similarly to above will also be small and is satisfactory.

d-4 <u>Stresses in Fuselage due to Maximum Download on the Rear Attachment Fitting.</u>



Maximum download on fitting = <u>2950</u> lbs.

(Loading Case 'C' - Section B, sub-section b-4)

Load application at point P_c (see above sketch)

Considering Case 'A' load application at

point P_A = 2950 - 530 = <u>2420</u> lbs

R.F.

The down load is assumed to be equally divided between the edge members of the cut-out, FH and JK, which are simply supported on the longerons at their ends. The reactions on the longeron are reacted by the fuselage frame at A, and the gun-turret structure at B, this also acting as a simply supported beam.

i) Cut-out Edge Members FH and JK

The critical case for these members is C, where 1475 lb. is applied at the centre of each member and reacted by the longerons at each end.

$$\text{Maximum BM at centre} = \frac{1475}{2} \times 8.0'' = \underline{5900} \text{ lb.in.}$$

Edge member FH is a box section 2" x 3.5" deep at the centre, 18 swg D.T.D. 610.

Moment of inertia at centre

$$I = \left(\frac{2}{12} \times 3.5^3 \times 0.048\right) + \left(2 \times 2 \times 0.048 \times 1.8^2\right)$$
$$= \underline{0.965} \text{ in}^4$$

$$\therefore \text{Max. Stress } f = \frac{My}{I} = \frac{5900 \times 1.75}{0.965} = \underline{10,700} \text{ lb/in}^2$$

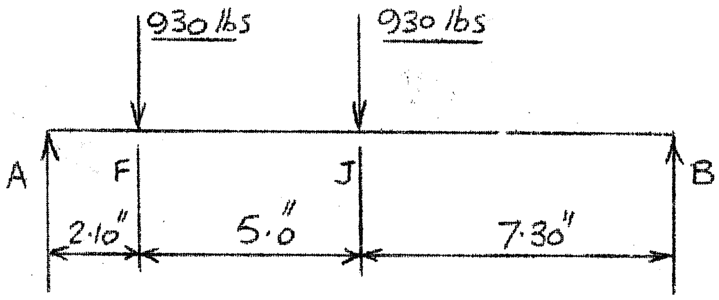
Allowable local instability stress

$$f_c = 3.62\left(\frac{0.048}{2}\right)^2 E_t = 0.002085\, E_t = \underline{19,000} \text{ lb/in}^2$$ 1.78

ii) Longerons AB and CD

The critical case for the longerons in bending is case A where the reaction at F and J on the longeron

$$= \left(\frac{2420}{2} \times \frac{12.3}{16}\right) = \underline{930} \text{ lbs}$$



$$\text{Reaction at B} = 930\left(\frac{2.10 + 7.10}{14.4}\right) = \underline{594} \text{ lb.}$$

and " at A $= (1860 - 594) = \underline{1266}$ lb.

$$\text{Maximum Bending Moment at J} = (594 \times 7.3) = \underline{4336} \text{ lb.in.}$$

R.F.

Assuming an effective skin width of 15t acting with the section.

Section centroid from C.G. of channel only:-

$$y = \frac{0.076 \times 1.538}{(0.076 + 0.228)} = \underline{0.384''}$$

and Moment of inertia of section ,

$$I = I_c + (0.228 \times 0.384^2) + 0.076(1.538 - 0.384)^2$$

where I_c = moment of inertia of channel about own centroid = $\underline{0.27}$ in^4

$$\therefore \quad I = 0.27 + 0.0336 + 0.1012 = \underline{0.4048} \ in^4$$

then Max. tensile stress $= \dfrac{4336 \times 1.19^2}{0.4048} = \underline{12,800}$ lb/in^2

Compressive Stress in channel $= \dfrac{4336 \times 1.116}{0.4048}$

$= \underline{12000}$ lb/in^2

Allowable local instability stress (flange)

$$= 0.58\left(\frac{0.048}{0.75}\right)^2 E_t$$

$$= 0.00238 \ E_t$$

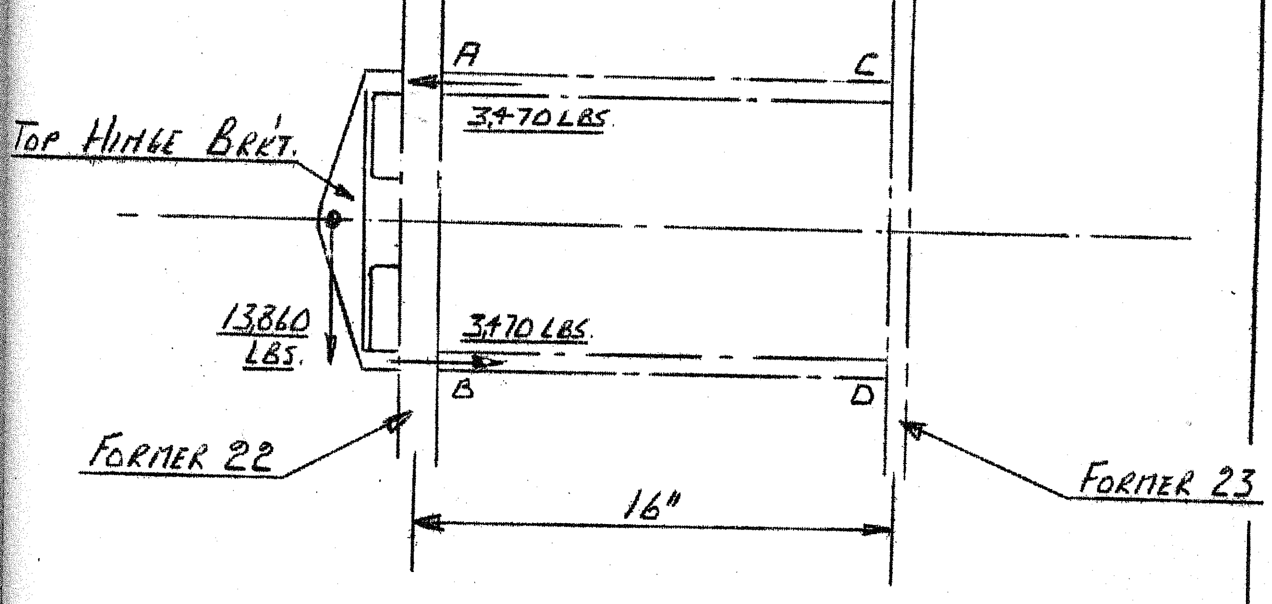
$= \underline{20000}$ lb/in^2 1.66

Critical stress for inter-rivet buckling, rivet pitch 1.0", flange thickness t = 0.048"

$= \underline{28500}$ lb/in^2 > 2

(R.Ae.S. 02.01.15)

d-5 Stresses in Fuselage Top Panel ABCD due to Upper Hinge Bracket Offset



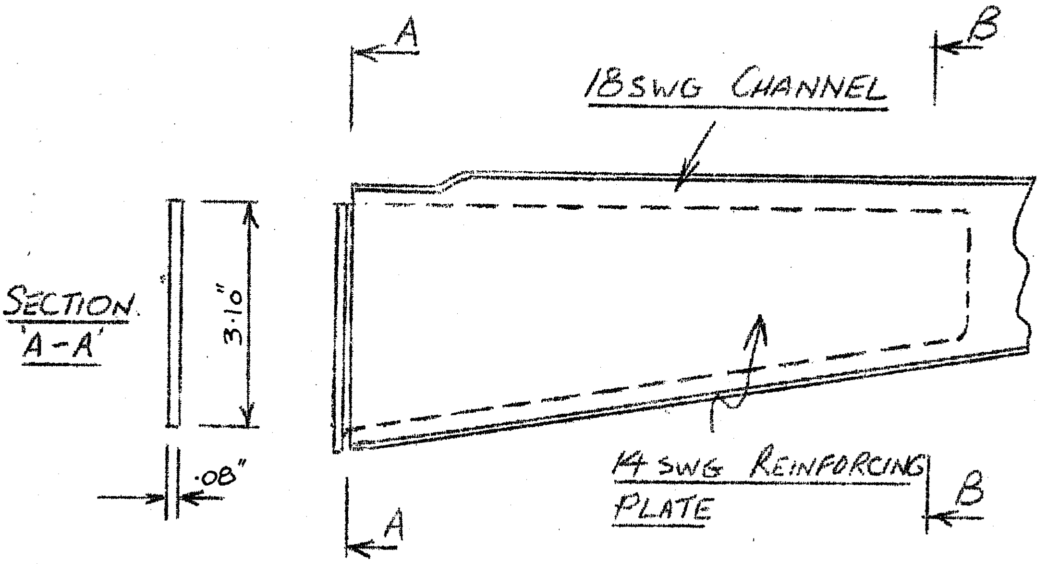
R.F.

Moment due to offset upper hinge load of 13,860 lbs, puts equal and opposite loads of 3470 lbs into members AC and BD. (See section D, sub-sect. d-2, para. i). These loads induce shearing loads in the panel.

Assuming uniform load distribution along AC and BD

$$\text{Shear flow} = \frac{3470}{16} = \underline{217} \text{ lb/in.}$$

i) Edge Members AC and BD



1) Section 'A-A'

$$\text{Load on section} = \underline{3470} \text{ lbs}$$

$$\text{Section Area} = 3.10'' \times 0.08'' = \underline{0.248} \text{ in}^2$$

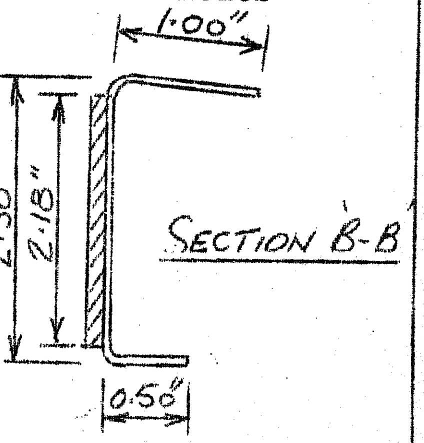
$$\therefore \text{Stress} = \frac{3470}{0.248} = \underline{14,000} \text{ lb/in}^2$$

Assuming simply supported section edges, the local instability stress

$$f_c = 3.62\left(\frac{0.08}{3.1}\right)^2 E_t$$

$$= \underline{20,000} \text{ lb/in}^2$$ 1.43

2) Section 'B-B' thro' the last rivet holes approx. $5\frac{1}{2}$" from Section 'A-A'



Then load on the section

$$= 3470 - (5.50 \times 217)$$

$$= \underline{2275} \text{ lbs}$$

Section Area =

$$(4'' \times 0.048'') + (2.18'' \times .08'')$$

$$= \underline{0.367} \text{ in}^2$$

$$\therefore \text{Stress} = \frac{2275}{0.367} = \underline{6200 \text{ lb/in}^2}$$ > 2

R.F.

ii) Riveting at Panel Edge

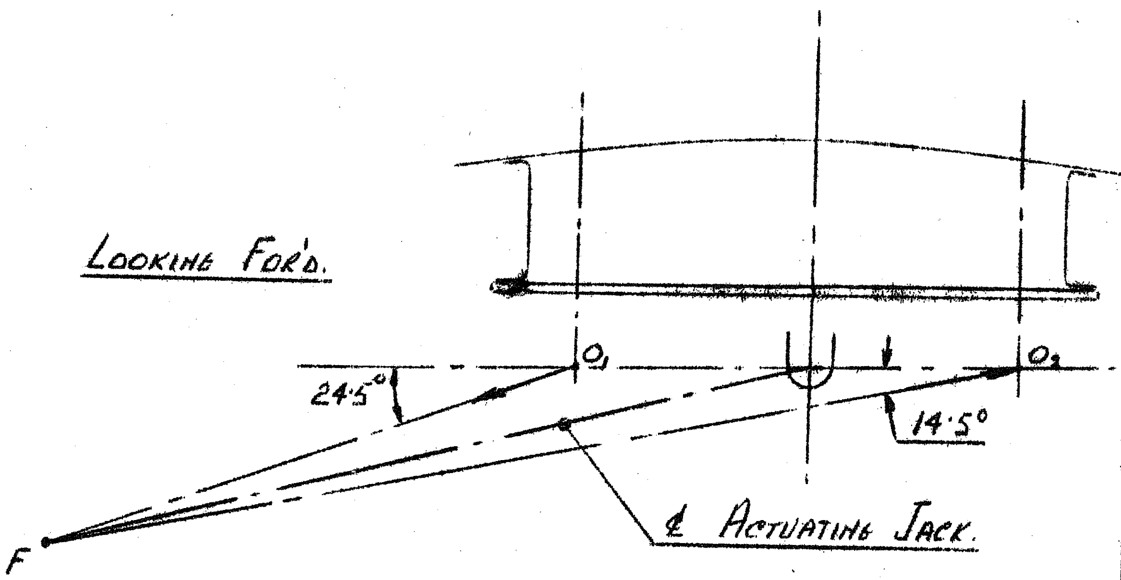
Shear flow = 217 lb/in

Rivets are 5/32" dia L.69 snaphead
Rivet pitches:-

Along AC and BC	....	0.63"
On former 22	....	0.50"
On former 23	....	1.00"

∴ Min. rivet strength/in = 560 lbs > 2

d-6 Actuating Jack and Forward Attachment



Maximum side load jack must withstand = 2310 lbs (at positions O_1 and O_2 on above sketch.)

$$\therefore \text{Max. tensile load on jack} = \frac{2310}{\text{Cos } 24.5^\circ}$$

$$= \underline{2540} \text{ lbs. } (O_1F)$$

and the maximum compressive load on jack $= \dfrac{2310}{\text{Cos } 14.5^\circ}$

$$= \underline{2390} \text{ lbs. } (O_2F)$$

(the slight angle in the plan view is neglected).

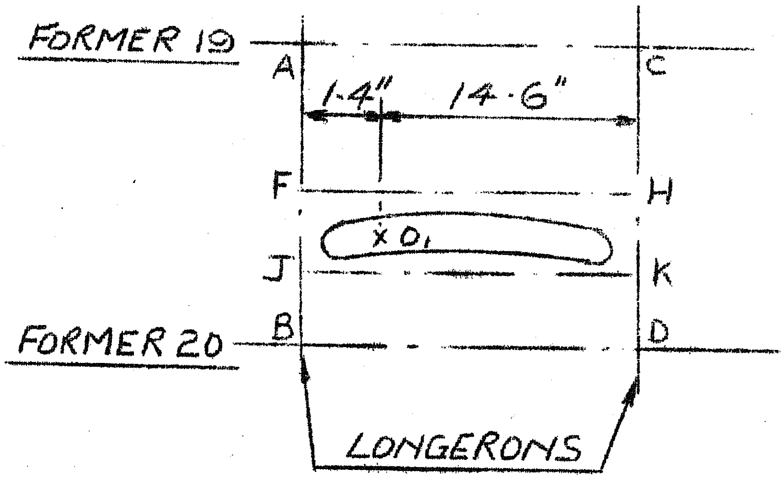
Hence the maximum vertical load on the cut-out structure due to the jack $= 2540 \times \text{Sin } 24.5^\circ$

$$= \underline{1095} \text{ lbs. (downwards)}$$

or $2390 \text{ Sin } 24.5^\circ = \underline{992}$ lbs (upwards)

R.F.

i) Structure around Cut-out under Vertical Load



Maximum vertical load on the cut-out = 1095 lb (due to jack)

+ 530 lbs (Case B)

= 1625 lbs acting downward at O_1

Making the same assumptions as in Section D, Sub.Sect.d-4, the edge members FH and JK are subjected to a smaller load than at the rear cutout, and are therefore satisfactory.

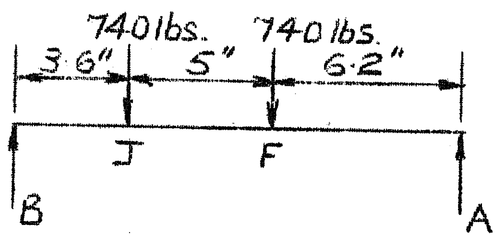
Considering longeron AB. The load at F and J due to the vertical load 1625 lbs

$$=\left(\frac{1625}{2} \times \frac{14.6}{16}\right) = \underline{740} \text{ lbs}$$

$\therefore$ Reaction at 'A'

$$= 740\left(\frac{3.60 + 8.60}{14.8}\right)$$

$$= \underline{610} \text{ lbs}$$



and Reaction at 'B'

$$=\left(2 \times 740\right) - 610$$

$$= \underline{870} \text{ lbs}$$

Max. B.M. at 'F' = 3780 lb.in.

This B.M. is smaller than that at the rear cut-out, and as the longeron section is identical it is obviously O.K.

R.F.

ii) <u>Actuating Jack and Attachments</u>

Max. load in the jack = <u>2540</u> lbs (tension)

and <u>2390</u> lbs (comp)

Assuming a jack section equal to that of the screw extension, and a max. length of 33"

P_{cr} (strut) = <u>6110</u> lbs > 2

The jack is obviously O.K. in tension.

By inspection, the wing and fuselage attachments are quite adequate to cater for the applied loads.

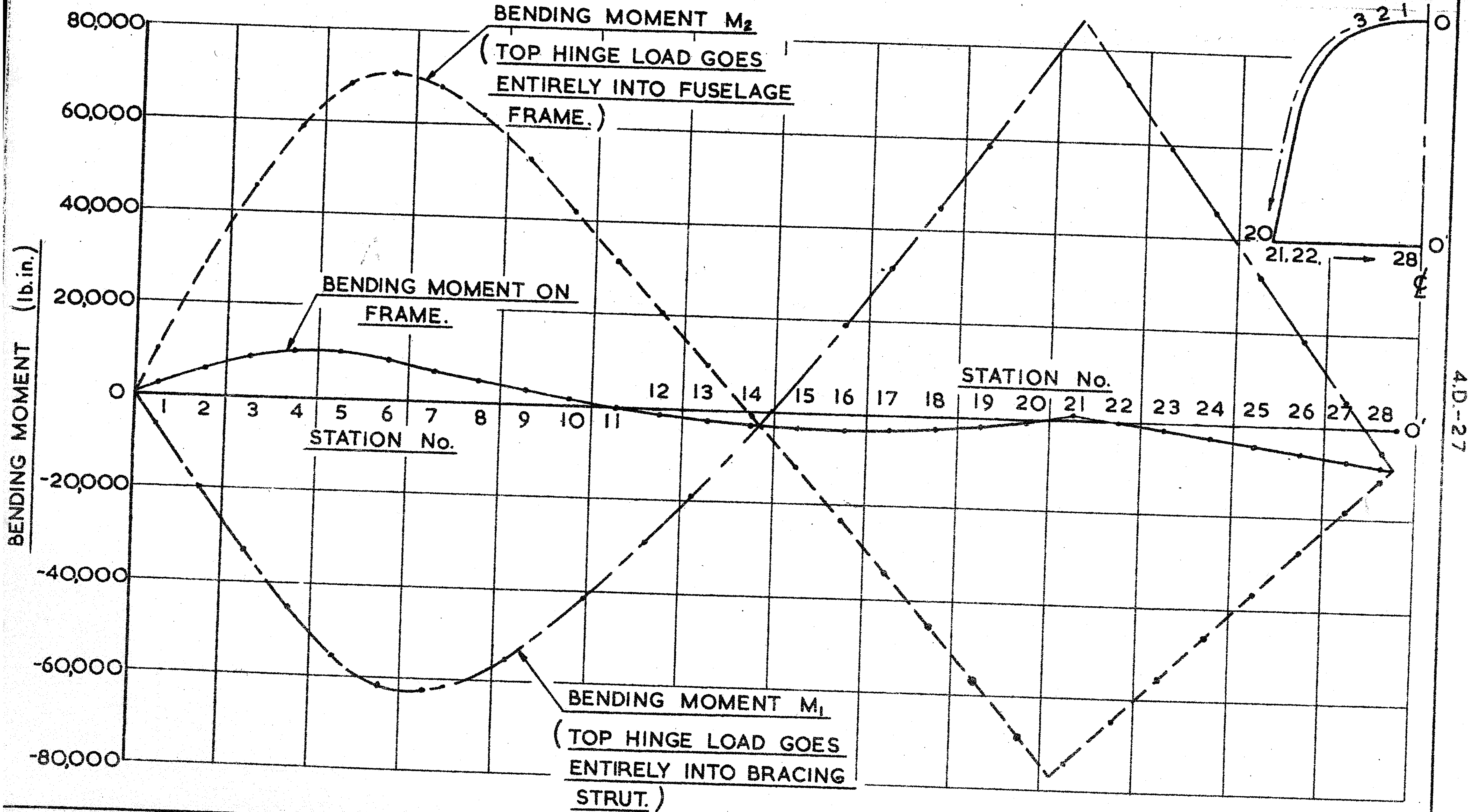

80,000
60,000
40,000
20,000
0
-20,000
-40,000
-60,000
-80,000
BENDING MOMENT (lb.in.)
BENDING MOMENT M_2
(TOP HINGE LOAD GOES ENTIRELY INTO FUSELAGE FRAME.)
BENDING MOMENT ON FRAME.
BENDING MOMENT M_1
(TOP HINGE LOAD GOES ENTIRELY INTO BRACING STRUT.)
STATION No.
1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18 19 20 21 22 23 24 25 26 27 28
3 2 1
O
20
21,22. → 28
O'
℄

FIG. 5.

AERODYNAMIC ANALYSIS

Flight Department Memo DF/RES/7.

C.P. Position of wing based on full half wing area, i.e. including tip and area below boundary layer fence, is as follows :-

Spanwise = $0.432\frac{b}{2}$ = 44.25" from root

Chordwise = $0.27\ \bar{c}$ = 23.25" from L.E.

Aerodynamic characteristics of wing to be fitted to Lancaster PA.474 :

1) Derivation of lift curve slope using R.Ae.S. data sheets.

From Data Sheet Wings 01.01.01 the equivalent angle of sweepback is

$$\Lambda_e = 38^\circ\ 18'$$

From Data Sheet Wings 01.01.05, for a 15% thick section at a R.N. of 25×10^6 with a trailing edge angle of 20° and transition assumed to occur at 0.1_c the lift curve slope in two dimensional incompressible flow is :

$$a_o = 5.52 \text{ per radian}$$

The compressibility correction is negligible (one per cent), so that, (in the notation of R.Ae.S. Data Sheet Wings 01.01.01) we have

$$(a_o)M \triangleq a_o = 5.52 \text{ per radian}$$

The lift curve slope of the wing with sweep = 45°, taper ratio λ = 1 and aspect ratio A = 2.382 is:

$$a_1 = 2.237$$

2) Stability Derivatives

C_L	Without Test Wing		With Test Wing		% Increase	
	n_v	b_r	n_v	b_r	n_v	b_r
0	0.039	-0.120	0.061	-0.129	57	7.5
0.5	0.044	-0.115	0.066	-0.123	50	7
1.0	0.059	-0.109	0.081	-0.115	37	5.5
1.5	0.084	-0.104	0.106	-0.109	26	5

As stated in R.A.E. Tech. Memo Aero 101 this comparative increase in the value of n_v should improve the handling characteristics which were somewhat marginal at low speeds.

A. BONDED METAL TO PLYWOOD SHEAR PANEL TEST

(Report No. T.S.M. 146)

B. COMPRESSION TESTS ON STRUT WITH SWAGED ENDS

(Report No. T.S.M. 145)

Department of Aircraft Design,
The College of Aeronautics,
Cranfield.

November 1955.

Report No. T.S.M. 146

BONDED METAL TO PLYWOOD SHEAR PANEL TEST FOR "LANCASTER" PA 474 SWEPT WING STRUCTURE

by

D. HOWE, D.C.Ae., A.F.R.Ae.S.

BONDED METAL TO PLYWOOD SHEAR PANEL TEST

A

25th November, 1955

Report No. TSM 146

Introduction

The test shear panel is shown in Fig. 1. It is representative of a section of the main spar of the swept wing structure to be fitted to "Lancaster" PA 474. The shear web consists of 18G light alloy with a ply covering on either side. The metal only is attached by $\frac{1}{8}$" dia. snaphead rivets to the top and bottom boom members.

Shear Panel Test

The specimen was loaded in diagonal compression, shear strain being measured by electrical resistance strain gauges.

The variation of shear strain with load is shown in Fig. 2. From this the mean shear strain at buckling is seen to be 2.16×10^{-3}. The corresponding load was 3.3 tons. Final failure was due to buckling and occurred at a load of 6.75 tons.

(a) Buckling Conditions

Using determined Shear Modulus of 4.23×10^{6} lb/sq.in. for a strain of 2.16×10^{-3} (see Appendix).

Effective metal stress at buckling = 9,150 lb/sq.in.

Using $E = 11.2 \times 10^{6}$ lb/sq.in., the corresponding buckling coefficient, K, based on the panel between the stringers is 6.7.

Assuming all the applied load is reacted by the web, the theoretical load to give a stress of 9,150 lb/sq.in. is 5,600 lb.

Actual load at buckling = 7,400 lb.

Therefore percentage load carried by shear panel = 75.6

(b) Failing Stress

Assuming this percentage to hold at failure, actual load carried by panel = 11,450 lb.

Corresponding Shear Stress = 18,500 lb/sq.in.

Results

Initial Buckling Shear Stress	9,150 lb/sq.in.
Failing Shear Stress	18,500 lb/sq.in.

Signed: D. HOWE

Date: 28th November, 1955.

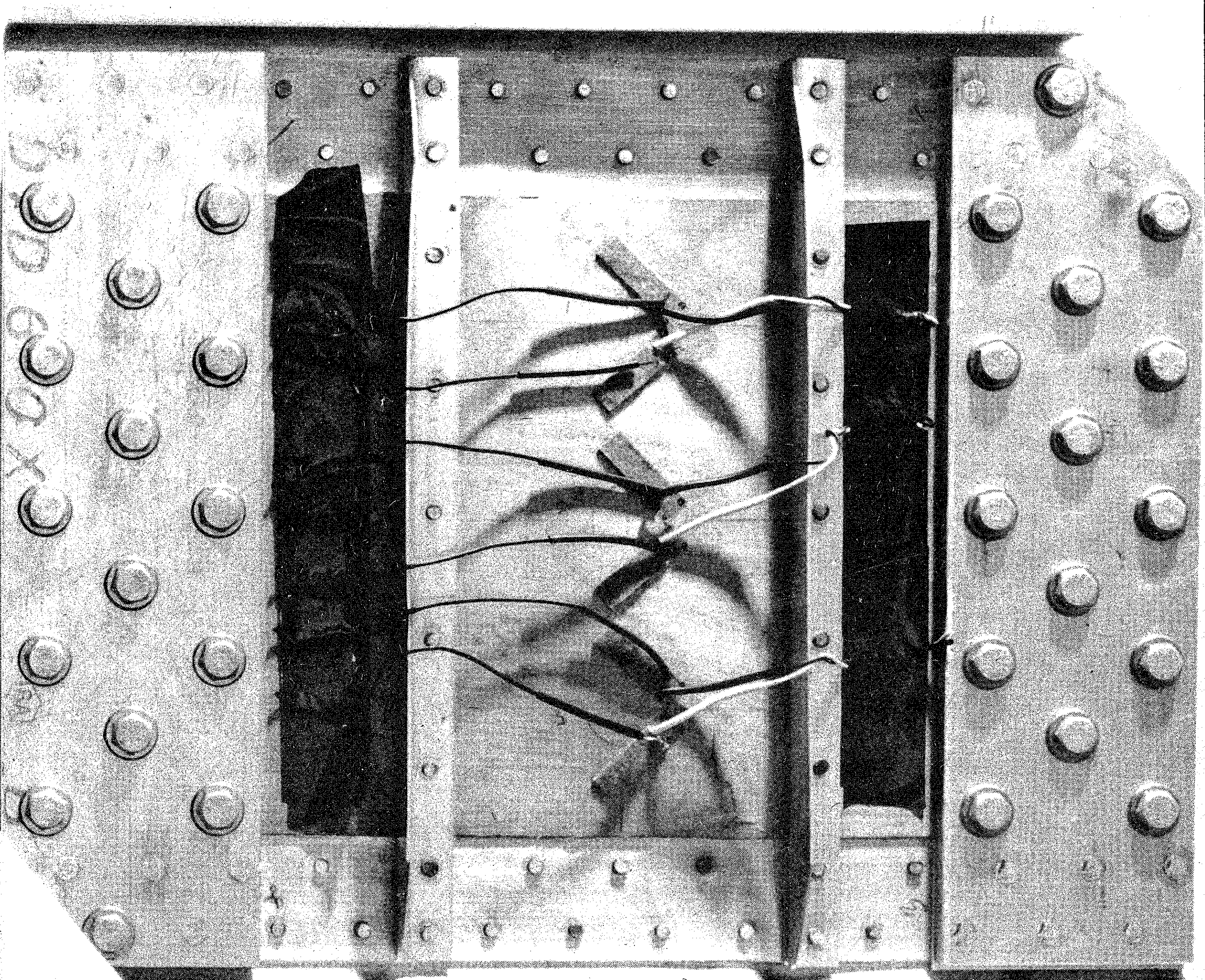


FIG.1.

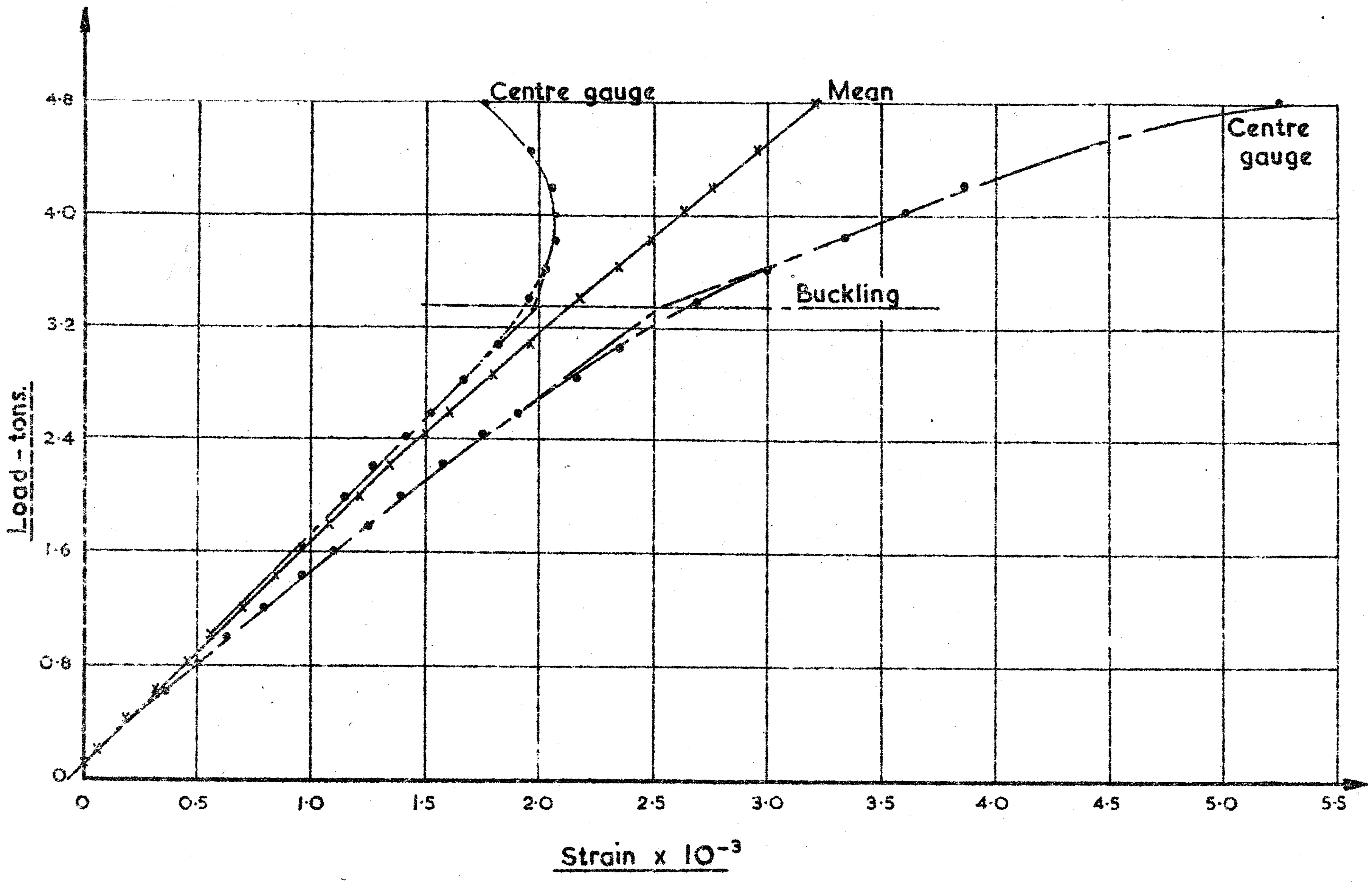


FIG. 2. PANEL. LOAD-STRAIN CURVES.

APPENDIX

CONTROL TESTS

Tensile control tests were used to obtain the strain gauge calibration factor, Young's Modulus and Poisson's Ratio for the composite wood-metal material of the shear panel.

The gauge factor and Young's Modulus, E, were determined by measuring the strain on a 0.5" wide specimen using electrical and mechanical instruments. The gauge length was 2.0 in. Poisson's Ratio, σ, needed to calculate the Shear Modulus, G, was estimated by measuring the 45° diagonal strain on a 1.5" wide specimen.

The stress-strain curve had a linear portion consisting of three definite gradients and the following results were obtained.

Strain	E lb/sq.in	σ	G lb/sq.in.
0 - 0.00015	12.9×10^6	0.317	4.90×10^6
0.00015 - 0.00023	11.2×10^6	0.325	4.23×10^6
0.00028 - 0.00040	9.62×10^6	0.335	3.61×10^6

The moduli are "effective" values based on metal thickness. i.e. the metal is assumed to carry all the load.

Metal Thickness = 0.05"   Total Thickness = 0.138"

Strain Gauge Factor = 1.90 (Nominal value 2.17)

Department of Aircraft Design
The College of Aeronautics
Cranfield

October 1955.

Report No. T.S.M. 145

COMPRESSION TEST ON SUPPORT STRUT FOR LANCASTER PA 474 SWEPT WING STRUCTURE

by

C. K. TROTMAN, B.Sc. (Eng.), A.F.R.Ae.S.

B.

COMPRESSION TEST ON SUPPORT STRUT (Report No. T.S.M. 145)

(Drg. No. M-167-2)

Introduction

The support strut (Drg. No. M-167-2) for the swept wing structure being fitted to Lancaster PA.474 consists of a nominal 2.5/8" o/d x 17 s.w.g. T45 tube with swaged ends incorporating $2\frac{1}{2}$" o/d x 17 s.w.g. T45 local liners, and is 70.23" in overall length. Due to lack of information on the probable behaviour of the swaged ends it was decided to test one to destruction. The strut chosen was one which had been rejected for use due to the eccentricity of the ends exceeding the prescribed tolerance.

The test was carried out on 25th February 1955.

Test Specimen

The actual dimensions of the test specimen are as follows :-

	Mean outside diameter	= 2.626"
	Mean thickness	= 0.059"
∴	Average cross-sectional area	= 0.476 sq.ins.
	Eccentricity of centre-line of swaged ends relative to tube centre-line	= + 0.128" and -0.070"

A slight amount of ovality was measured towards the ends of the circular portion of the strut, the maximum difference in two diameters at right angles to each other being 0.017".

The maximum difference in wall thickness across any one of the sections measured was 0.007".

Test Arrangement

The swaged ends were bolted to the free legs of stiffened tee-section end plates as shown in Fig.4. This simulated the method of attachment to be used on the aircraft, but no attempt was made to represent the stiffness of the attachment points. The specimen was then loaded in the 150 ton Denison Compression Testing Machine.

Tinsley Type 6K electrical resistance strain gauges were cemented to the swaged ends as indicated in Fig. 1. The positions were chosen in an attempt to measure the maximum strains at points of stress concentration due to diffusion of load from the swages into the tube proper. Dial gauges were used to measure the average strain in the central portion of the strut (gauges A and B) and the amount of bowing (gauges C and D) at the centre. Their dispositions are given in Fig. 1.

Test Results

Table 1 gives the readings taken during loading, and also shows the readings obtained on unloading from 7.5 tons. The strain gauge readings are plotted in Fig. 2 and those of the dial gauges in Fig. 3.

The strut failed due to instability of the swage at the upper end at 10.5 tons.

From the plot of $\frac{A + B}{2}$ the mean contraction on the middle portion of the tube over 21.4" gauge length, the modulus of elasticity of the material in compression is calculated to be 13,400 tons/sq.ins.

The maximum percentage electrical strain recorded was for gauge 5, which gave a reading of 0.502 at 10 tons ≡ 0.535 at 10.5 tons by extrapolation. With a gauge factor of 2.18 and assuming the material still behaves elastically the stress at this point at failure is thus 32.85 tons/sq.inch, compared with the average stress at the centre of the strut at failure of 22.1 tons/sq.inch.

The Southwell plot in Fig. 3b of δ v. δ/P (where $\delta = \sqrt{C^2 + D^2}$ = lateral bowing, and P = applied load) indicates that the strut would have failed flexurally at about 13.23 tons had there been no instability at the ends.

Fig. 5 shows the top of the strut after failure.

At the bottom of the strut, appreciable distortion was evident from about 8.5 tons load at the flare out on the centre-line of the swage. This is shown in Fig. 6.

Conclusions

The strut failed at a load of 10.5 tons due to instability of the upper swaged end. The stress distribution at the ends is complex and the local strains measured do not vary linearly with load, thus indicating a continuous redistribution of stress.

Unloading at 7.5 tons showed no appreciable permanent set.

Signed : C. K. TROTMAN

Date : 21st October 1955.

TABLE 1

LOAD (Tons)	Strain Gauges - % Electrical Strain												Dial Gauges - 10^{-3} ins.			
	1	2	3	4	5	6	7	8	9	10	11	12	A	B	C	D
1	.032	.032	.035	.028	.040	.032	.028	.025	.028	.020	.020	.020	3.5	4.5	-2	2.5
2	.065	.060	.068	.055	.080	.060	.050	.040	.055	.045	.038	.042	7	7	-2.5	6
3	.105	.090	.098	.082	.122	.092	.082	.070	.080	.062	.055	.068	11	9	-3	11
4	.140	.122	.132	.108	.168	.122	.110	Gauge Slipping	.112	.088	.085	.092	15.5	11.5	-5	17
5	.178	.155	.175	.138	.218	.155	.138		.142	.112	.105	.112	20	13.5	-8	25
6	.215	.188	.210	.175	.270	.185	.168		.175	.138	.132	.135	25	15	-10	35
7	.252	.228	.252	.205	.320	.222	.202		.210	.158	.155	.155	30	16.5	-11	47
7.5	.272	.252	.272	.225	.352	.242	228		.230	.175	.165	.165	33	17	-11.5	56
0	.01	.01	.008	.005	.012	.006	.012		.006	.006	.008	.008	0	0	-2	0
7.5	.272	.252	.275	.225	.352	.245	.230		.235	.175	.165	.168	33	17	-12	56
8	.288	.275	.295	.240	.372	.260	.250		.255	.188	.180	.178	36	17	-12	65
9	.328	.328	.335	.275	.438	.305	.312		.312	.210	.200	.195	43.5	17	-13	91
10	.355	.375	.368	.312	.502	.365	.418		.442	.232	.220	.210	53	14	-13.5	136

Strain Gauges - Tinsley Type 6K, Gauge Factor = 2.18

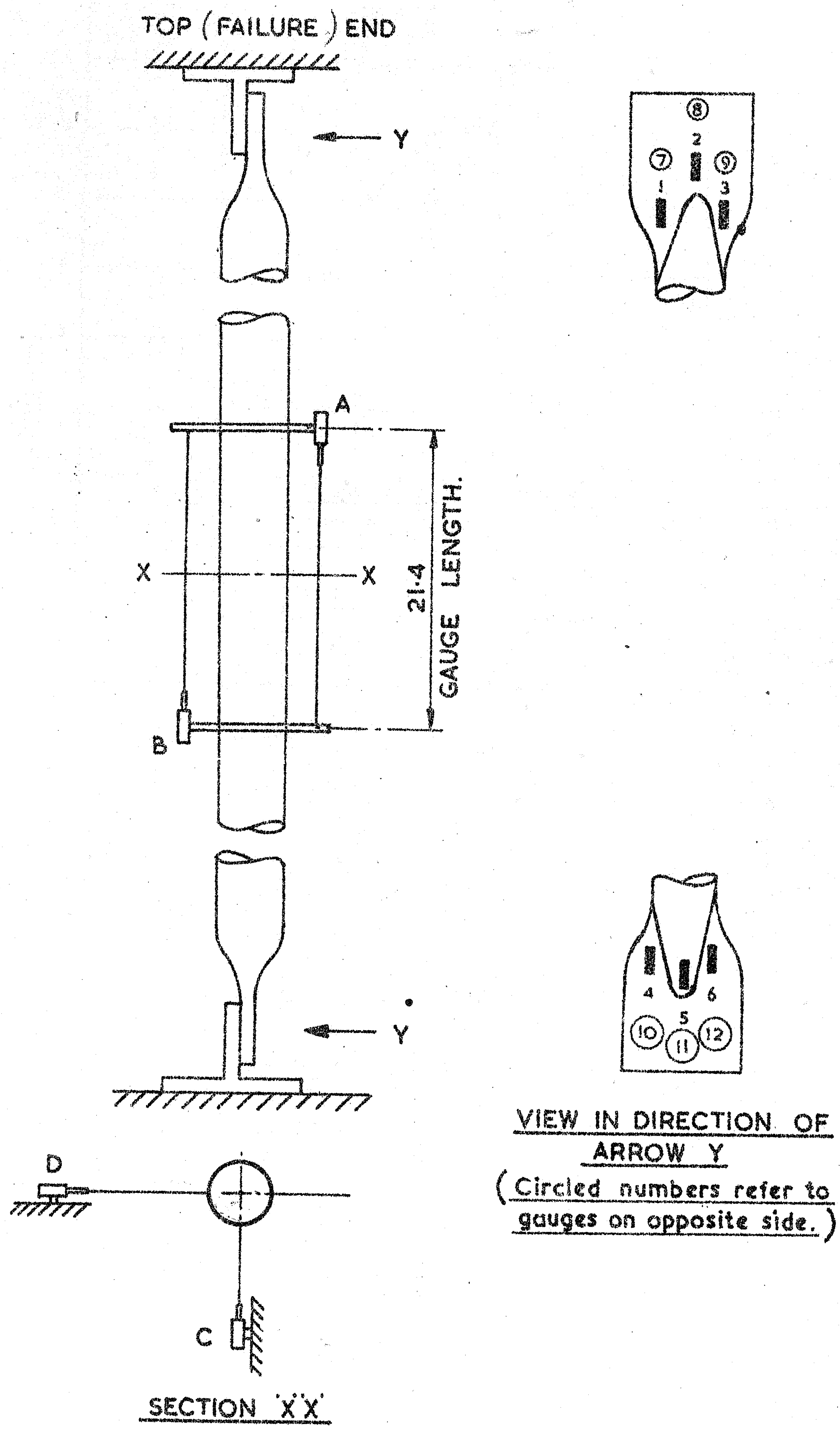


FIG. I. – DIAL GAUGE & STRAIN GAUGE POSITIONS.

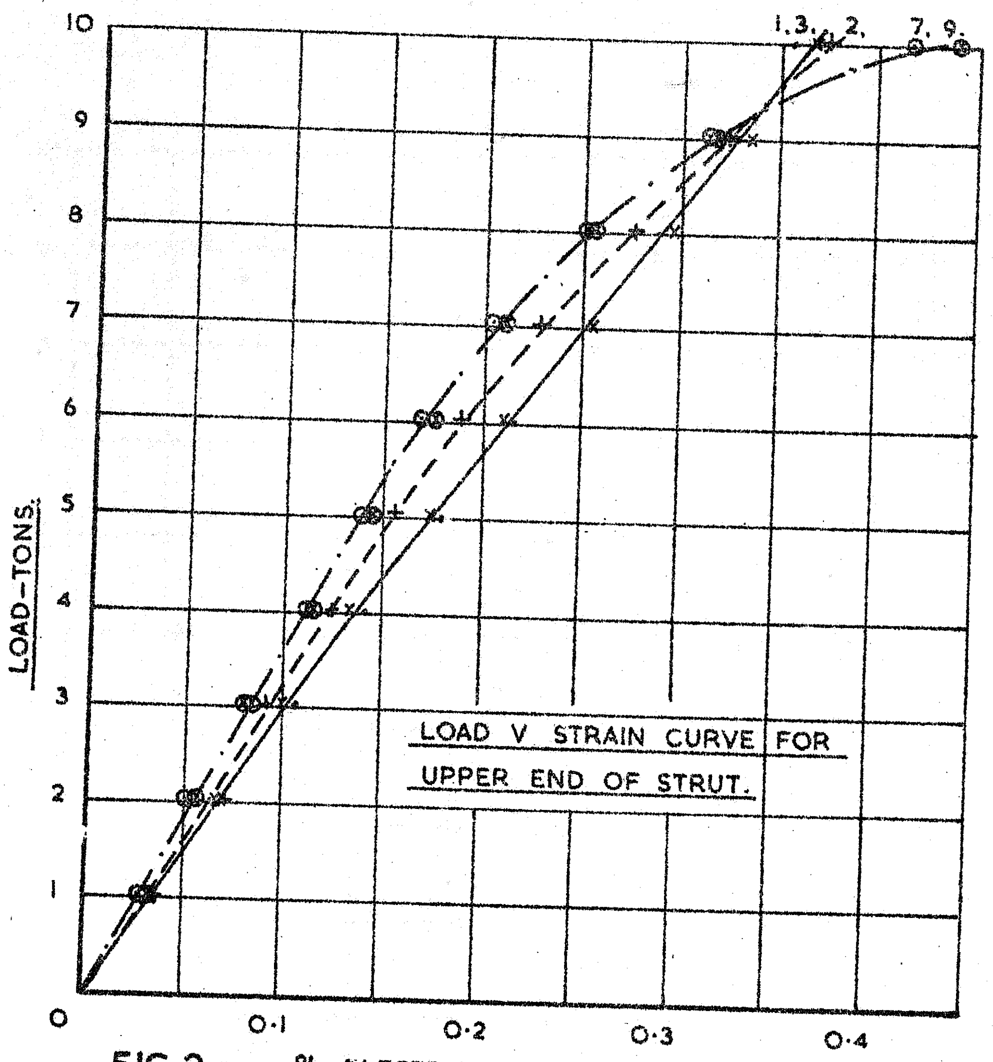


FIG. 2.a. % ELECTRICAL STRAIN

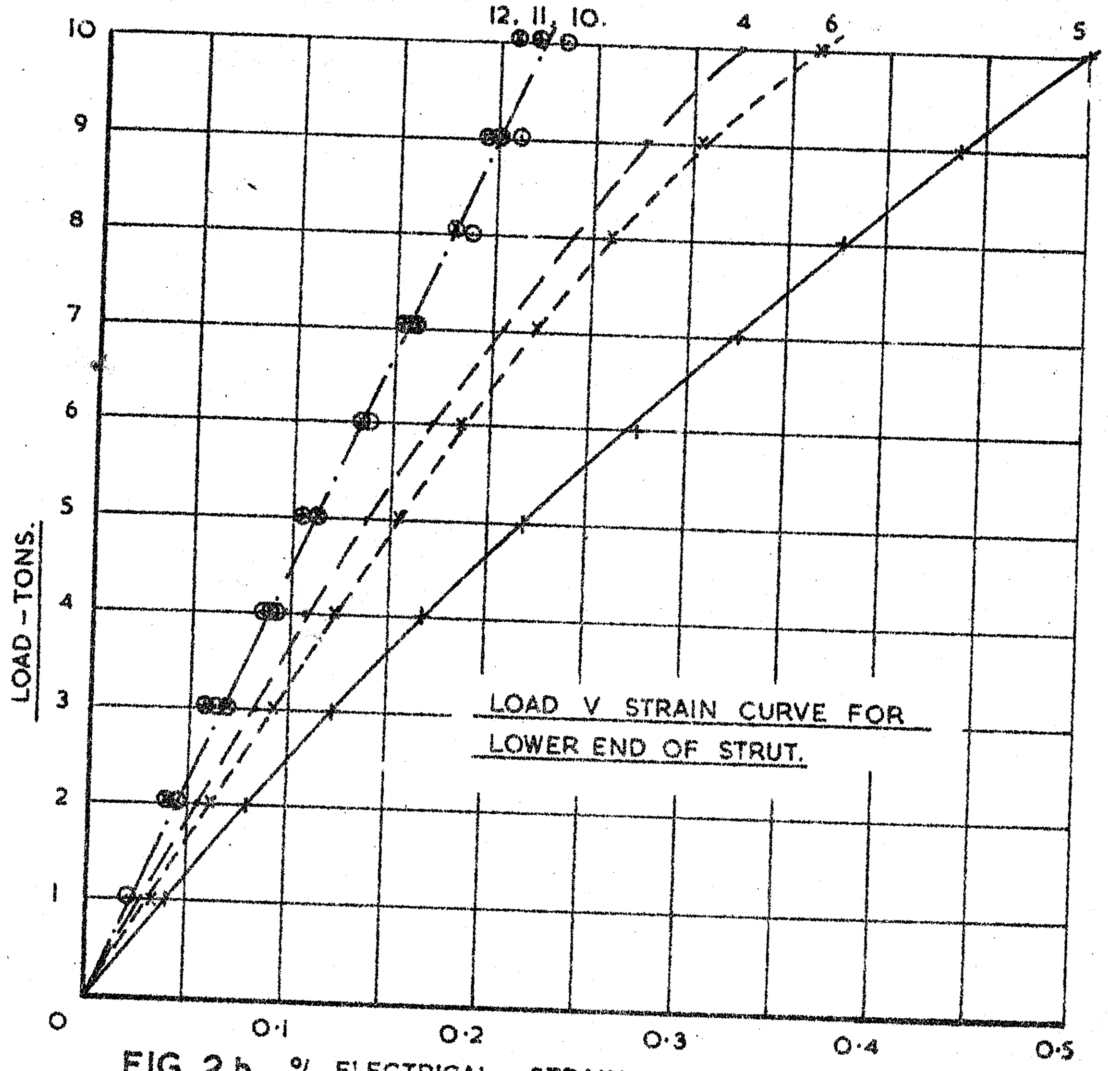


FIG. 2.b. % ELECTRICAL STRAIN

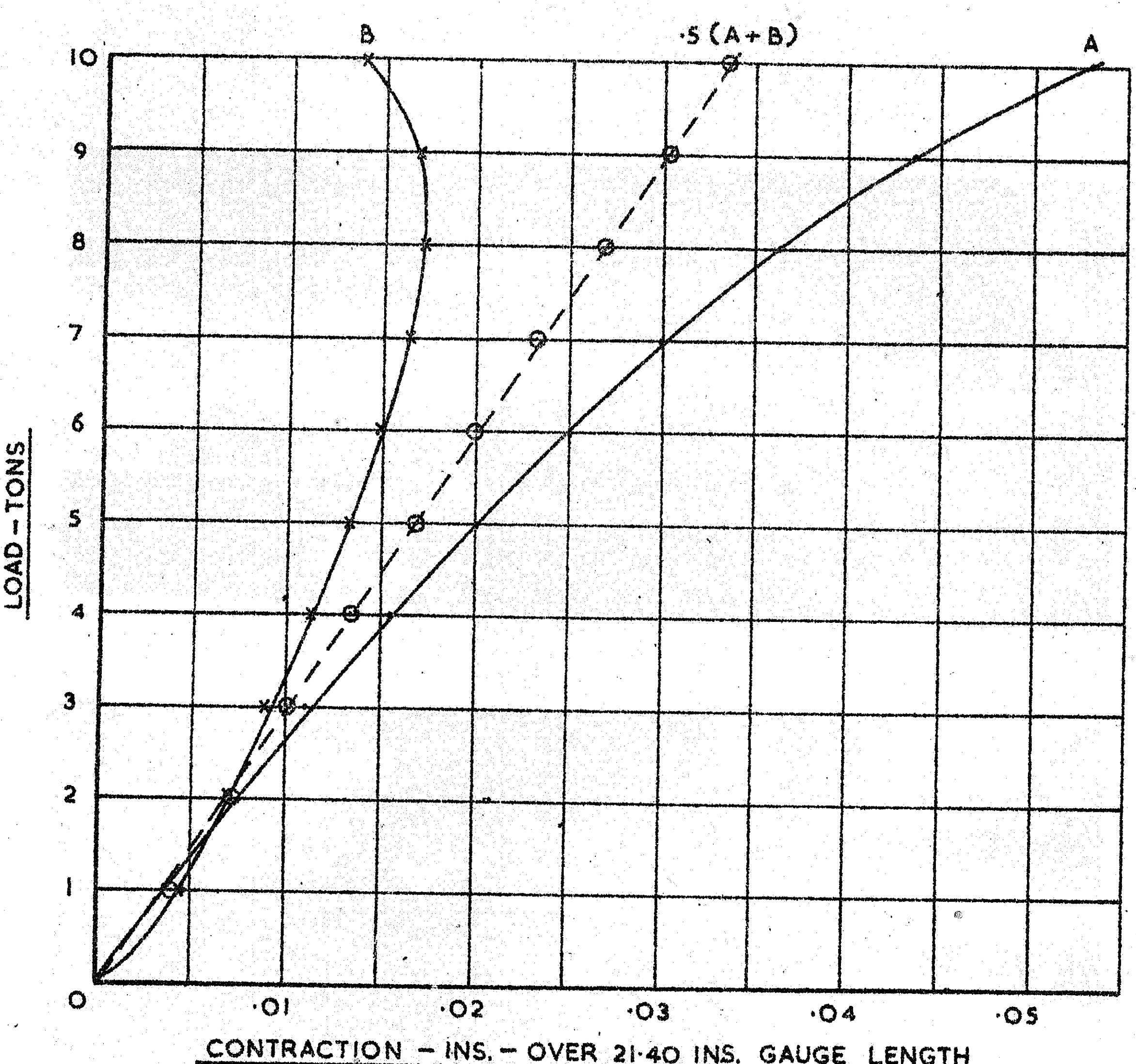


FIG. 3a.

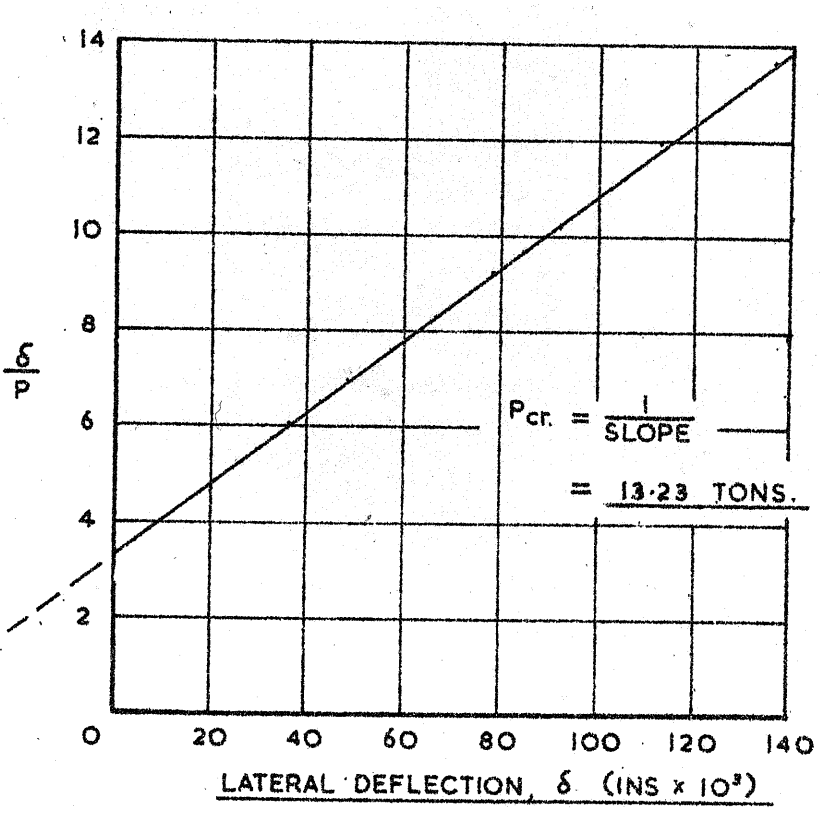


FIG. 3.b. SOUTHWELL PLOT.

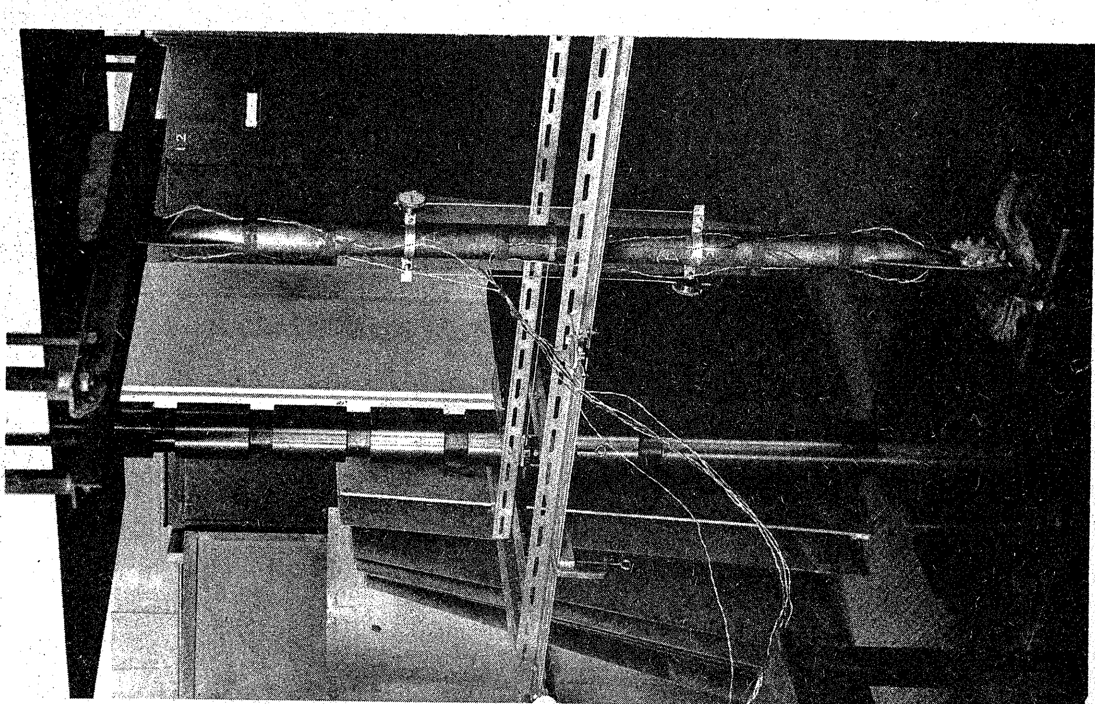


FIG. 4.

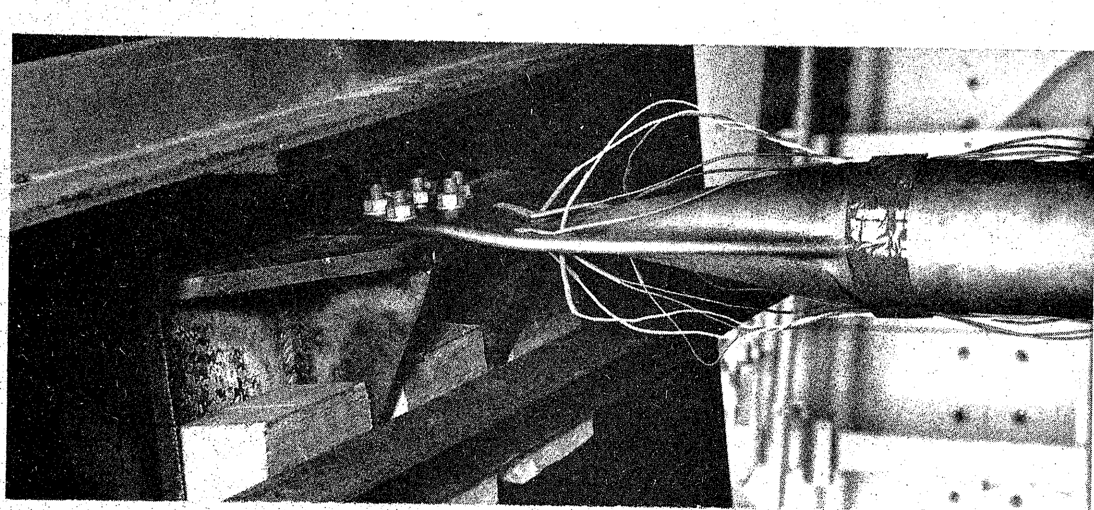


FIG. 5.

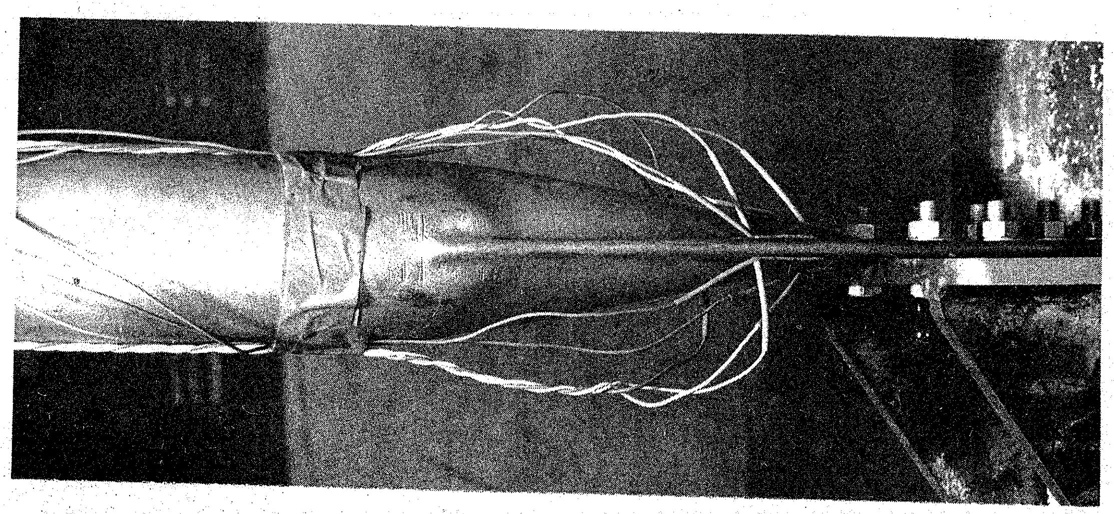


FIG. 6.

WEIGHT AND C.G. DATA

Aircraft Weighed Weight = 47,651 lbs (A/C Weight Form) (Ref. CAI/202)

Weight check carried out with aircraft in flying trim with regard to fuel and oil and including the manometer installation.

C.G. at above condition then falls 48.02" aft of datum.

Additional Items:-

Unit	Weight lbs.	Arms ins.	Moment lb. in.
Swept Wing	409	244.41	99963.69
Wing Spar Extension	36	205.41	7394.76
Boundary Layer Fence	40	208.41	8336.40
Swept Wing Yawmeter	20	281.02	5620.40
Actuating Jack	13	169.51	2023.60

$\Sigma_W = 518$ lbs. $\Sigma_M = 123{,}338.85$ lb.ins.

$$(47{,}651 + 518)x = 123{,}338.85 \text{ lb.in.}$$

$$\therefore \quad x = \underline{2.56''}$$

Aircraft A.U.W. in flying trim with swept wing and attachments installed = 48169 lbs.

C.G. then falls 50.58" aft of datum and is well within limits. (See Drg. No. M-170-1).

Standard crew will comprise pilot, flight engineer and two flight test observers, but provision will be made to carry three flight test observers.

Assuming, then, a crew of five:-

Additional Weight = 1000 lbs.

$\therefore$ Standard aircraft A.U.W. = 49169 lbs.

This is within the maximum aircraft A.U.W. of 50,000 lbs. and is therefore satisfactory.