

QcX
Avro
CF105
R-7-0562-9
ANALYZED



TECHNICAL REPORT

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT:

C-105

REPORT NO. _____

SHEET NO. 01

PREPARED BY

DATE

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CONFIDENTIAL

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STAG 22 STP. 87	

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INTRODUCTION

THE PRINCIPAL FUNCTION OF THE FIN BOX STRUCTURE IS TO TRANSFER LOADS FROM THE FIN TO THE WING STRUCTURE.

THE FIN BOX IS A FABRICATED STRUCTURE MADE PRINCIPALLY OF 75S-T6 AL ALLOY. THE BASIC COMPONENTS THAT MAKE UP THE BOX ARE THE VERTICAL PLATES, HORIZONTAL PLATES, DIAPHRAGMS, AND LONGITUDINAL ANGLES.

VERTICAL COUPLE LOADS FROM THE FIN SKIN ARE CARRIED BY THE VERTICAL PLATES TO A SERIES OF DIAPHRAGMS SPACED AT VARYING INTERVALS ALONG THE BOX. THE DIAPHRAGMS THEN TRANSFER THE LOADS TO THE WING SKIN VIA THE HORIZONTAL PLATES. TORSIONAL SHEAR FROM THE FIN IS TRANSFERRED TO THE UPPER HORIZONTAL PLATES THROUGH THE LONGITUDINAL ANGLES ATTACHING TO THE PLATES & THE FIN SKIN.

FIN LOADS USED IN THE ANALYSIS ARE TAKEN FROM THE LOAD CURVE ON PG. 1.03. THE CURVE HAS BEEN MODIFIED TO ACCOUNT FOR THE REDISTRIBUTION OF LOAD IN THE REGION OF THE ELEVATOR JACK PANEL, SINCE THE PANEL IS NOT DESIGNED TO REACT END LOAD FROM THE HORIZ. PLATES.

THIS REPORT IS AN ANALYTICAL SUBSTANTIATION OF THE FIN BOX STRUCTURE & ATTACHMENTS.

A. V. ROE CANADA LIMITED
MALTON, ONTARIO

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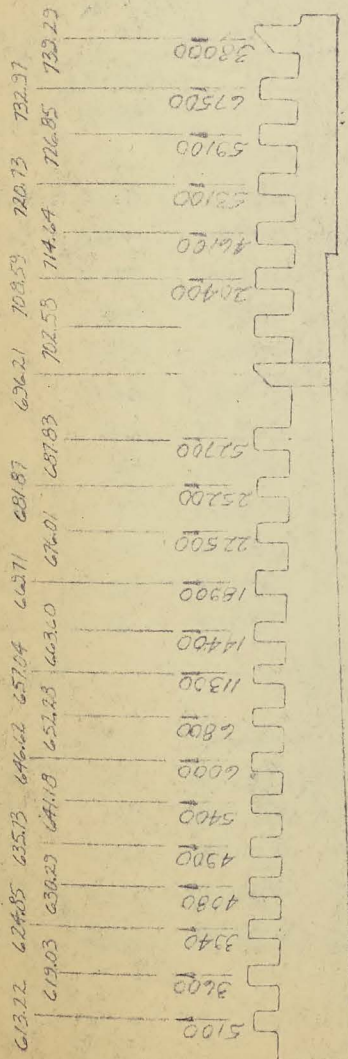
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FIN BOX VERTICAL PLATES
FINGER LOADS - RPD CASE

REF. PG. 1.05 FOR LOADS



LOADS SHOWN ARE
FOR SIDE, PUMPING
TENSION AND COMPRESSION.

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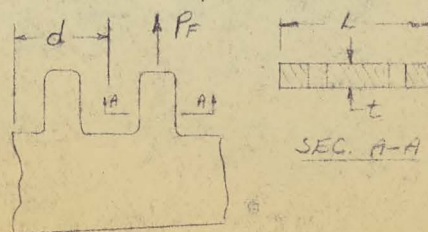
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FIN BOX VERTICAL PLATES

FINGER LOADS & STRESSES

1	2	3	4	5	6	7	8
STA.	*P = FIN SKIN END LOAD	d	P = P · d	t	L	A (COMP) = t · L	A (TENS)
	REF. PG. 103	QUICK NO. 7-0162-92			DRAWING NO. 7-0162-68		
613.22	600	8.49	5100	.114	23	.262	.177
619.03	620	5.81	3600	.130	23	.299	.202
624.85	700	5.62	3940	.144	1.5	.216	.216
630.29	750	5.44	4080	.160	23	.368	.248
635.73	900	5.44	4900	.173	1.65	.286	.268
641.18	1000	5.44	5440	.189	23	.435	.293
646.62	1080	5.55	6000	.201	1.65	.332	.312
652.20	1200	5.66	6800	.216	2.3	.497	.319
657.04	2000	5.66	11300	.228	1.55	.354	.354
663.60	2450	5.88	14400	.252	2.3	.580	.300
669.71	3050	6.20	18900	.261	2.5	.652	.456
676.01	3700	6.08	22500	.258	2.3	.594	.465
681.87	4270	5.31	25200	.292	1.5	.438	.438
687.83	4400	11.7	51500	.308	1.7	.524	.524
696.21				.310	2.5	.775	.542
702.58				.347		.566	.607
708.59	6750	3.02	20400	.362		.305	.634
714.64	7600	6.07	46100	.377		.342	.660
720.73	8700	6.10	53100	.393		.382	.686
726.85	9700	6.10	59100	.409		1.021	.716
732.97	10850	6.22	67500	.425	2.5	1.061	.744
739.29	12000	3.16	38000	.441	3.0	1.320	.991



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FIN BOX VERTICAL PLATES
FINGER LOADS & STRESSES

1	2	3	4	5	6	7	8	9
STA.	+ PF	AFT FUS. PRESSURE LOAD	MISCAL- ANOUS LOADS	TOTAL FINGER LOAD	AREA (SQ. IN.)	AREA (TENS.)	FC REF = 5/2 [6 OR 7]	M.S. = 67000/8
	PG. 1.05	PG. 6.17			PG. 1.05	PG. 1.05		
613.22	5100			5100	.242	.177	14,400	HIGH
613.03	3600			3600	.209	.202	8300	
624.85	3340	5060		3000	.216	.216	20,800	
630.29	4080	2600		6,480	.368	.248	13,500	
635.73	4900	1300		6,200	.286	.268	11,600	
641.18	5440	490		5,930	.435	.203	10,100	
646.62	6000	60		6,060	.332	.312	3,700	
652.28	6800			6,800	.437	.393	3,000	
657.04	11,300	+ 310	-3975 ⁺	-15,585	.354	.354	-22,000	
663.60	14,400	-1,040		13,360	.580	.300	12,100	
669.71	18,900	-2,460	+2350 ⁺	18,790	.652	.456	20,600	
674.01	22,500	-4,350		-24,850	.554	.465	-22,400	HIGH
681.87	25,200	-5,900		-31,100	.436	.438	-25,500	.65
687.83	52,700	-8,450		-61,100	.524	.524	-58,500	.115
696.21							REF. PG. 1.15	
702.58					.866	.407		
708.53	20,400		2500 Δ 3000 □ 5000 ○ 6000 ● 2500 Δ 3000 ○	25,000	.305	.434	20,400	HIGH
714.64	46,100			57,190	.942	.660	43,200	.55
720.73	53,100			58,660	.982	.688	42,500	.57
726.85	53,100			53,100	1.021	.716	41,200	.67
732.97	67,500			67,500	1.061	.744	40,000	.47
739.29	38,000			38,000	1.320	.391	12,200	HIGH

* $F_c = F_t = .33 \times 72,000 = 67,000$ PSI.

+ $STRETCH\ LOAD = 7250/2 = 3975$ PSI.

+ " " = $7250 \times 10 / 2 = 2350$ PSI.

ASTRUT LOAD STA 708 $F_t = 16,000/4 = 2500$ PSI.
ASTRUT LOAD STA 714 $F_t = 14,400/2 = 2500$ PSI.

PG. 2.22

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FIN BOX VERTICAL PLATES
FINGER ATTACHMENTS

THE FOLLOWING ANALYSIS CONTAINS SOME
OF THE MORE CRITICAL FINGER ATTACHMENTS.
ATTACHMENTS NOT LISTED MAY BE SUBSTANTIATED
BY COMPARING THE LOADS ON PG. 106 WITH
ANALYSIS.

STA.	LOAD (Pg. 106)	ATTACHMENTS	ALLOW. LOAD	M.S. =
681.87	31100	6-1/4" LOCK BOLTS	55800	.79
687.83	61100	6-5/8" LOCK BOLTS	87500	.43
696.21		(REF PG 114)		
732.97	67500	6-5/8" LOCK BOLTS	87500	.30

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FIN BOX VERTICAL PLATES

STA. 732.31 - STA. 742.5

LOADS

CRITICAL CASE IS R.P.D.

LOADS ARE FROM SIDE AIR LOAD ON FIN &
AFT FUS. INERTIA & AIR LOADS.

FIN SKIN LOADS (PER PLATE)

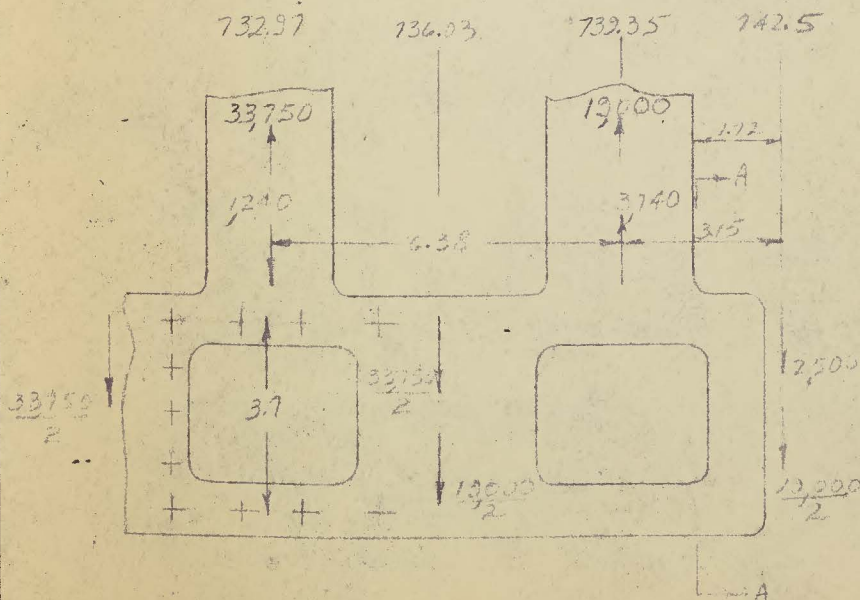
STA. 732.37 = $67500/2 = 33750$ LBS. (Pg. 1.06)

STA. 732.35 = $38000/2 = 19000$ LBS. (Pg. 1.06)

AFT FUS. INERTIA & SHEAR LOAD

TOTAL SHEAR ACTING @ STA. 742.5 = 10,000 LBS. (ESTIMATED)

LOAD PER PLATE = $\frac{10000}{4} = 2500$ LBS.



NOTE: ONLY OUTER PLATE ANALYSIS IS SHOWN,

HOWEVER, IT MAY BE USED TO SUBSTANTIATE

THE STRENGTH OF INNER PLATES SINCE DIMENSIONS

& MATERIAL THICKNESSES ARE SAME. INNER

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FIN BOX VERTICAL PLATES

STA 723.21 - STA 742.5

ATTACHMENT TO FIN @ STA 732.35

$$\text{LOADING ELEMENTS} = 19000 + 3740 \text{ (PG. 110)} \\ = 22740$$

$$\text{ALLOW LOAD } 5 - \frac{5}{16} \text{ IN. PLATES } 5 \times 7300 \\ 36500 \text{ LBS.}$$

$$\text{M.S.} = \frac{36500}{22740} - 1 = \underline{\underline{.61 \text{ M.S.}}}$$

BENDING SEC. A-A

$$\text{A} = 20 \text{ IN.}^2 \quad \text{MOMENT} = 1.72 \left(\frac{19000}{2} + 37500 \right) \text{ (PG. 110)} \\ = 22600 \text{ IN. LBS.}$$

$$I = 2 \times 20 \left(\frac{3.7}{2} \right)^2 = 1.37$$

$$S_b = \frac{20600 \times 3.7}{1.37 \times 2} = 21500 \text{ PSI.}$$

SPEC. A-A
(ASSUMED)

NOT CRITICAL

SHEAR STRESS

$$\text{M. F. SHEAR LOAD} = 33750 / 2 = 16875 \text{ LBS. (PG. 110)}$$

ASSUME PLATE THICKNESS AS .20 IN.

$$f_s = \frac{16875}{37 \times .20} = 22500 \text{ PSI.}$$

ALLOW. BUCKLING STRESS (REF. MIL-D-50336)

$$YR = 2.2$$

$$(b/t)_c = \frac{3.7}{.20 \times 2.2} = 8.4$$

$$F_{CR} = 38500$$

$$\text{M.S.} = \frac{38500}{22500} - 1 = \underline{\underline{.69 \text{ M.S.}}}$$

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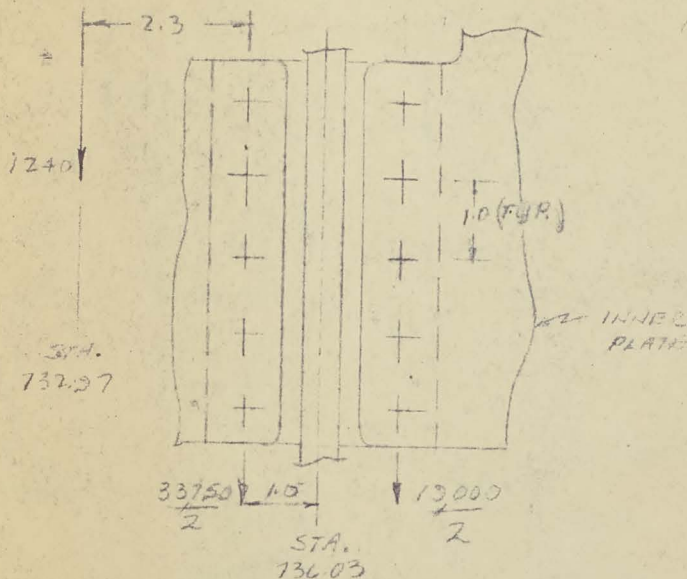
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FIN BOX VERTICAL PLATES

STA. 729.51 - STA. 747.5

ATTACHMENTS - INNER PLATE TO STA. 736.03 DIAPHR.



$$\begin{aligned} \text{MOMENT ON ATTACHMENTS} &= \left(\frac{33750}{2} - \frac{13000}{2} \right) 1.0 + 2.3 \times 1240 \\ &= 7400 + 2850 \\ &= 10250 \text{ IN. LBS.} \end{aligned}$$

$$I_P (\text{ATTACHMENTS}) = 8.5$$

$$\text{LOAD ON CRIT. BOLT FROM MOMENT} = \frac{10250 \times 1.84}{8.5} = 2220$$

$$\text{LOAD PER BOLT FROM SHEAR} = \frac{33750}{2} + 1240 = 3680 \#$$

$$\begin{aligned} \text{RESULT LOAD} &= \left(3680^2 + 2220^2 \right)^{1/2} \\ &= 4250 \# \end{aligned}$$

$$\text{M.S.} = \frac{4650}{4250} - 1 =$$

0.09175

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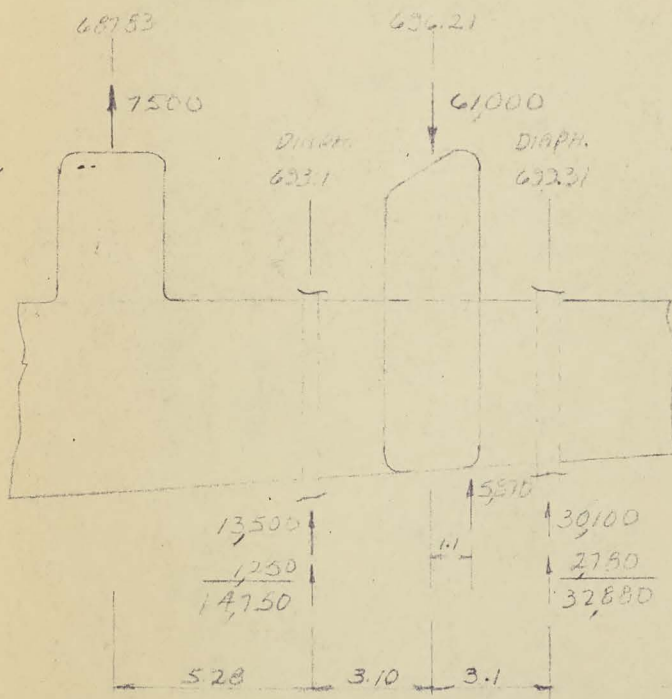
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FIN BOX VERTICAL PLATE

FINGER LOADS DUE TO FIN BOX 440" STRESS



JACK LOAD VERT. COMPONENT = 5870 LBS. (Pg. 2.22)

DIAPH. 623.1 LOADS = 14750# (Pg. 2.23)

DIAPH. 622.31 LOADS = 32880# (Pg. 2.24)

ASSUME LOADS ARE BRACED @ CORNERS IN
= STA. 687.53 & STA. 636.21.

$$\begin{aligned} \text{STA. 636.21 REACTION} &= (11.48 \times 32880 + 9.48 \times 5870 \\ &\quad + 5.28 \times 14750) / 8.38 \\ &= 376,000 + 55,400 + 78,000 \\ &\quad 8.38 \\ &= 61,000 \text{ LBS.} \end{aligned}$$

$$\begin{aligned} \text{STA. 687.53 REACTION} &= 61,000 - (32880 + 5870 + 14750) \\ &= 61,000 - 53,500 \\ &= 7,500 \text{ LBS.} \end{aligned}$$

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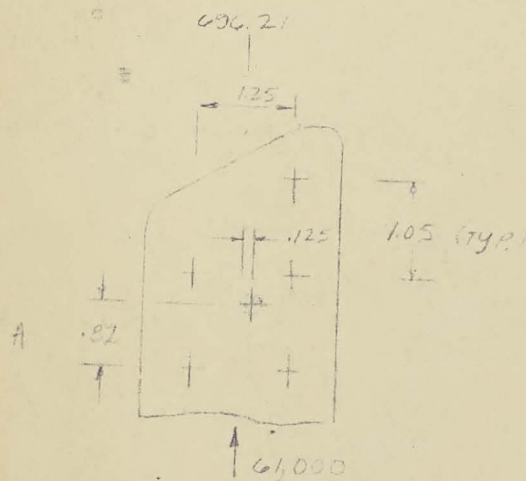
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FIL FOR VERT. PLATES

FINGER LOADS DUE TO ELEV. JACK & CYCLE FITTING

STA 696.21 ATTACH. POINTS

LOAD ON FINGER = 61,000 LBS. (REF. PG. 1.13)



MOMENT ON ATTACH. LIPS $61,000 \times 125 = 7,500 \text{ IN. LBS.}$
 $I.P. = 4.95$

LOAD ON CRIT. BOLT FROM MOMENT

HORIZ. COMPONENT = $7,500 \times .82 = 1,240 \text{ LBS.}$
 4.95

VERT. COMPONENT = $7,500 \times .85 = 1,230 \text{ LBS.}$
 4.95

LOAD PER BOLT FROM VERT. LOAD = $61,000 = 12,200 \text{ LBS.}$

RESULTANT $\sqrt{(12,200 - 1,230)^2 + (1,240)^2}$
 $13,500 \text{ LBS.}$

ALLOW. LOAD = 14,500 LBS.

M.S. = $14,500 - 1 = 13,500$

.08 M.S.

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FIN 21 ST. 11.5

FINGER R. 11.5 DUE TO ELEV. 11.5 / 11.5
STA. 636 21 FINGER STRESS

END LOAD = 6,000

AREA = 2 (33.23)

= 1.52

$f_c = \frac{6000}{1.52}$

1.52

= 40,000 PSI

M.S. 67000 - 11.5

40000

.67 M.S.

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PART II - DIAPHRAGMS

DESCRIPTION	
WEB STRESSES - STA. 607.63 - STA. 736.03	2.02
FLANGE LOADS - STA. 607.63 - STA. 736.03	2.03
FLANGE ATTACHMENTS - STA. 669.7	2.05
FLANGE ATTACHMENTS - STA. 681.57	2.06
FLANGE ATTACHMENTS - STA. 657.34	2.08
STA. 607.63 DIAPH.	2.10
STA. 667.83 DIAPH.	2.11
LOAD ON STA. 693.11 & 699.31 DIAPHRAGMS	2.13
STA. 693.11 DIAPH.	2.20
STA. 699.31 DIAPH.	2.23
STA. 711.32 DIAPH.	2.26
	2.29

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DIAPHRAGMS

DESCRIPTION

THE PRINCIPAL FUNCTION OF THE DIAPHRAGMS IS TO TRANSFER FIN SKIN DIFFERENTIAL LOADS TO THE WING UPPER & LOWER SURFACES.

ALL DIAPHRAGMS BETWEEN STA. 407.63 & STA. 687.83 ARE FABRICATED PARTS THE FLANGES BEING BOLTED TO THE WEBS. DIAPHRAGMS BETWEEN STA. 688.11 & STA. 736.03 ARE FORGED PARTS INCORPORATING INTEGRAL FLANGES & WEBS.

IN THE ANALYSIS THAT FOLLOWED, IN GENERAL, ONLY THE MOST CRITICAL ITEMS ARE ANALYZED. DIAPHRAGMS NOT INCLUDED IN ANALYSIS MAY BE SUBSTANTIATED BY LOAD & ANALYSIS COMPARISON.

A. V. ROE CANADA LIMITED
MALTON - ONTARIO
TECHNICAL DEPARTMENT (Aircraft)

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DIAPHRAGMS

WEB STRESSES - STA 607.63 - STA 736.03

1	2	3	4	5	6	7	8	9
STA.	p	d	$P_r = p \cdot d$	MISCELLANEOUS LOADS	$S_{(TOTAL)} = 4 + 5$	$h =$ WEB DEPTH	$t =$ WEB THICKNESS	$f_s =$ $S/h \cdot t$
	REF. PG. 1.03							
607.63	550	13.6	7500	(REF. PG. 2.11)				
624.85	700	14.1	9870	(REF. 7/0562/25				
635.73	900	10.88	9800		9800	8.6	2x.081	7000
646.62	1100	11.10	12200	$\frac{722}{2} =$	12200	7.8	2x.031	8600
657.24	1300	11.32	21500	$\frac{7050}{2} =$	25475	6.8	2x.125	15000
669.70	3000	12.41	37200		40725	6.4	2x.150	21200
681.87	4300	8.84	38000		38000	5.7	2x.188	17700
687.83	4400	11.7	51500	(REF. PG. 2.17)				
693.11				(REF. PG. 2.23)				
693.31				(REF. PG. 2.24)				
705.58								
711.60	7000	6.05	42300	(REF. PG. 2.31)				
717.47	7200	6.67	48100	$\frac{2500 \Delta}{2280 \circ}$	52850	4.9	.50	31800*
723.79	2300	6.12	57000		57000	4.9	.50	35000*
729.01	10400	6.12	63600		63600	4.9	.50	33000*
736.03	11000	5.63	62000		62000	4.9	.50	38000*

* $f_s = \frac{S}{h \cdot t} \times 3$

$h \cdot t = 2$

† REF. PG. 5.03

Δ FROM STRUT LOAD = $\frac{10000}{4} = 2500$ # (PG. 5.06)

○ FROM ENGINE LOAD = $\frac{3.8 \times 12000}{2} = 2280$ # (PG. 2.29)



AVRO AIRCRAFT LIMITED

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FIN ATTACHMENT
TO FUSELAGE

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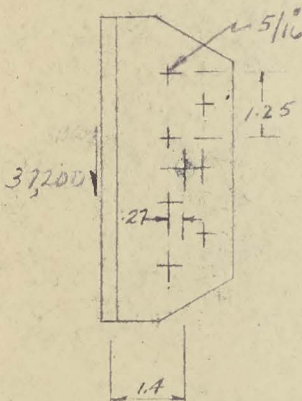
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DIAPHRAGMS

FLANGE ATTACHMENTS - STA 463.7
VERTICAL ATTACHMENT TO WEB

LOAD ON FLG. = 37,200# (PG. 2.05)

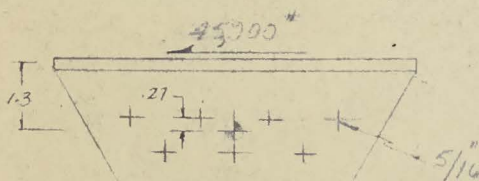
MOMENT = $1.4 \times 37,200 = 52,000$ IN. LBS. $I_F = 11.6$

LOAD ON CRITICAL BOLT FROM MOMENT

HORIZ. COMPONENT = $\frac{1.87 \times 52,000}{11.6} = 8,400$ #VERT. COMPONENT = $\frac{.27 \times 52,000}{11.6} = 1,210$ #LOAD PER BOLT FROM VERT. SHEAR = $37,200 / 7$
= 5,300RESULTANT = $\left[(5,300 + 1,210)^2 + 8,400^2 \right]^{1/2}$
= 10,400 LBS.M.S. = $\frac{12,900}{10,400} - 1 =$

.23 M.S.

HORIZONTAL ATTACHMENT TO WEB



LOAD ON FLG. = 45,900# (PG. 2.05)

MOMENT = $1.3 \times 45,900 = 59,600$ IN. LBS. $I_F = 12.8$

LOAD ON CRIT. BOLT FROM MOMENT

VERTICAL COMPONENT = $\frac{59,600 \times 2.0}{12.8} = 9,300$ #HORIZ. COMPONENT = $\frac{59,600 \times .27}{12.8} = 1,260$ #LOAD PER BOLT FROM HORIZ. SHEAR = $45,900 / 7$
= 6,550 LBS.RESULTANT = $\left[(6,550 + 1,260)^2 + 9,300^2 \right]^{1/2}$
= 12,100 LBS.M.S. = $\frac{12,900}{12,100} - 1 =$

.06 M.S.



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FIN ATTACHMENT
TO FUSELAGE.

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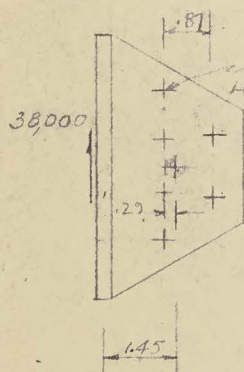
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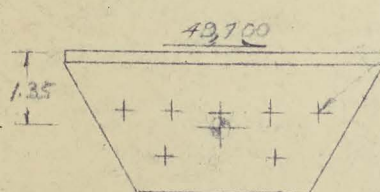
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DIAPHRAGMSFLANGE ATTACHMENTS - STA. 681.87VERTICAL ATTACHMENT TO WEB

LOAD ON FLG. = 38,000 (PG. 2.05)

MOMENT = $1.45 \times 38,000 = 55,000 \text{ IN. LBS.}$ $I_P = 9.94$

LOAD ON CRIT. BOLT FROM MOMENT

HORIZ. COMPONENT = $\frac{1.42 \times 55,000}{9.94} = 9,000$ VERT. COMPONENT = $\frac{.29 \times 55,000}{9.94} = 1,600$ LOAD PER BOLT FROM VERT. SHEAR = $\frac{38,000}{6} = 6,330$ RESULTANT = $\left[(6,330 + 1,600)^2 + 9,000^2 \right]^{1/2} = 13,000 \text{ LBS.}$ M.S. = $\frac{14,600}{13,000} - 1 =$.12 M.S.HORIZONTAL ATTACHMENT TO WEB

LOAD ON FLG. = 49,700 (PG. 2.05)

MOMENT = $1.35 \times 49,700 = 67,100 \text{ IN. LBS.}$ $I_P = 11.23$

LOAD ON CRITICAL BOLT FROM MOMENT

VERT. COMPONENT = $\frac{1.8 \times 67,100}{11.23} = 21,200$ HORIZ. COMPONENT = $\frac{.35 \times 67,100}{11.23} = 2,080$ LOAD PER BOLT FROM HORIZ. SHEAR = $\frac{49,700}{7} = 7,100 \text{ LBS.}$ RESULTANT = $\left[(21,200)^2 + (2,080 + 7,100)^2 \right]^{1/2} = 24,400$ M.S. = $\frac{14,600}{24,400} - 1 =$.02 M.S.

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT No.

SHEET No.

209

AIRCRAFT:

C-105

PREPARED BY

DATE

McJABE

12-8-55

CHECKED BY

DATE

DIAPHRAGMS

FLANGE ATTACHMENTS-- STA 681.87

HORIZ. FLANGE SHEAR

AT OUTER ROW OF ATTACHMENTS TO WEB,

$$f_s = \frac{49700 \times 3}{5.6 \times 35 \times 2} = 38000 \text{ PSI.}$$

$$F_s = 45000 \text{ PSI.}$$

$$M.S. = \frac{45000 - 1}{38000} =$$

.18 M.S.

VERT. FLANGE SHEAR

AT OUTER ROW OF ATTACHMENTS TO WEB

$$f_s = \frac{38000 \times 3}{4.65 \times 35 \times 2} = 35000 \text{ PSI.}$$

$$F_s = 45000$$

$$M.S. = \frac{45000 - 1}{35000} =$$

.28 M.S.

A. V. ROE CANADA LIMITED
MALTON ONTARIO
TECHNICAL DEPARTMENT (Aircraft)

REPORT No. _____

SHEET No. 2.10

AIRCRAFT:

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12-8-55

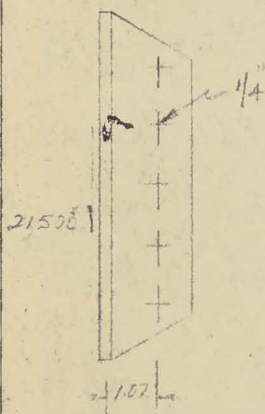
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DATE

DIAPHRAGMS

FLANGE ATTACHMENTS - STA 657.94

VERTICAL ATTACHMENT TO WEB



$$\text{LOAD ON FLG.} = 21500 \# \text{ (PG. 2.05)}$$

$$\text{MOMENT} = 102 \times 21500 = 22,000 \text{ IN. LBS}$$

$$I_p = 11.0$$

$$\text{LOAD ON 1ST BOLT FROM MOMENT} = \frac{22,000 \times 2.1}{11.0}$$

$$= 4200 \#$$

$$\text{LOAD PER BOLT FROM SHEAR} = \frac{21500}{5} = 4300 \#$$

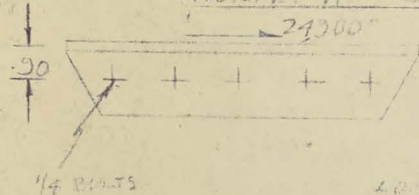
$$\text{RESULTANT LOAD} = (4200^2 + 4300^2)^{1/2} = 6000 \#$$

$$\text{ALLOW. LOAD} = 8560$$

$$M.S. = \frac{8560 - 1}{6000}$$

$$.43 \text{ M.S.}$$

HORIZ. ATTACHMENT TO WEB



$$\text{LOAD ON FLG.} = 24300 \# \text{ (PG. 2.05)}$$

$$\text{MOMENT} = .90 \times 24300 = 22,400 \text{ IN. LBS}$$

$$I_p = 15.6$$

$$\text{LOAD ON 1ST BOLT FROM MOMENT} = \frac{22,400 \times 2.5}{15.6}$$

$$= 4000 \#$$

$$\text{LOAD PER BOLT FROM SHEAR} = \frac{24300}{5} = 5000 \#$$

$$\text{RESULTANT} = (4000^2 + 5000^2)^{1/2} = 6400 \#$$

$$M.S. = \frac{8560 - 1}{6400}$$

$$.34 \text{ M.S.}$$

A. V. ROE CANADA LIMITED
MALTON - ONTARIO
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. _____

SHEET NO. 2.12

AIRCRAFT:

C-105

PREPARED BY

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11-23-55

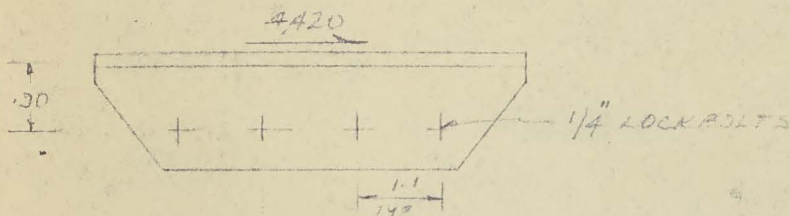
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DATE

DIAPHRAGMS

STA. 607.63

HORIZ. FLANGE ATTACHMENT TO WEB



$$\text{MOMENT ON ATTACHMENTS} = 150 \times 4420 = 3380 \text{ IN. LBS.}$$

$$I_p = 6.0$$

$$\text{LOAD ON CRIT. BOLT FROM MOMENT} = \frac{1.65 \times 3380}{6} = 1030 \text{ LBS.}$$

$$\text{LOAD PER BOLT FROM SHEAR} = \frac{4420}{4} = 1105 \text{ LBS.}$$

$$\text{RESULTANT} = (1030^2 + 1105^2)^{1/2} = 1550 \text{ LBS.}$$

$$\text{ALLOW. LOAD } 1/4" \text{ BOLTS IN } 2 \times .081 = 6240 \text{ LBS.}$$

$$\text{M.S.} = \frac{6240}{1550} - 1 = \text{HIGH M.S.}$$

FLANGE STRESS NOT CRITICAL
VERTICAL ATTACHMENTS TO WEB O.K.

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. _____

SHEET NO. _____

2.13

AIRCRAFT:

C-105

PREPARED BY

MCCABE

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11-25-55

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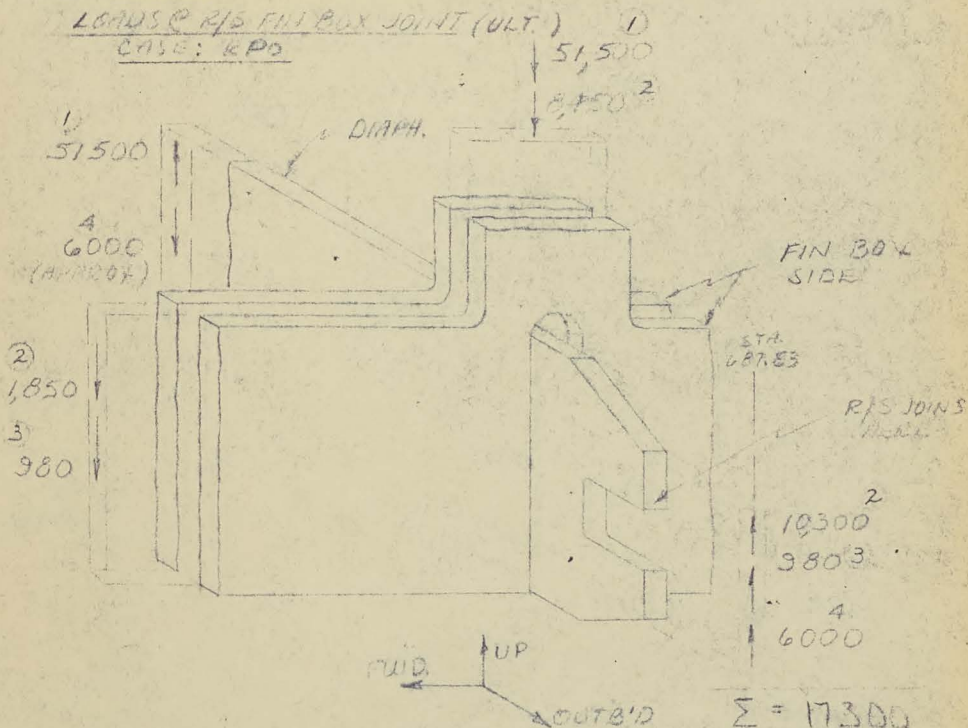
DATE

DIAPHRAGMS

STA. 687.83

LOADS @ R/S FIN BOX JOINT (ULT.)

CASE: RPD



- 1 DIRECT LOAD FROM FIN SKIN DUE TO SIDE LOAD ON FIN (PG 203)
- 2 AFT FUS. INERTIA LOAD (REF. PG. 6.17)
- 3 FIN & FIN BOX INERTIA LOAD (STA. 624 - STA. 687) (ESTIMATED)
- 4 SPAR SHEAR DUE TO FIN ROLL MOMENT (ESTIMATED, RYALE)

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2/14/56

REPORT NO.

SHEET NO.

2.15

PREPARED BY

DATE _____

McCABE

7-7-55

CHECKED BY

DATE _____

AIRCRAFT:

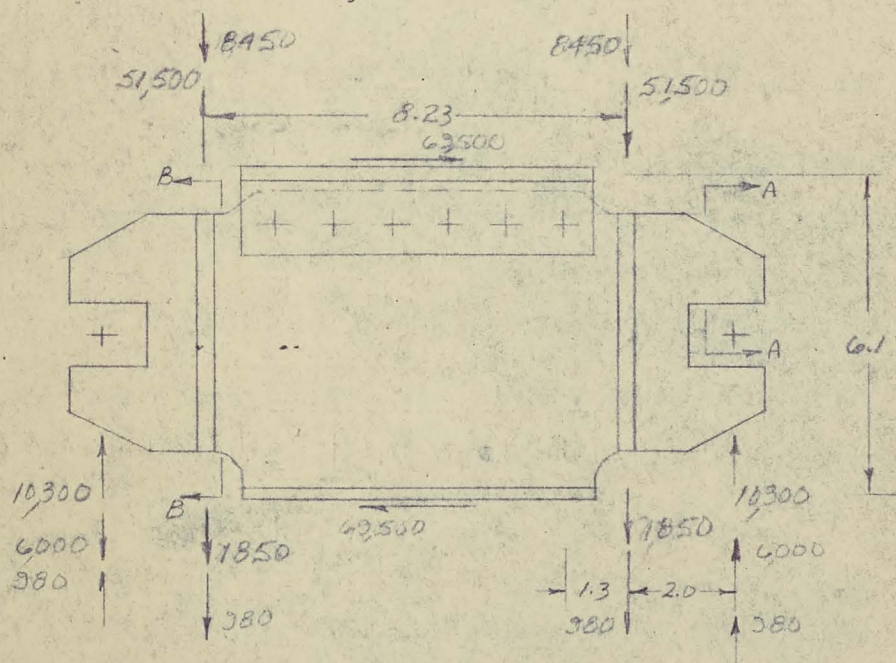
C-105

DIAPHRAGM

57A. 687. 63

CRITICAL DESIGN CASE FOR WEB AND FLANGES
IS R.P.D.

(APPLIED LOADS - REF PG. 2.13)



FLANGE COUPLER LOADS = $\frac{51,500 \times 8.23}{6.1} = 69,500 \text{ LBS.}$

A. V. ROE CANADA LIMITED
MALTON, ONTARIO
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO.

SHEET NO. 216

AIRCRAFT:

C-105

PREPARED BY

DATE

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10-18-55

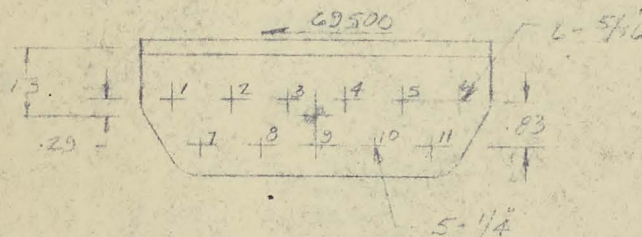
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DATE

DIAPHRAGM

STA. 687.83

UPPER FLANGE ATTACHMENT TO WEB



MOMENT ON ATTACHMENTS = $1.3 \times 69,500 = 90,300$ IN. LBS.

$M/E = 90,300 / 47.97 = 1,880$

ITEM	2	3	4	5	6	7	8	9	10	11
AREA	Z	Y	$A \cdot Z$	$A \cdot Y$	LOAD FROM MOMENT		LOAD FROM		TOTAL	RESULT.
					HOR. Z	VERT.	HOR. Z	VERT.	MOMENT	
					$M \cdot Z \cdot A$	$M \cdot Y \cdot A$	$69,500 \cdot A$	$Z \cdot A$	$7 + 9$	$8 + 10$
1	1.55	.29	2.8	.12	12.1	850	8160	7540	8390	11700
2			1.67		4.3					
3			.56		.5					
4			.56		.5					
5			1.67		4.3					
6	1.55	.29	2.8	.12	12.1	850	8160	7540	8390	11700
7	1.0	.54	2.2	.29	4.8	-1010	4130	4860	3820	5620
8			1.1		1.2					
9			0		0					
10			1.1		1.2					
11	1.0	.54	2.2	.29	4.8	-1010	4130	4860	3850	5620
Σ	14.3			2.17	45.3					

$M.S. = 14,600 - 1 = 11,700$

.25 M.S.



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. _____

SHEET NO. 2.17

AIRCRAFT:

C-105

PREPARED BY

MCCABE

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7-7-55

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DIAPHRAGMSSTA. 62.23WEB STRESS

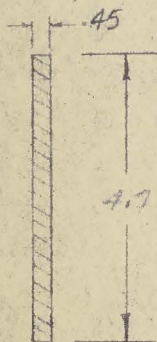
WEB SHEAR = 51,500 LBS. (PG. 2.15)

WEB AREA $4.7 \times .45 = 2.11$

$$f_s = \frac{51,500 \times 3}{2.11 \times 2} = 36,600 \text{ PSI}$$

$$F_s = 45,000 \text{ PSI}$$

$$M.S. = \frac{45,000}{36,600} - 1 =$$

.23 M.S.SEC. B-B (PG. 2.15)SHEAR - SEC. A-A (PG. 2.15)

CRIT. CASE SYMM. FLT. (REF. PG. 2.14 FOR LOADS)

SPAR SHEAR = 16,100 + 1,330 = 17,430 LBS.

AREA = $16 \times .45 = .72 \text{ IN}^2$

$$f_s = \frac{17,430 \times 3}{.72 \times 2} = 36,400 \text{ PSI}$$

$$M.S. = \frac{45,000}{36,400} - 1 =$$

.23 M.S.FLANGE SHEAR STRESS

FLANGE LOAD = 69,500 LBS. (PG. 2.15)

$$f_s = \frac{69,500 \times 3}{2(6.9 \times .20) \times 2} = 37,800 \text{ PSI}$$

$$F_s = 45,000 \text{ PSI}$$

$$M.S. = \frac{45,000}{37,800} - 1 =$$

.19 M.S.

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT:

C-105

REPORT No. _____

SHEET No. 221

PREPARED BY

DATE

MCCABE

11-24-55

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DATE

DIAPHRAGMS

LOADS ON STA. 633.11 & STA. 633.31 DIAPHRAGMS

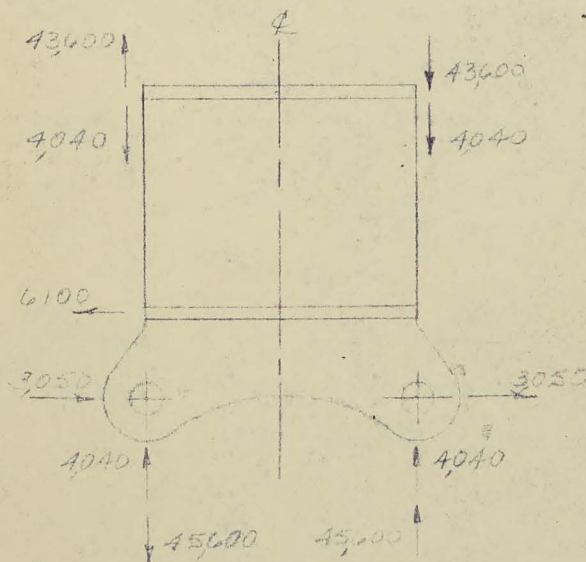
LOADS ON DIAPHRAGM LOGS

$$\text{LOADS FROM VERT. LOAD} = \frac{8080}{2} = 4040 \text{ LBS.}$$

$$\text{LOADS FROM SIDE LOAD} = \frac{52.4 \times 6100}{7.0} = 45600 \text{ LBS.}$$

$$\text{AND } \frac{6100}{2} = 3050"$$

SUMMARY OF LOADS ON DIAPHS. STA. 633.11 & STA. 633.31 FROM FUS. STRET LOADS



A SIMPLE DISTRIBUTION WILL BE ASSUMED IN DETERMINING THE AMOUNT OF THE ABOVE LOADS DISTRIBUTED TO ADJACENT DIAPHRAGMS.

STATION

4.27

$$\text{STA. 633.11 DIAPH.} = 1.23/4.2 \times \text{LOAD} = .31 \times \text{LOAD}$$

$$\text{STA. 633.31 DIAPH.} = 4.21/4.2 \times \text{LOAD} = .69 \times \text{LOAD}$$

A. V. ROE CANADA LIMITED
MALTON - ONTARIO
TECHNICAL DEPARTMENT (Aircraft)

REPORT No. _____

SHEET No. 222

AIRCRAFT:

C-105

PREPARED BY

DATE

MCCABE

11-24-55

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DATE

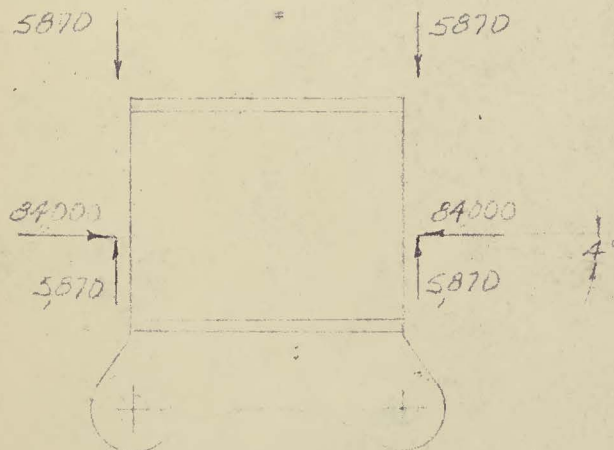
DIAPHRAGMS

LOADS ON STA. 693.11 & STA. 693.31 DIAPHRAGMS

LOADS FROM ELEV. JACK ARE IMPOSED ON DIAPHS STA. 693.11 & STA. 693.31 BY ELEV. JACK FITTING.

ELEV. JACK LOAD = 105,200[#] LBS. (REF 7/05-2/26, PG. 14)
ELEV. JACK LOAD INCORPORATES A 1.25 CONTROL FACTOR. SINCE LOAD WILL BE COMBINED WITH TUS. STRUT LOADS, 1.25 FACTOR WILL BE REMOVED. THEREFORE,

$$\text{REDUCED ELEV. JACK LOAD} = \frac{105,200}{1.25} = 84,300 \text{ LBS.}$$



$$\text{JACK LOAD HORIZ. COMPONENT} = 84,300 \cos 4^\circ = 84,000 \text{ LBS.}$$

$$\text{JACK LOAD VERT. COMPONENT} = 84,300 \sin 4^\circ = 5,870 \text{ LBS.}$$

IN DISTRIBUTING JACK LOADS TO DIAPHRAGMS 693.11 & 693.31 USE RATIOS USED IN DISTRIBUTING STRUT LOADS. (REF. PG. 2.21)

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. _____

SHEET No. 2.25

AIRCRAFT:

C-105

PREPARED BY

McCABE

DATE

11-30-55

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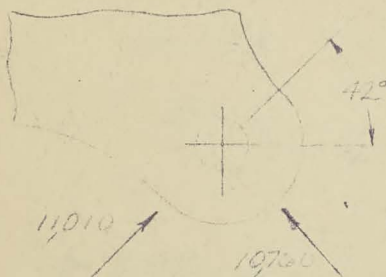
DATE

DIAPHRAGMS

STA. 653.11 DIAPHRAG. 7

LUGS

TRANSVERSE & AXIAL COMPRESSION LOADS



$$\begin{aligned} \text{TRANSVERSE LOAD} &= (14,500 + 12,500) \sin 42^\circ \\ &= 13,500 \sin 42^\circ \\ &= 10,300 + 710 \\ &= \underline{11,010 \text{ LBS.}} \end{aligned}$$

$$\begin{aligned} \text{AXIAL LOAD} &= (14,500 + 12,500) \cos 42^\circ \\ &= 13,500 \cos 42^\circ \\ &= 11,400 - 640 \\ &= \underline{10,760 \text{ LBS. (COMPRESSION)}} \end{aligned}$$

MARGIN OF SAFETY

CRITICAL MARGIN OF SAFETY IS FOR CONTINUOUS
AXIAL & TRANSVERSE LOADS.

$$R_A = \frac{8,370}{27,500} = .30$$

$$R_T = \frac{23,420}{26,800} = .87$$

$$M.S. = \frac{1}{(R_A^{10} + R_T^{10})^{0.25}} - 1$$

$$= \frac{1}{(33^{10} + 87^{10})^{0.25}} - 1 = \underline{\underline{.56 \text{ M.S.}}}$$

DIAPHRAGM ATTACHMENT TO ELEV. JACK FITTING

REF. 7/0562/26, PG. 19

DIAPHRAGM FLANGE ATTACHMENT TO HOOD PLATES

REF. PG. 3.33

TECHNICAL DEPARTMENT (Aircraft)

REPORT No.

SHEET No. 2.27

AIRCRAFT:

C-105

PREPARED BY

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DATE

12-1-55

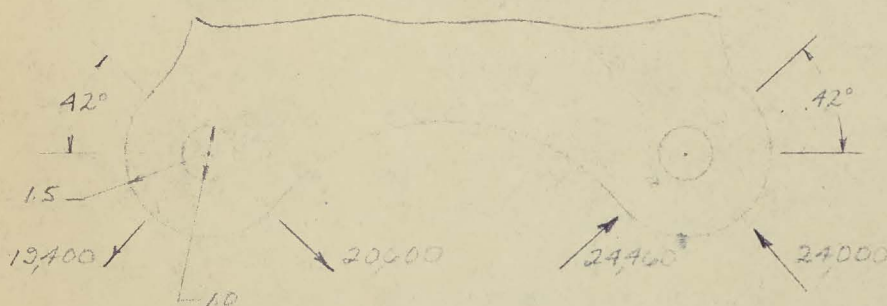
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DATE

DIAPHRAGMS

STA. 695.31 DIAPHRAGM

LUGS



TRANSVERSE & AXIAL TENSION LOADS (LEFT)

$$\begin{aligned} \text{TRANSVERSE LOAD} &= (31,500 - 2,100) \sin 42^\circ + 2,100 \cos 42^\circ \\ &= 19,200 + 1,560 \\ &= \underline{20,600 \text{ LBS.}} \end{aligned}$$

$$\begin{aligned} \text{AXIAL LOAD} &= (31,500 - 2,100) \cos 42^\circ - 2,100 \sin 42^\circ \\ &= 21,300 - 1,400 \\ &= \underline{19,700 \text{ LBS.}} \end{aligned}$$

TRANSVERSE & AXIAL COMPRESSION LOADS (RIGHT)

$$\begin{aligned} \text{TRANSVERSE LOAD} &= (31,500 + 2,100) \sin 42^\circ + 2,100 \cos 42^\circ \\ &= 22,300 + 1,560 \\ &= \underline{24,460 \text{ LBS.}} \end{aligned}$$

$$\begin{aligned} \text{AXIAL LOAD} &= (31,500 + 2,100) \cos 42^\circ - 2,100 \sin 42^\circ \\ &= 25,400 - 1,400 \\ &= \underline{24,000 \text{ LBS.}} \end{aligned}$$



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT No.

SHEET No.

3.03

AIRCRAFT:

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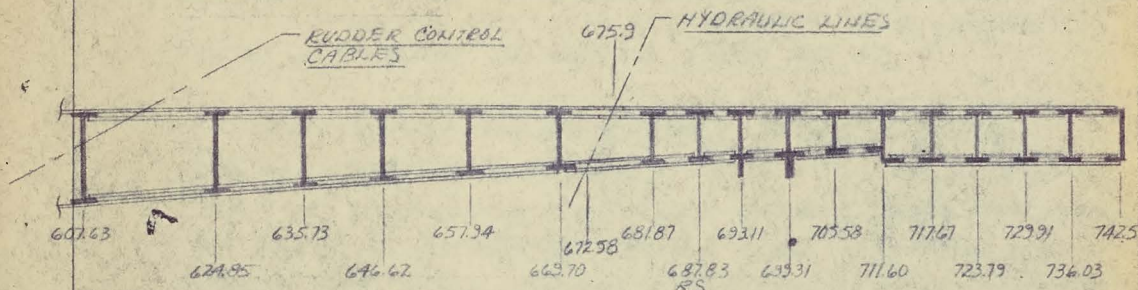
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DIAGRAM - LOCATION & SIZES - HORIZ. PLATES



PROFILE OF FINBOX STA. 607.63-742.5

FINBOX SPLICE PLATES - UPPER & LOWER SURFACES STA. 607.63 - STA. 742.50

STA. TO STA.	UPPER SURFACE		LOWER SURFACE	
	INNER	OUTER	INNER	OUTER
607.63 - 624.85	.204	.204	.125	.188
624.85 - 635.73	↑	↑	↑	↑
635.73 - 646.62	↑	↑	↑	↑
646.62 - 657.34	↑	↑	↑	↑
657.34 - 669.70	↑	.204	.125	.188
669.70 - 681.87	↑	.250 †	.204	.204
681.87 - 687.83	.204	↑	.204	.204
687.83 - 693.11	.250	↑	.204	.204
693.11 - 699.31	.188	↑	.188	.188
699.31 - 705.58	↑ *	↑	↑	↑
705.58 - 711.60	↑ *	↑	↑	↑
711.60 - 717.67	↑ *	↑	↑	↑
717.67 - 723.79	↑ *	↑	↑ *	↑
723.79 - 729.91	↑	↑	↑ *	↑
729.91 - 736.03	↑	↑	↑ *	↑
736.03 - 742.50	.188	.250	.188 *	.188

NOTE: ALL PLATES ARE 75S-76

* LIGHTENING HOLE



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO.

SHEET NO.

304

AIRCRAFT:

C-105

PREPARED BY

DATE

McCabe

7-12-55

CHECKED BY

DATE

SEC. 1. - UPPER & LOWER SURFACE PLATES - STA 607.63 - STA 681.87

THE FOLLOWING ANALYSIS COVERS UPPER & LOWER PLATES BETWEEN STA. 607.63 & STA. 681.87

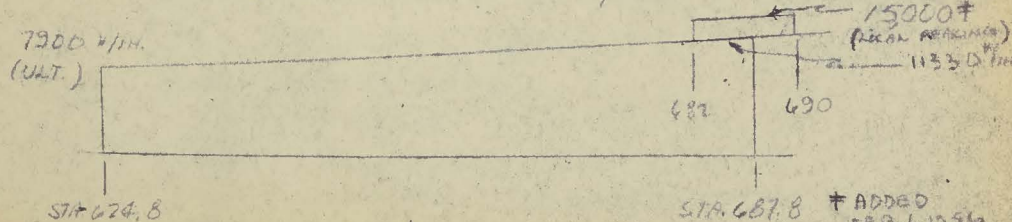
LOWER SURFACE INNER ~~INNER~~ PLATES BETWEEN STA. 607.63 & STA. 669.70 ARE .125 THICK. A 1.0" WIDE STRIP OF MAGNESIUM ATTACHED BETWEEN ^{INNER} ~~INNER~~ PLATES WILL BE USED TO DECREASE PANEL SIZE IN ORDER TO RAISE THE PANEL CRITICAL BUCKLING STRESS. ANALYSIS OF INNER PLATE BETWEEN STA. 657.94 & STA. 669.70 SUBSTANTIATES STRENGTH OF ALL LOWER SURFACE PLATES BETWEEN STA. 607.63 & STA. 669.70.

UPPER SURFACE INNER & OUTER PLATES BETWEEN STA. 624.85 & STA. 669.70 ARE BASICALLY .204 THICK, BUT ARE MACHINED DOWN TO .14 (NOMINAL) * IN AREA WHERE NET SECTION IS NOT CRITICAL. AS IN LOWER SURFACE PLATES, A MAGNESIUM STRIP IS BEING USED TO INCREASE THE PANEL CRITICAL BUCKLING STRESS. ANALYSIS OF INNER PLATE BETWEEN STA. 657.94 & STA. 669.70 SUBSTANTIATES STRENGTH OF ALL UPPER SURFACE PLATES BETWEEN STA. 624.85 & STA. 669.70.

A SEPARATE ANALYSIS CONSIDERING THE EFFECTS OF ACCESS HOLES IN PLATES BETWEEN STA. 607.63 & STA. 624.85 (UPPER) AND STA. 669.70 & STA. 681.87 (UPPER & LOWER) IS INCLUDED.

WING SKIN END LOAD

UPPER SURFACE COMPRESSION & LOWER SURFACE TENSION



NOTE: UPPER SURFACE TENSION & LOWER SURFACE COMPRESSION ARE 50% OF ABOVE LOADS

* NOTE: SUBSEQUENT ACTION ELIMINATED MACHINED RECESS IN PLATES AND RESTORED THEM TO THE ORIGINAL UNIFORM .204 THICKNESS (12-7-55)



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

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AIRCRAFT:

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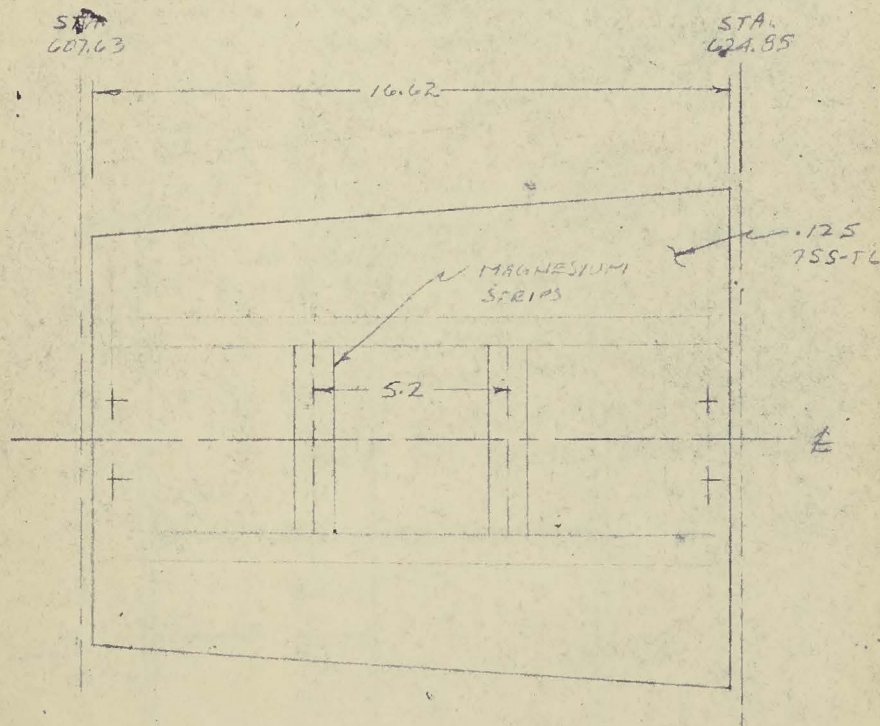
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LOWER SURFACE SPLICE PLATE - INNER STA. 607.63 - STA. 624.85



LWR. SURFACE SKIN LOAD STA. 624.85

TENSION = 7900#/IN. (PG 3.04)

COMPRESSION = $7900/2 = 3950\#/IN.$ COMPRESSION

$$\text{LOAD PER IN. ON PLATE} = \frac{3950 \times 17.22}{2 \times 16.62} = 2050\#/IN.$$

$$f_c = 2050 / .125 = 16400 \text{ PSI.}$$

ALLOW. BUCKLING STRESS

$$K=1.9, b=5.2$$

$$(b/t)e = 5.2 / .125 \times 1.9 = 21.9$$

$$F_{cr} = 21000 \text{ PSI. (LOCKHARD SM. 336)}$$

$$M.S. = \frac{21000}{16400} - 1 =$$

28 M.S.



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO.

SHEET NO. 306

AIRCRAFT:

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DATE

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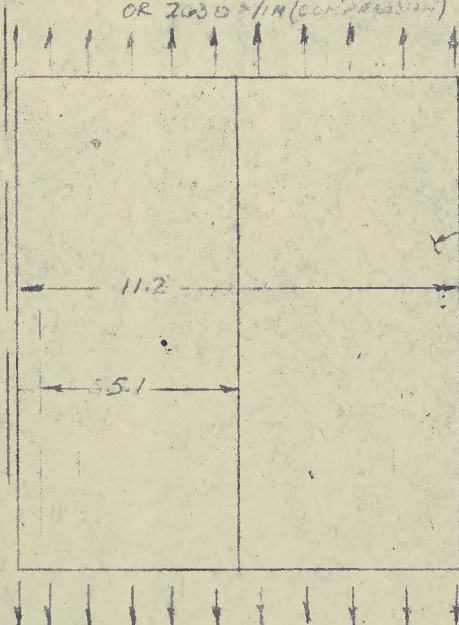
LOWER SURFACE SPICE PLATE - INNER - STA. 657.9 - STA. 669.7WING LOWER SURFACE END LOADS

MAX. TENSION STA. 663 = 10000 #/IN (PG 304)

MAX. COMPRESSION = 10000/2 = 5000 #/IN.

STA.
657.9STA.
669.75260 #/IN (TENSION)
OR 2630 #/IN (COMPRESSION)

REF. DWG. 7-1002-323

5260 #/IN (TENSION)
OR 2630 #/IN (COMP.)

$$\text{MAX. TENSION LOAD ON PLATE} = \frac{10000 \times 11.2}{2} = 5260 \text{ #/IN.}$$

$$\text{MAX. COMPRESSION LOAD} = \frac{5000 \times 11.2}{2} = 2630 \text{ #/IN.}$$

TENSION STRESS

TENSION LOAD PER IN. = 5260 #/IN.

FOR 1/4" LOCK BOLTS @ 11.2 IN (AVG.), TENSION EFFICIENCY

$$\text{FACTOR} = 1 - \frac{.25}{1.15} = .782$$

$$SE = \frac{5260}{.782} = 54000 \text{ PSI.}$$

$$.125 \times .782$$



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO.

SHEET NO. 307

AIRCRAFT:

C-105

PREPARED BY

McCabe

DATE

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DATE

LOWER SURFACE SPLICE PLATE - INNER STA. 657.3 - STA. 669.7
TENSION STRESS

$$F_t = .93 \times 72,000 = 67,000 \text{ PSI.}$$

$$M.S. = \frac{67,000}{54,000} - 1 = \underline{\underline{.24 M.S.}}$$

COMPRESSION STRESS

COMPRESSION LOAD PER IN. = 2630. #/IN. (PG. 3.06)

$$f_c = \frac{2630}{.125} = 21,000 \text{ PSI}$$

ALLOWABLE BUCKLING STRESS (REF LOCKWED SM 336)

$$a = 72$$

$$b = 5.1$$

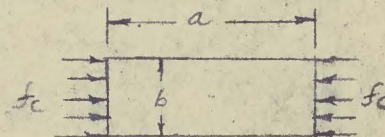
$$b/a = 5.1/72 = .71$$

$$K = 1.50$$

$$(b/t)_c = \frac{5.1}{.125 \times 1.2} = 21.5$$

$$F_{cc} = 22,000 \text{ PSI}$$

$$M.S. = \frac{22,000}{21,000} - 1 = \underline{\underline{.02 M.S.}}$$

ATTACHMENTS

$$\text{MAX. LOAD ON PLATE} = 11.4 \times 5260 = 60,000 \text{ LBS. (PG. 3.06)}$$

$$\text{ALLOW. LOAD } 1/4" \text{ LOCK BOLT IN } .125 = 4280^*$$

ATTACHMENT IS BY 19-1/4" LOCK BOLTS

$$\text{ALLOWABLE LOAD} = 13 \times 4280 = 81,500 \text{ LBS.}$$

$$M.S. = \frac{81,500}{60,000} - 1 = \underline{\underline{.36 M.S.}}$$



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SHEET NO.

308

AIRCRAFT:

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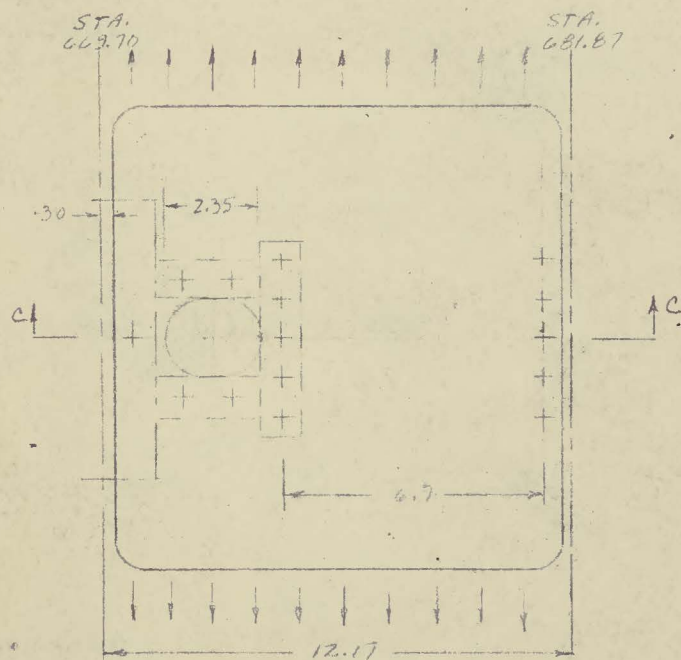
MCCABE

7-12-55

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LOWER SURFACE SPLICE PLATE-INNER (STA. 669.7-STA. 681.87)



AVG. LOWER SURFACE LOAD STA. 669.7 - STA. 681.87

TENSION = 10,700 #/IN. } PG 304

COMPRESSION = 5,350 #/IN.

SEC. C-CTENSIONNET TENSION LOAD ON PLATE $\frac{10700 \times 12.17}{2(12.17 - 3.7)} = 7700 \text{ #/IN.}$ $FL = \frac{7700}{1.20} = 38500 \text{ PSI.}$ $FL = 93 \times 72000 = 67000 \text{ PSI.}$ $M.S. = \frac{67000}{38500} - 1 =$.74 M.S.



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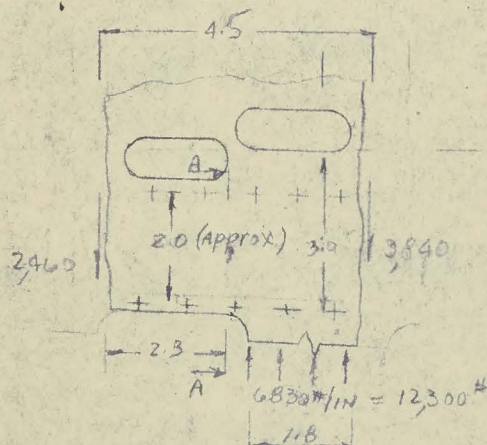
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UPPER SURFACE SPICE PLATE - OUTER (STA. 407.43 - STA. 424.95).

ASSUME END LOAD IN PLATE IS DIVERTED AROUND
SLOTS IN CENTER OF PLATE IN MANNER SHOWN IN
SKETCH BELOW.



ASSUME MAXIMUM END STRESS & SHEAR STRESS
COMBIN =

$$f_c = 6830 / .204 = 33,500 \text{ PSI.}$$

INCLUDE COMPRESSIVE STRESS

$$\text{AS A COLUMN; } P = .289 \times 704 = .059, L = 2.0, C = 4.0$$

$$\frac{L^2}{P} = \frac{2.0}{.059} = 17$$

$$P = 21.053$$

$$F_c = 67,000 \text{ PSI.}$$

$$R_c = 33,500 / 67,000 = .50$$

$$f_s = \frac{3840}{3.0 \times 2.04} = 16,000 \text{ PSI}$$

$$F_s = 44,000 \text{ PSI}$$

$$R_s = 16,000 / 44,000 = .36$$

MARGIN OF SAFETY

$$R_s + R_c = 1$$

$$M.S. =$$

$$.44 \text{ M.S.}$$



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UPPER SURFACE SPLICE PLATE - INNER - STA 657.9 - STA 669.7

AVERAGE WING SKIN LOAD STA 657-669.7 = 10,000 #/IN. (PG 3.04)

$$\text{LOAD PER IN. ON PLATE} = \frac{16.8}{11.2} \times \frac{10,000}{2} = 5260 \text{ #/IN.}$$

PANEL THICKNESS IS MAINTAINED FROM .20 TO .14 MAX. AT CENTER.

ALLOW. BUCKLING STRESS

$$a = 6.1$$

$$b = 4.2$$

$$t = .135 \quad (\text{.14 NOMINAL})$$

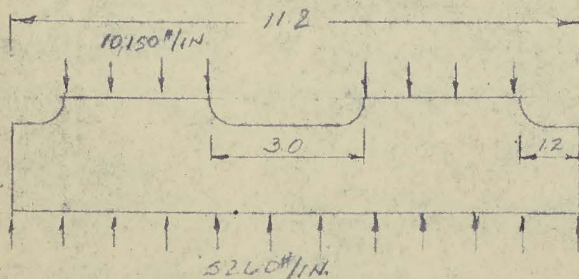
$$1/R = 1.9$$

$$(b/t)c = 4.5 / .135 \times 1.9 = 17.5$$

$$F_c = 34,000$$

$$S_0 = 4520 / .135 = 33,500 \text{ PSI.}$$

$$M.S. = \frac{34,000}{33,500} - 1$$

.01 M.S.NET SECTION

LOAD PER IN. ON PLATE = 5260 #/IN.

$$\text{NET LOAD AT SLOTS} = \frac{11.2}{5.8} \times 5260 = 10,150 \text{ #/IN.}$$

$$S_c = 10,150 / .20 = 51,000 \text{ PSI}$$

$$F_c = .93 \times 72,000 = 67,000 \text{ PSI}$$

$$M.S. = \frac{67,000}{51,000} - 1 =$$

.31 M.S.



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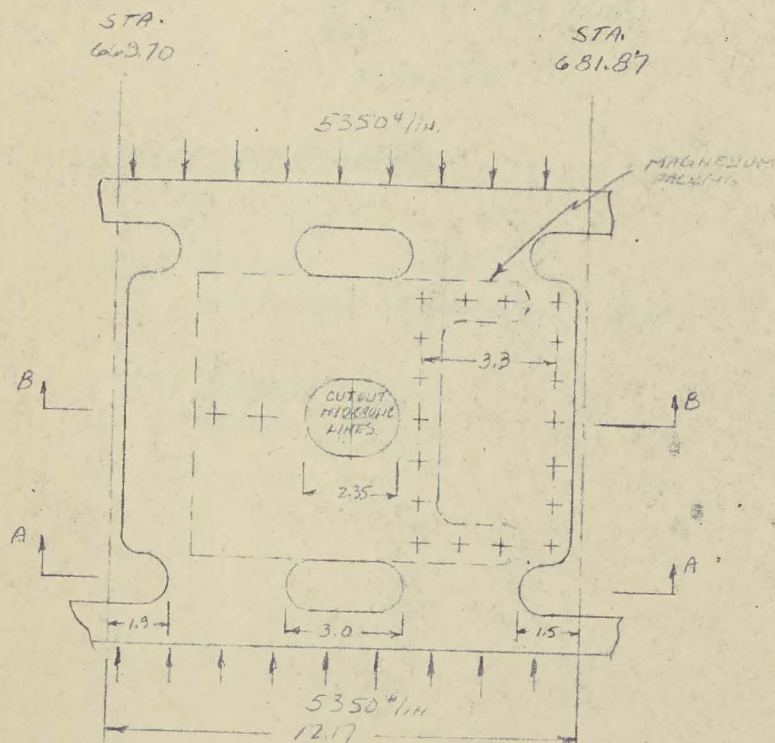
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UPPER SURFACE SPLICE PLATE - INNER (STA. 669.7 - STA. 681.87)



AVG. UPPER SURFACE LOAD STA. 669.7 - 681.87 = 10,700 #/IN COMPRESSION (PG. 3.04)
LOAD ON PLATE = $10,700 / 2 = 5,350 \text{ #/IN.}$

SEC. A-A

$$\text{NET LOAD} = \frac{5350 \times 12.17}{5.17} = 11,300 \text{ #/IN.}^2$$

$$S_c = 11,300 / 1.20 = 56,500 \text{ PSI.}$$

$$F_c = .93 \times 12,000 = 6,700 \text{ PSI.}$$

$$M.S. = \frac{67000}{56500} - 1 =$$

.19 M.S.



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UPPER SURFACE SPLICE PLATE - INNER (STA. 609.7 - STA. 601.87)
CUTOUT FOR HYDRAULIC LINES

SEC B-B

$$\text{NET LOAD} = \frac{5350 \times 12.17}{3.12} = 7130 \text{ \#/IN.}$$

$$f_c = 7130 / .20 = 35700 \text{ PSI.}$$

ALLOW. BUCKLING STRESS

$$K = 1.9$$

$$b = 3.3$$

$$(b/t)_c = 3.3 / 20 \times 1.9 = 8.7$$

$$F_{cr} = 53,000 \text{ PSI (WILHELM SIM 336)}$$

$$M.S. = \frac{59,000}{35,700} - 1 =$$

.65 M.S.



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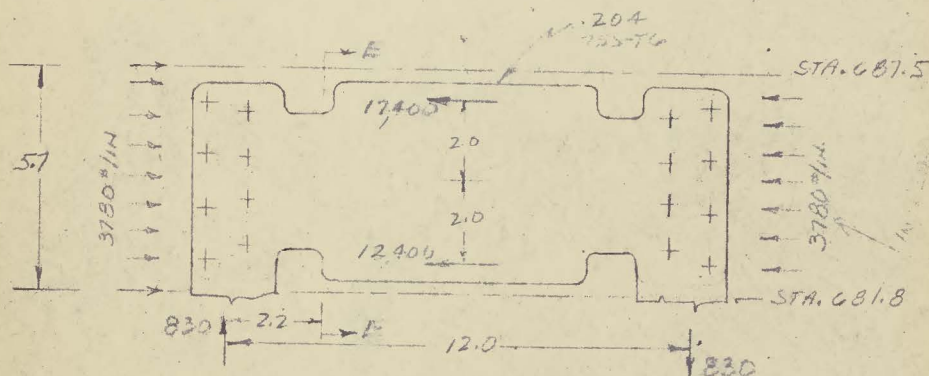
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UPPER SURFACE PLATES.

INNER PLATE STA 681.81 - 687.83



CRITICAL CASE IS R.P.O.

$$\text{SKIN END LOAD} = \frac{2}{3} \text{ SYMM. CASE} = \frac{2}{3} 11,330 = 7560 \text{ #/IN. (PG. 3.14)}$$

$$\text{LOAD ON PLATE} = \frac{7560}{2} = 3,780 \text{ #/IN.}$$

$$\text{DIAPH. 687.5 LOAD} = \frac{67500}{4} = 17,400 \text{ # (PG. 2.05)}$$

$$\text{DIAPH. 681.8 LOAD} = \frac{49200}{4} = 12,400 \text{ # (PG. 2.05)}$$

ASSUME MOMENT ON PLATE IS REACTED BY COUPLE LOADS AT ATTACHMENTS AS SHOWN IN SKETCH.

$$\text{COUPLE LOADS} = 2 (17,400 - 12,400) / 12 = 830 \text{ LBS.}$$

SEC. E-E STRESSES

$$F/C = .275 \quad \text{MOMENT} = 2.2 \times 830 = 1,830 \text{ IN. LBS.}$$

$$K = 1.5$$

$$\text{ALLOW. MOMENT} = 1.5 \times 67,000 \times .275 = 27,000 \text{ IN. LBS.}$$

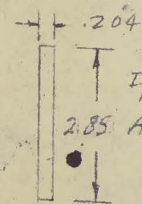
$$\text{END LOAD} = 5.7 \times 3780 + (17,400 + 12,400) / 2 = 34,500 \text{ LBS.}$$

$$f_c = 34,500 / .58 = 63,000 \text{ PSI.}$$

$$R_o = \frac{1830}{27,000} = .06 ; R_c = \frac{63,000}{67,000} = .94$$

$$M.S. = \frac{1}{(.06 + .94)} - 1 =$$

.90 M.S.



SEC. E-E

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UPPER SURFACE PLATES

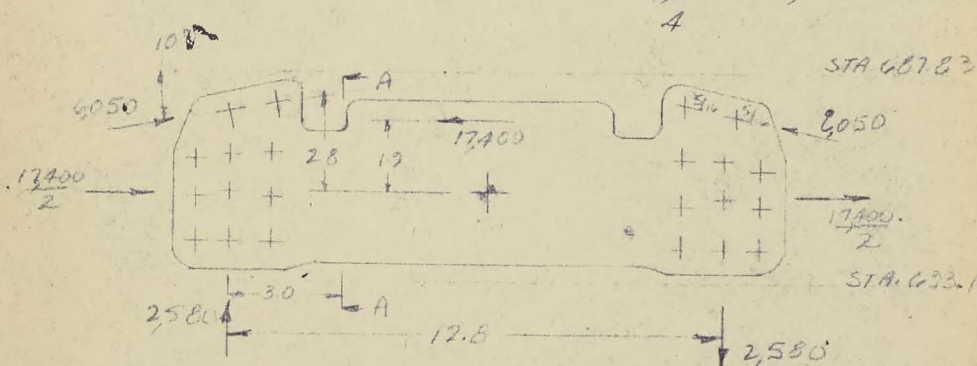
INNER PLATES. STA. 687.83 - STA. 711.6

STA. 687.83 - STA. 693.1 PLATE

PLATE IS LOADED BY STA. 687.83 DIAPH. & BY WING SKIN END LOAD.

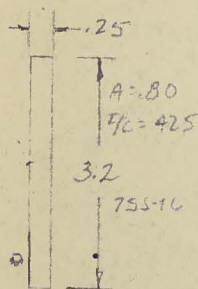
LOAD FROM WING SKIN = 4050* (PG 3.19)

LOAD FROM DIAPH. 687.83 = $\frac{42500}{4} = 10625$ LBS. (PG. 2.05)



ASSUME MOM. IT ON PLATE IS REACTED BY COUPLE LOADS AS SHOWN IN SKETCH.

COUPLE LOADS = $\frac{1.0 \times 17400}{12.8} = 1359.4$ LBS.



SEC. A-A STRESS

MOMENT = $3.0 \times 2580 + 4050 \cos 10^\circ \times 2.8$
= $7750 + 11650 = 19400$ IN. LBS.

END LOAD = $\frac{17400}{2} + 4050 \cos 10^\circ$
= $8700 + 5950 = 14650$ LBS.

ALLOW. MOMENT = $1.5 \times .93 \times 72000 \times .425 = 42600$ IN. LBS.

$f_c = \frac{14650}{.80} = 18312.5$ PSI

$R_B = \frac{24200}{42600} = .57$

$R_C = \frac{18312.5}{.93 \times 72000} = .27$

$M.S. = \frac{1}{(.57 + .27)} - 1 =$

.19 M.S.

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UPPER SURFACE PLATES

OUTER PLATE STA 66 - STA 742.5

STRESSES @ SEC. A-A

$$f_c = \text{NET COMPRESSIVE STRESS SEC. A-A} = \frac{5800 \times 6.07}{1.25 (6.07 - 2.85)} = 43700 \text{ PSI.}$$

$$\text{NET SHEAR SEC. A-A} = \frac{2005 \times 6.07}{(6.07 - 2.85)} = 4740 \text{ #/IN.}$$

BENDING STRESSES @ SEC. A-A

$$\text{TOTAL SHEAR} = 4740 \times 3.22 = 15300 \text{ #}$$

$$\text{MOMENT} = \frac{15000 \times 1.2}{2} = 9200 \text{ IN. LBS.}$$

$$I/c (A-A) = \frac{3.22^2 \times 1.25}{6} = .43$$

$$\text{ALLOW. MOMENT} = 1.5 \times .43 \times 66000 = 42600 \text{ IN. LBS.}$$

$$R_c = \frac{43700}{66000} = .66$$

$$R_s = \frac{9200}{42600} = .22$$

$$M.S. = \frac{1}{(.66 + .22)} - 1 = .74 \text{ M.S.}$$

STRESSES @ c

$$f_c = \frac{5800}{1.25} = 23000 \text{ PSI.}$$

$$f_s = \frac{6000}{1.25} = 24,000 \text{ PSI.}$$

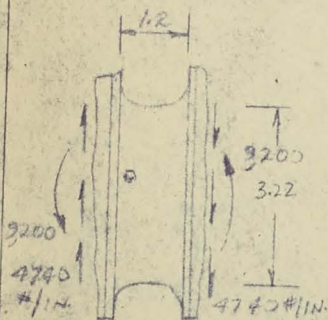
$$R_c = \frac{23000}{66000} = .35$$

$$R_s = \frac{24000}{44000} = .55$$

$$R_s^2 + R_c^2 = 1$$

$$M.S. =$$

$$.54 \text{ M.S.}$$



VIEW B

(PG. 3.24)

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LOWER SURFACE PLATE

INNER PLATE STA. 681.87 - STA. 682.53

RND CASE

WING SKIN LOAD PER IN. = $\frac{2}{3}$ 34 MM. FLIGHT

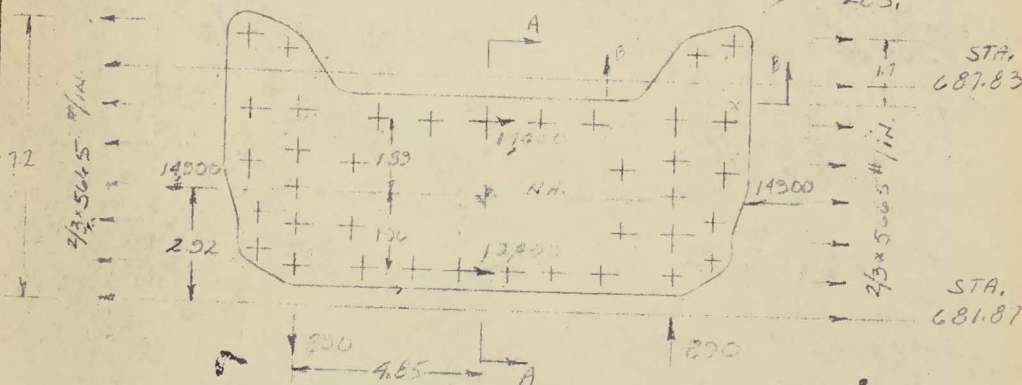
= $\frac{2}{3}$ 11330 #/IN. (PG. 2.04)

LOAD FROM STA. 681.83 DIAPH. = $\frac{67500}{4}$ (PG. 2.05)

= 16875 LBS

LOAD FROM STA. 681.87 DIAPH. = $\frac{49700}{4}$ (PG. 2.05)

12400 LBS.



STRESSES AT SEC. A-A

MOMENT (P. 1) = $\frac{2}{3} \times 5665 \times \frac{72 - 2.32}{2} + 800 \times 4.85$

= 2575 + 4520 = 7095 IN. LBS.

END LOAD = $\frac{2}{3} 5665 \times 7.2 = 27200$ LBS.

$S_b = \frac{685 \times 2.5}{1.6} = 10800$ PSI.

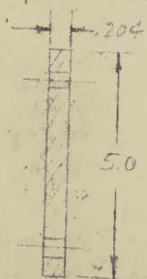
$f_c = 27200 \times 0.232 = 6300$ PSI

$\pm$ TOTAL = 30800 + 10800 = 41600 PSI.

$F_c = 66000$

M.S. = $\frac{41600}{1.6} = 26000$

58 M.S.



SEC. A-A

I = 1.6

A = 382



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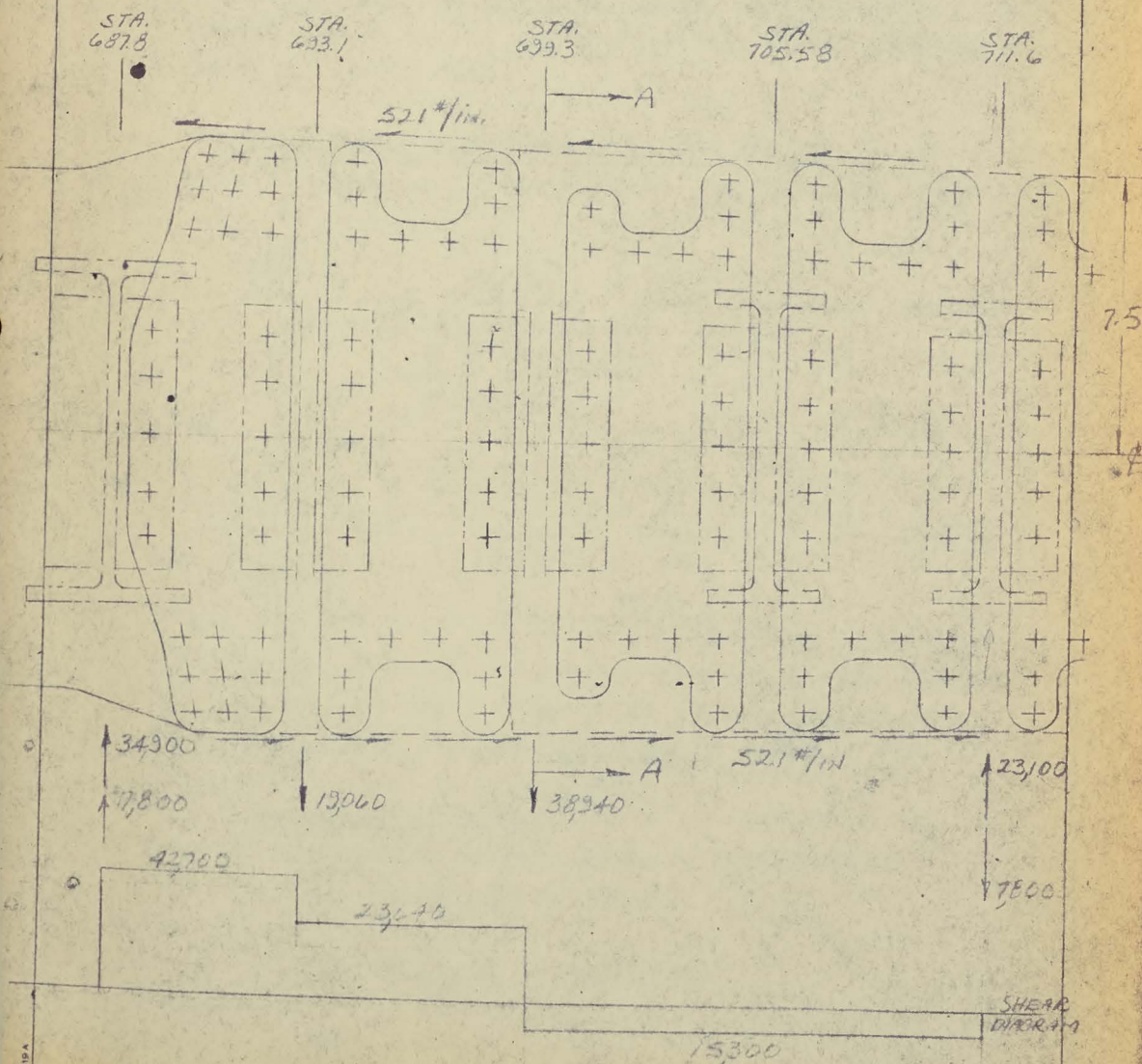
LOWER SURFACE PLATES

INNER PLATES - STA. 687.83 - STA. 711.6

LOADS ON INNER & OUTER PLATES

LOCAL SHEAR FROM JACK LOADS REF. PG.

PRIMARY SHEAR = 521 #/IN. (REF. PG.)

PRIMARY SHEAR LOAD @ STA. 711.6 = $521 \times 15 = 7800 \#$ 

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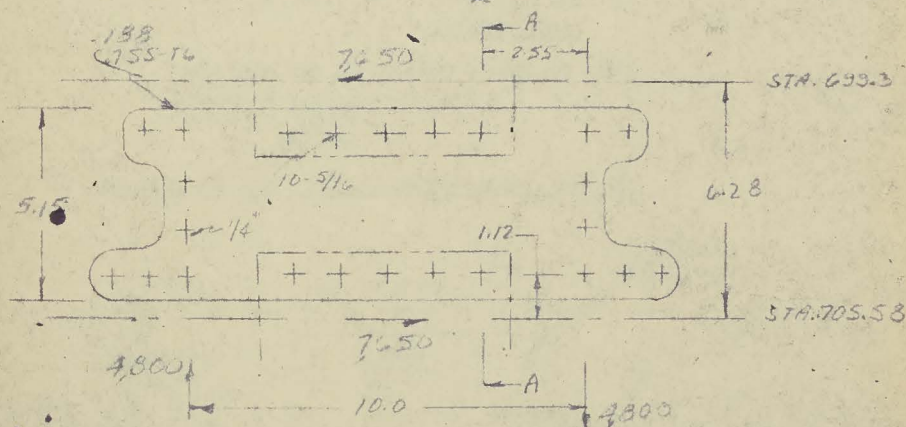
LOWER SURFACE PLATES

INNER PLATES STA. 687.83--STA 711.6

IF THE LOWER SURFACE FIN BOX STRUCTURE IS TO BE CAPABLE OF RECEIVING LOADS BETWEEN STA. 487.83 & STA. 711.6, THE SERIES OF PLATES BETWEEN DIAPHRAGMS MUST ACT AS AN INTEGRAL WEB; CONSEQUENTLY, THE PLATE ATTACHMENTS TO DIAPHRAGMS MUST TRANSMIT BEAM SHEAR LOADS.

INNER PLATE STA. 633.3 - STA. 705.58

SHEAR ON PLATE = $\frac{15000}{2} = 7500$ LBS. (Pg. 32)



ASSUME MOMENT ON PLATE IS REACTED BY COUPLE
LOADS AT ROW OF $1/4"$ BOLTS SHOWN IN SKETCH.

COURT 10925 = $\frac{76.50 \times 6.25}{10} = 4,800 \text{ LBS.}$

BENDING SEC. A-A

$$\text{MOMENT} = 2.55 \times 4800 = 12,250 \text{ IN. LBS.}$$

$$I/c = 5.5^2 \times 100 = .83$$

$$f_b = \frac{13250}{1.53} = 19000 \text{ psi}$$

NOT CRITICAL



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LOWER SURFACE PLATES

INNER PLATE STA. 699.3 - STA. 705.58

ATTACHMENT TO DIAPHRAGM

SHEAR ON ATTACHMENTS = 7650 LBS. (PG. 330)

MOMENT = $1/2 \times 7650 = 3825$ IN. LBS.

$I_F = 14.2$

LOAD ON CRIP. BOLT FROM MOMENT = $\frac{3825 \times 2.38}{14.2}$

1430 LBS.

LOAD PER BOLT FROM SHEAR = $\frac{7650}{2} = 3825$ LBS.

RESULTANT = $\left(\frac{1430^2 + 3825^2}{2} \right)^{1/2}$

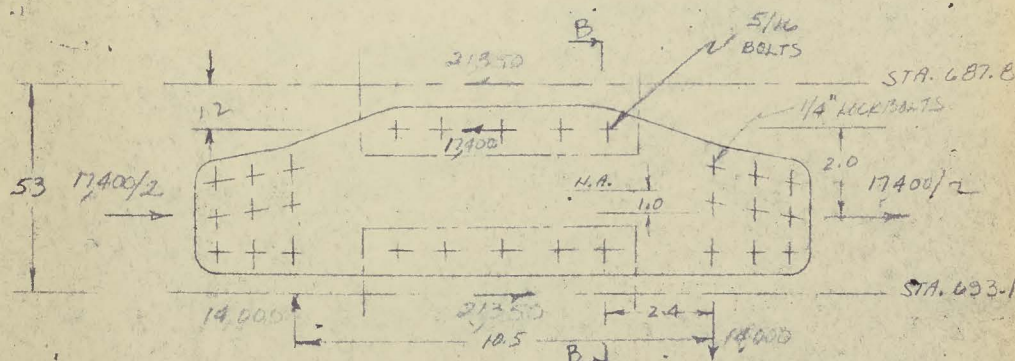
= 2090 LBS.

ALLOWABLE LOAD = 6600 LBS.

M.S. = $\frac{6600}{2090} = 3.16$

HIGH M.S.

INNER PLATE STA. 687.8 - STA. 693.1



SHEAR LOAD ON PLATE = $\frac{43700}{2} = 21,850$ LBS. (PG. 329)

LOAD FROM STA. 687.8 DIAPH. = $\frac{69500}{4} = 17,375$ LBS. (PG. 205)

ASSUME MOMENT ON PLATE IS REACTED BY COUPLE LOADS ON INNER ROW OF $1/4$ " BOLTS AS SHOWN IN SKETCH.

COUPLE LOADS = $21,850 \times 5.3 + 17,375 \times 2.0$

10.5

= 14,000 LBS.



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LOWER SURFACE PLATESINNER PLATE STA. 687.83 - STA. 693.11BENDING SEC. B-B (PG. 331)

$$\begin{aligned} \text{MOMENT} &= 2.4 \times 14,000 + 1.0 \times 17,400/2 \\ &= 33,600 + 8,700 \\ &= 42,300 \text{ IN. LBS.} \end{aligned}$$

$$I/c = \frac{42,300}{6} = .60$$

$$S_b = \frac{42,300}{.60} = 70,500 \text{ PSI.}$$

$$\text{END LOAD} = 17,400/2 = 8,700 \text{ LBS.}$$

$$f_c = \frac{8,700}{.204 \times 4} = 10,700 \text{ PSI.}$$

$$K = 1.5$$

$$F_c = 72,000 \text{ PSI.}$$

$$\begin{aligned} \text{ALLOW. MOMENT} &= 1.5 \times .60 \times 72,000 \\ &= 64,800 \text{ IN. LBS.} \end{aligned}$$

$$R_b = \frac{42,300}{64,800} = .65$$

$$R_c = \frac{10,700}{72,000} = .15$$

$$M.S. = \frac{1}{(.65 + .15)} - 1 =$$

.25 M.S.

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SEC. 3 - ECCENTRIC LOADING DUE TO BEND IN PLATES

DUE TO THE WING SURFACES JOINING THE
FIN BOX AT A SLIGHT ANGLE, AN ECCENTRIC
LOAD IS INDUCED IN THE FIN BOX PLATES.
THE EFFECT OF THIS CONDITION IS CONSIDERED
IN THE ANALYSIS THAT FOLLOWS.

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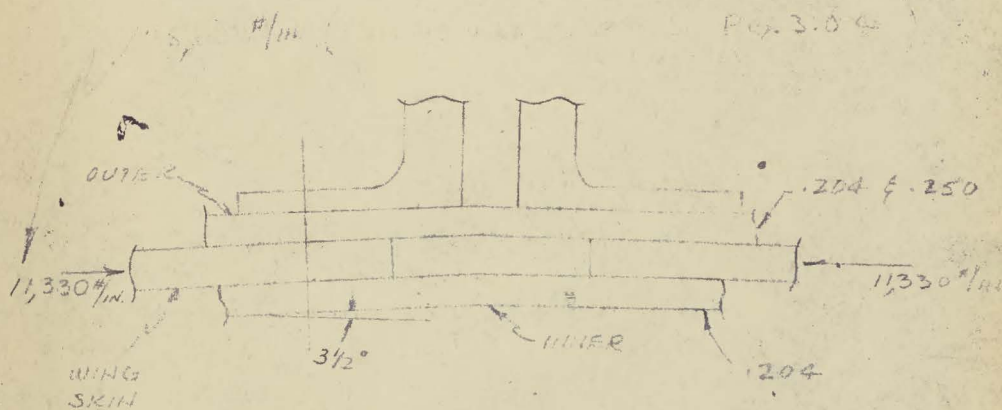
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WING SKIN FINISH 10% ELECTRICITY
UPPER SURFACE STA. 607.02 - STA. 612.52



MAX. WING SKIN LOAD = 11,330 #/in. (FIG. 3.04)

NET LOAD PER IN. IN SINGLE SPlice PLATE BETWEEN

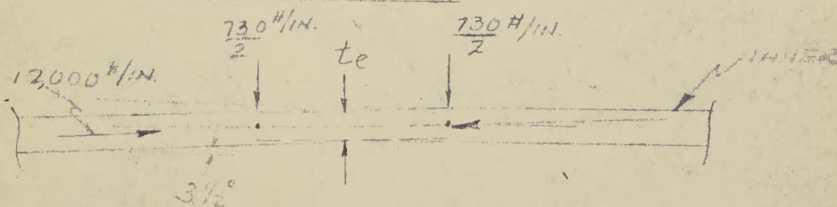
SLOTS IN PLATE = $\frac{5.7 \times 11,330}{2.7 \times 2} = 12,000 \text{ #/in.}$

VERT. COMPONENT DUE TO ANGULARITY OF PLATE

AT FIN BOX = $12,000 \sin 3 \frac{1}{2}^\circ$

= 730 #/in.

STRESS IN INNER PLATE



ASSUME COMPONENT IS DIRECTED AS SHOWN IN
SKETCH, THEN, BY USING AN EFFECTIVE THICKNESS
ACROSS JOINT IN PLATE, LOAD PATH CAN BE ASSUMED
TO BE IN A STRAIGHT LINE.

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WING SKIN FINISH JOINT ECCENTRICITY
UPPER SURFACE STA. 711.6 - STA. 742.5

EFFECTIVE THICKNESS OF PLATE = $t_e = .150$ IN.

AREA LWR. PLATE @ STA. 736 = $2.0 \times .150$
= .30

$f_t = \frac{15,000}{.30} = 50,000$ PSI.

$F_t = .93 \times 72,000 = 67,000$

$M.S. = \frac{67,000}{50,000} - 1 = .34$

.34 M.S.

LOWER SURFACE STA. 711.6 - STA. 742.5

NET AREA OF LOWER PLATES IS GREATER
THAN UPPER SURFACE PLATES, BECAUSE
THEY DO NOT CONTAIN SLOTS; THEREFORE,
STRUCTURE CAN BE SUBSTANTIATED BY
COMPARING TO UPPER SURFACE (STA. 711.6
TO STA. 742.5)

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TECHNICAL DEPARTMENT (Aircraft)

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LONGITUDINAL ANGLES & TEES
DESCRIPTION

THE FIN BOX LONGITUDINAL ANGLES & TEES RUN IN A FORE & AFT DIRECTION & RELATE TO THE FIN BOX STATIONED AT THE INTERSECTION OF THE VERT. & HORIZ. PLATES. THEY PERFORM THE FOLLOWING FUNCTIONS.

1) ALL ANGLES & TEES BETWEEN STA. 607 & STA. 742.5 SERVE TO ABORT THE VERT. COMPONENT OF WING SKIN BENDING INDUCED BY BEND IN FIN BOX HORIZ. PLATES.

2) OUTER ANGLES ON THE UPPER SURFACE PLATES AT INTERSECTION OF FIN BOX ACT TO TRANSMIT FIN BOX BENDING LOADS FROM THE FIN SKIN TO THE FIN BOX UPPER PLATES.

3) ANGLES & TEES BETWEEN STA. 624 & STA. 687, IN ADDITION TO (1) ABOVE, CARRY LOADS INDUCED BY FIN BOX BENDING.

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT:

C-105

REPORT NO.

SHEET NO. 403

PREPARED BY

DATE

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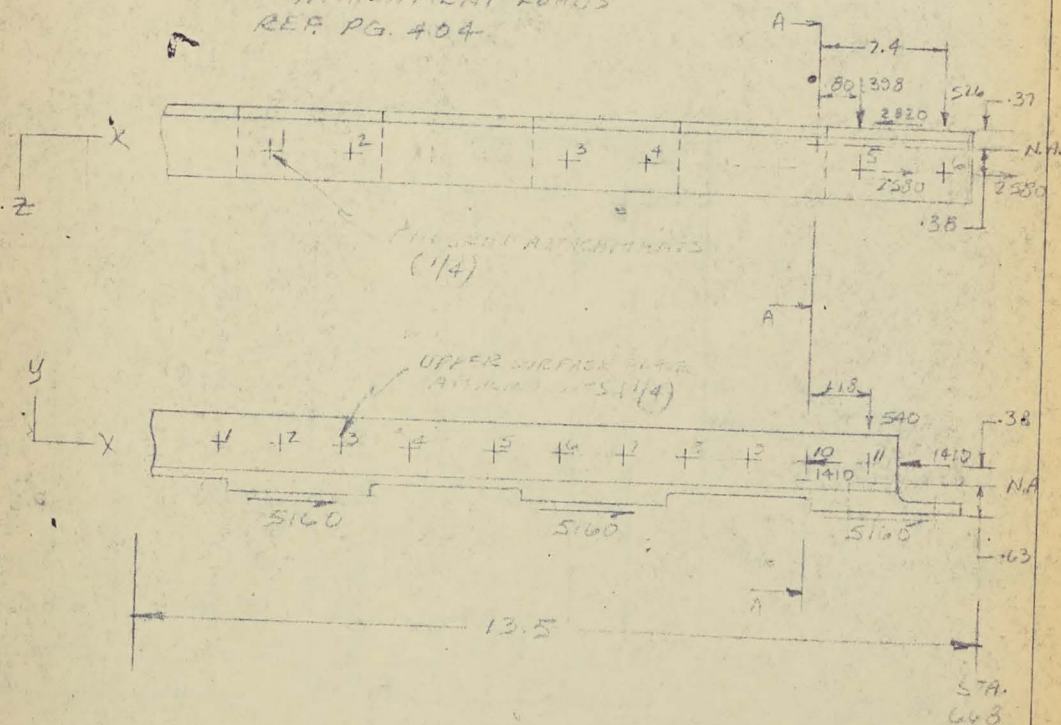
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FIN BOX LONGITUDINAL ANGLES STEES

STA 627 - STA. 642 - UPPER SURFACE - OUTER ANGLE

FOR ATTACHMENT LOADS
REF. PG. 404



$$\begin{aligned}
 M_y(A-A) &= 2 \times 2580 \times 38 + 520 \times 37 - 2.4 \times 520 - .80 \times 398 \\
 &= 1960 + 1040 - 1260 - 320 \\
 &= 1420 \text{ IN. LBS.}
 \end{aligned}$$

$$\begin{aligned}
 M_z(A-A) &= 5160 \times 6.5 + 2 \times 1410 \times 38 - 1.18 \times 540 \\
 &= 3260 + 1070 - 640 \\
 &= 3690 \text{ IN. LBS.}
 \end{aligned}$$

$$\begin{aligned}
 \text{END LOAD} &= 5160 - 2 \times 1410 \\
 &= 5160 - 2820 = 2340 \# (\text{TENS.})
 \end{aligned}$$

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FIN BOX LONGITUDINAL ANGLES & TENS

STA. 607 - STA. 647 - UPPER SURFACE - OUTER ANGLE
ATTACHMENT TO FIN BOX HORIZ. PLATES

SHEAR PER ANGLE = $2300 \times 13.5 / 2 = 15500 \#$

MOMENT = $1.7 \times 15500 = 15500 \text{ IN. LBS.}$

$M/I = 15500 / 176.4 = 88$

(SEE SKETCH PG. 413)

	A	X	A · X ²	LOAD FROM MOMENT $\frac{M \cdot X \cdot A}{I}$	LOAD FROM SHEAR $\frac{15500}{11}$
1	1.0	-6.33	40.	556	1410
2		-5.13	26.4	451	
3		-3.54	15.6	347	
4		-2.25	7.0	234	
5		-1.04	1.1	92	
6		.21		19	
7		1.42	2.0	175	
8		2.61	6.8	230	
9		3.81	14.6	335	
10		4.98	25.	440	
11	1.0	6.16	37.9	540	1410
Σ	11.0		176.4		

MAX. BOLT LOAD =
 $(17.3 + 55.2) \frac{1}{2} = 1510 \#$

ALLOW. LOAD = 4650#

M.S. = $\frac{4650}{1510} = \underline{\underline{HUGH 1.5}}$

ATTACHMENT TO FIN SKIN

MOMENT = $.70 \times 15500 = 10900 \text{ IN. LBS.}$

$\frac{M}{I} = \frac{10900}{134.3} = 81$

$\frac{M}{I} = 81$

(SEE SKETCH PG. 413)

	A	X	A · X ²	LOAD FROM MOMENT $\frac{M \cdot X \cdot A}{I}$	LOAD FROM SHEAR $\frac{15500}{6}$
1	1.0	6.50	42.3	526	2580
2		4.55	24.5	400	
3		.83	.7	67	
4		.72	.5	58	
5		4.06	24.5	398	
6	1.0	6.50	42.3	526	2580
Σ					

MAX. BOLT LOAD =
 $(52.6 + 25.8) \frac{1}{2} = 2640 \#$

ALLOW. LOAD = 4650#

M.S. = $\frac{4650}{2640} = \underline{\underline{HUGH 1.5}}$

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

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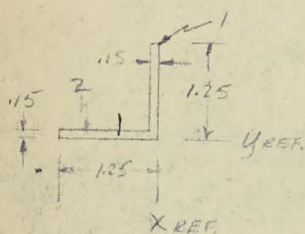
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FIN BOX LONGITUDINAL ANGLES & TEES

STA. 607-STA. 609 - UPPER SURFACE - OUTER ANGLE

SEC. H-A (PG. 403)

SECTION PROPERTIES



	A	Z	AZ	AZ ²	I _o
1	.157	.625	.1170	.073	.025
2	.145	.075	.0124	.0069	
Σ	.352	.7	.1294	.0799	.025

$$I_y = I_x = .0799 + .025 = .1049$$

$$= .0339 - .048 = .051$$

SEC. A-A

(PG. 403)

STRESSES

$$f_{bz} = \frac{3600 \times .88}{.051} = 63,600 \text{ PSI. (COMPRESSION)}$$

$$f_{by} = \frac{1420 \times .37}{.051} = 10,300 \text{ PSI. (COMPRESSION)}$$

$$f_t = \frac{2340}{.332} = 6,450 \text{ PSI. (TENS.)}$$

$$f_{(TOTAL)} = 63,600 + 10,300 - 6,450$$

$$= 67,250 \text{ PSI.}$$

$$f_c = 73,000 \text{ PSI.}$$

$$M.S. = \frac{73,000}{67,250} - 1 =$$

.16 M.S.



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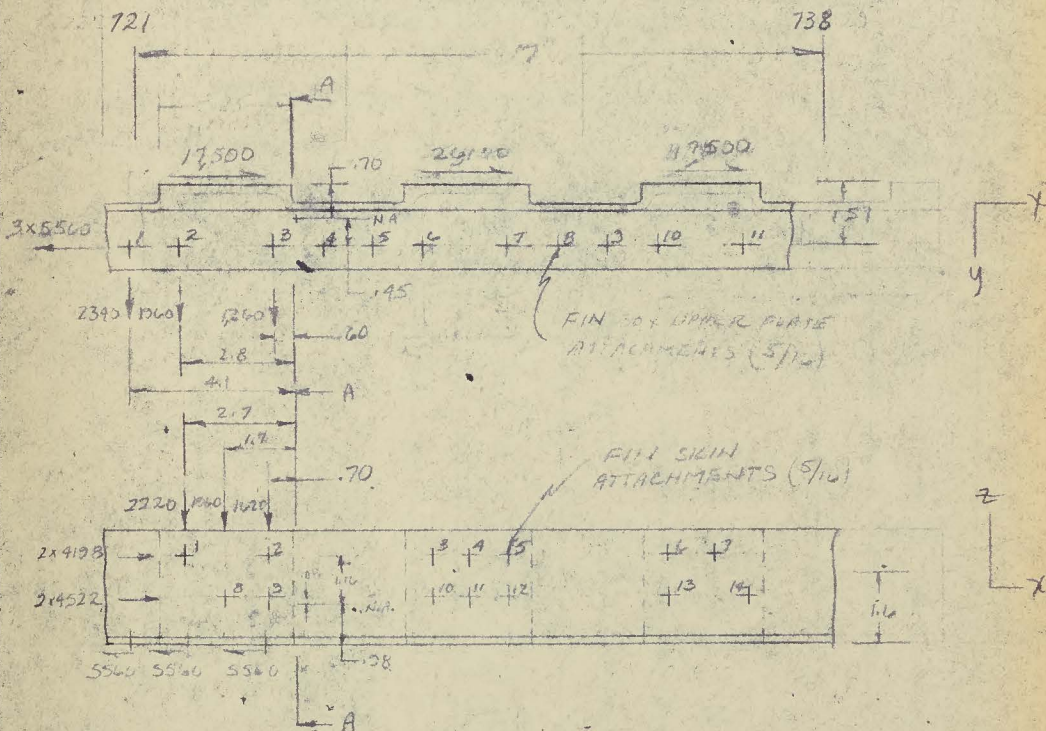
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FIN BOX LONGITUDINAL ANGLES & TEE'S

STA. 669 - STA. 742.5 - UPPER SURFACE - OUTER ANGLE

(FOR ATTACHMENT LOADS REF. PGS 4.07 & 4.08)



$$M_y (A-A) = 3 \times 5560 \times 70 + 2 \times 4522 \times 108 + 2 \times 4198 \times 116 - 2.7 \times 2220 - 1.7 \times 1960 - 70 \times 1260$$

$$= 16,700 + 720 + 9740 - 6,000 - 3,320 - 1,130$$

$$= 18,710 \text{ IN. LBS.}$$

$$M_z (A-A) = 17,500 \times 70 + 3 \times 5560 \times 45 - 4.1 \times 2340 - 2.8 \times 1960 - 60 \times 1260$$

$$= 12,250 + 7500 - 3700 - 5500 - 750$$

$$= 3,800 \text{ IN. LBS.}$$

$$\text{END LOAD} = 17,500 - 3 \times 5560$$

$$= 17,500 - 16,700 = 800 \text{ * (compression)}$$

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

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AIRCRAFT:

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FIN BOX LONGITUDINAL ANGLES & STRESSES

STA. 663 - STA. 792 - UPPER SURFACE - OUTER ANGLE
ATTACHMENT TO FIN SKIN

$$MOMENT = 1.6 \times 21100 = 37800 \text{ IN. LBS.}$$

$$M/I = 37800 / 371.8 = 101.7$$

(SEE SKIN 4.06)

	A	Z	X	A·Z ²	A·X ²	DIST FROM MOMENT		LOAD FROM SHEET 2 41100 14	TOTAL HORIZ.
						HORIZ.	VERT.		
						$\frac{M \cdot Z \cdot A}{I}$	$\frac{M \cdot X \cdot A}{I}$		
1	1.0	.51	-7.0	.26	49.0	162	2220	4360	4108
2	↑	↑	-5.09	↑	26.0	↑	1020	↑	↑
3			-1.03		1.1		330		
4			-.02		—		—		
5			1.0		1.0		320		
6			5.07		26.8		1010		
7			7.32		53.8	162	2320		4108
8			-6.17		38.8	-162	1300		4522
9			-5.03		26.0	↑	1020		↑
10			-1.03		1.1		330		
11			-.02		—		—		
12			1.0		1.0		320		
13	↓	↓	5.07	↓	25.8	↓	1010	↓	↓
14	1.0	.51	7.32	.26	53.8	-162	2320	4360	4522
Σ	14.0			3.64	304.2				

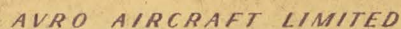
$$I_x = 304.2 + 3.6 = 307.8$$

$$MAX. BOLT LOAD = \left(\frac{2320 + 4522}{2} \right)^{1/2} = 5050^*$$

$$ALLOW. LOAD = 7300$$

$$M.S. = \frac{7300}{5050} = 1.44$$

1.44 M.S.



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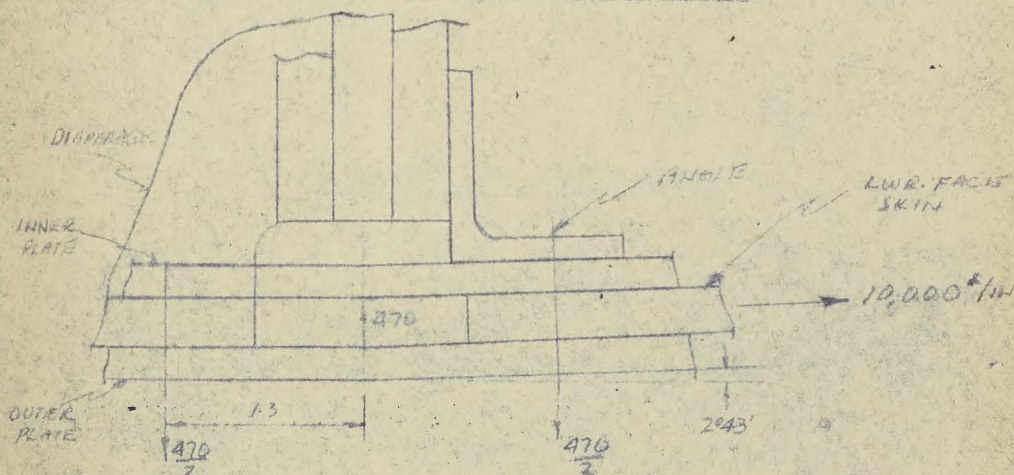
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FIN BOX LONGITUDINAL ANGLES & TEES

• STA 711.6 - STA 742.5 LWR SURFACE



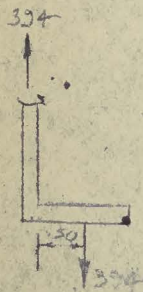
MAX. DIAPHRAGM LENGTH = 60,000" (PG. 2.05)

DIAPH. SPACING $\frac{1}{2}$ " APPROX.

$$\text{SKIN END LOAD/IN.} = \frac{60000}{6} = 10,000 \text{ #/IN.}$$

DUE TO ANGLE/ARITY BETWEEN FIN/BOY LWR.
SURFACE CONTOUR & WING SKIN CONTOUR, A
VERTICAL COMPONENT IS PRODUCED AT INTERSECTION
OF THE TWO SURFACES.

$$\begin{aligned}\text{VERT. COMPONENT} &= 10,000 \sin 2^\circ 43' \\ &= 470 \text{ \#/IN.}\end{aligned}$$



AS AN INITIAL CHECK, ASSUME TOTAL MONTHS
REACHED BY BARRY, THEN,

$$(F/c) = \frac{.75 \times 1.0}{6} = .125$$

RAILWAY MOMENT = $66,000 \times 1.5 \times .0037 = 366 \text{ in. lbs.}$

$$M.S. = \frac{366}{235} = 1.5574$$

5517.5

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FITTINGS

DESCRIPTION

THE FITTINGS DESCRIBED BELOW ARE
SUBSTANTIATED BY THE ANALYSIS THAT FOLLOWS.

FUS. VERT. STRUT PICKUP - STA. 663.47

THE VERT. STRUT FITTING IS ATTACHED
TO THE FIN BOX LWR. SURFACE BETWEEN
DIAPHRAGMS @ STA. 657.94 & STA. 663.70. IT
REACTS VERT. LOAD FROM THE FUS. STRUT @
STA. 663.47.

FUS. VERT. STRUT PICKUP - STA. 714.85

THE VERT. STRUT FITTING IS ATTACHED
TO THE FIN BOX LWR. SURFACE BETWEEN
DIAPHRAGMS @ STA. 711.6 & STA. 717.67. IT
REACTS VERT. LOAD FROM THE FUS. STRUT @
STA. 714.85.

FIN R/S SPR. ATTACHMENT FITTING - STA. 736.03

THE R/S ATTACHMENT FITTING
SERVES TO TRANSMIT SHEAR FROM THE
FIN R/S TO THE FIN BOX UPPER SURFACE
STRUCTURE.

FOR ANALYSIS OF FIN SHEAR FITTINGS
AT STAS. 646.62, 671.43, 696.23 & 726.03
REFER TO . FOR ANALYSIS OF
CENTER ENGINE THRUST MOUNT FITTING
STA. 624, REFER TO 7/0562/25.

X

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT:

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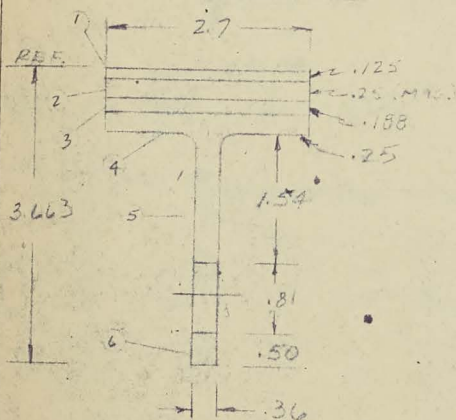
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FUR. VERT. STRUT PICKUP STA. 663.47

SECTION 4 PROPERTIES

SEC. A-A (PG. 5.03)



	AREA	Σ	AZ	AZ^2	I_0
1	.338	.062	.021	.001	
2	.438	.250	.109	.027	
3	.518	.469	.243	.114	
4	.675	.688	.464	.318	
5	.555	1.583	.879	1.390	.110
6	.180	3.413	.615	2.100	
Σ	2.704	$\Sigma = .86$	2.331	3.950	.110

$$I = 3.550 + .110 - .8 \times 2.331$$
$$= 4.060 - 2.0$$
$$= 2.06$$

SEC. B-B (PG. 5.03)

	A	Z	AZ	AZ ²	T ₀
1	.333		.021	.001	
2					
3	.513		.243	.114	
4	.675		.464	.318	
5	.248	.1276	.316	.404	.013
Σ	1.779	Z = .59	1.044	.837	.013

$$I = .937 + .013 - .59 \times 1.044$$

$$.850 - .616 = .234$$

VQ/T SHEAR ON THE ATTACHMENT TO PLATE

$$V = 7250 \text{ LBS.}$$

$\phi = .86$

$$VQ/I = 7350 \times .86 / 2.06 = 3,300 \text{ #/in.}$$

2-rows @ 1.25 each

SHEAR LOAD PER BOLT = $\frac{3300 \times 125}{2} = 2060 \#$

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

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AIRCRAFT:

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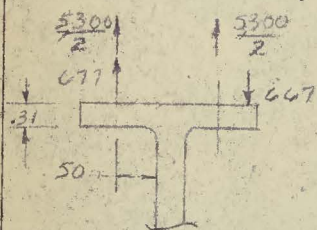
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FUS. VERT. STRUT PICK-UP - STA. 663.47
FLANGE BENDING



$$\text{MOMENT} = \left(\frac{5300}{2} + 6.77 \right) \cdot 50$$

$$= 1660 \text{ IN. LBS.}$$

ASSUME 2" LE. GTH OF FLG. EFFECTIVE

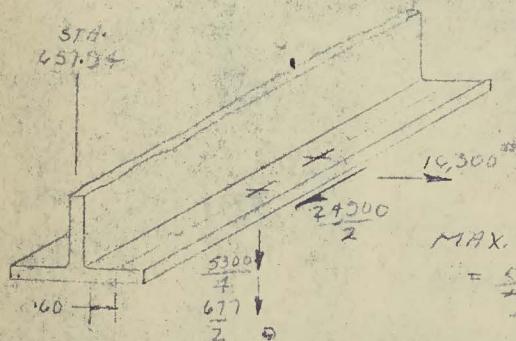
$$I/c = \frac{.31^2 \times 2.0}{6} = .032$$

$$\text{ALLOW. MOMENT} = 1.5 \times .032 \times 66,000$$

$$= 3300 \text{ IN. LBS.}$$

$$M.S. = \frac{3300}{1660} - 1 = .99 \text{ M.S.}$$

DIAPHRAGM FLANGE STRESSES (STA. 657.34)



FLANGE BENDING STRESS

$$\text{MAX. BOLT TENSION LOAD}$$

$$= \frac{5300}{2} + 6.77 = 1660 \#$$

$$\text{MAX. MOMENT FROM BOLT LOAD}$$

$$= .60 \times 1660 = 995 \text{ IN. LBS.}$$

ASSUME 1.5 IN. OF FLG. EFFECTIVE IN BENDING MOMENT, THEN

$$I/c = \frac{.25^3 \times 1.5}{6} = .0156$$

$$\text{ALLOW. MOMENT} = 66,000 \times .0156 \times 1.5$$

$$= 1,540 \text{ IN. LBS.}$$

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FUS VERT. STRUT PICK-UP - STA. 663.47

DIAPHRAGM FLANGE STRESSES

SHEAR STRESS

DIAPH. FLANGE LOAD 29,800# (PG. 2105)

$$\begin{aligned} \text{MAX. SHEAR STRESS} &= \frac{29,800}{2(7.8 \times .25)} \times \frac{3}{2} \\ &= 36,000 \text{ PSI.} \end{aligned}$$

$$F_s = 45,000 \text{ PSI.}$$

TENSION STRESS

FOR CASE AND TENSION STRESS IN FOL BOX LOWER SURFACE USE STRESS @ STA. 664. UPPER SURFACE:

$$\text{TENSION STRESS FOR CASE AND} = 6,670 \text{ PSI. (PG. 6.28)}$$

$$\begin{aligned} \text{LOAD IN INNER &\& \text{ OUTER PLATES OVER LENGTH} \\ \text{OF DIAPHRAGM} &= 6,670(188 + .125)7.8 \\ &= 10,300 \text{ LBS.} \end{aligned}$$

ASSUME ENTIRE LOAD IS CARRIED IN DIAPH. FLANGE, THEN

$$S_s = \frac{10,300}{7.8 \times .25} = 8,400 \text{ PSI.}$$

MARGIN OF SAFETY

$$R_D = 395 / 1540 = .65$$

$$R_s = 364 / 45,000 = .21$$

$$R_t = 6,400 / 66,000 = .13$$

$$M.S. = \frac{1}{[(.65 + .13) + .21]^{1/2}} - 1 =$$

.22 M.S.

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FUS. VERT. STRUT PICK-UP - STA. 663.47

ATTACHMENT TO DIAPHRAGM FLANGE (1/4" JO BOLTS)

CRITICAL ATTACHMENTS ARE @ STA. 663.

LOADS ON ATTACHMENTS

MAX. TENSION LOAD/BOLT FROM STRUT FITTING

$$= \frac{4100 + 600}{4}$$

$$= 1175 + 300 = 1475 \#$$

DIAPHRAGM LOAD, CASE 2RD. = 45300# (PG. 205)

SHEAR PER BOLT PER SHEAR FACE = $\frac{45300}{2 \times 12}$

$$= 1910 \#$$

LOAD FROM BENDING OF FIN BOX = 14300# (PG. 508)

SHEAR PER BOLT PER SHEAR FACE = $\frac{14300}{2 \times 6}$

$$= 1360 \#$$

RESULTANT SHEAR = $(1910^2 + 1360^2)^{1/2}$

$$= 2340 \#$$

MARGIN OF SAFETY

$$R_s = 2340 / 4100 = .57$$

$$R_t = 1475 / 2570 = .57$$

$$M.S. = \frac{1}{(.57 + .57)^{1/2}} - 1 =$$

24 M.S.

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FUS. STRUT PICK-UP - STA. 714.85

DIAPHRAGM FLG. BENDING (SEC. C-C) PG. 5.11

MOMENT ON FLG. = $.55 \times 5360 = 2940$ IN. LBS.

ASSUME 2" OF FLG. EFFECTIVE

$I/C = .31 \times 20 / 2 = .032$

MOM. MOMENT = $1.5 \times 71000 \times .032 = 3860$ IN. LBS.

SHEAR STRESSES IN DIAPH. FLG. DUE TO FIN

WILL BE COMPARED WITH BENDING STRESSES

DIAPH. FLANGE $4490 = 52,200$ (PG. 2.05)



FLG. SHEAR STRESS = $\frac{52,000 \times 2}{2(51 \times 31)} = 29,600$ PSI.

$F_s = 45,000$ PSI.

MARG. OF SAFETY

$R_D = 2940 / 3860 = .76$

$R_s = 29,600 / 45,000 = .55$

$M.S. = \frac{1}{(.76 + .55)^{1/2}} - 1 = .07$ M.S.

ATTACHMENTS (5/16 BLIND LOCKBOLTS)

BOLTS ARE HELD BY TENSION LOAD FROM FITTING & SHEAR LOAD FROM DIAPHRAGM

ASSUME 4-BOLTS EFFECTIVE IN CARRYING TENSION LOAD.

TENSION LOAD PER BOLT = $10,000 / 4 = 2,500$ LBS.

SHEAR LOAD PER BOLT FACE = $\frac{32,000}{215} = 3,250$ LBS.

DIAPH. TENSION LOAD = $3,275$

ALLOW. STRESS = $.31 \times 31 \times 128,000 / 2 = 6,150$ LBS.

$R_s = 3,275 / 6,150 = .53$

$R_t = 2,500 / 3,275 = .77$

$M.S. = \frac{1}{(.53 + .77)^{1/2}} - 1 = .07$ M.S.



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FIN ATTACHMENT
TO FUSELAGE

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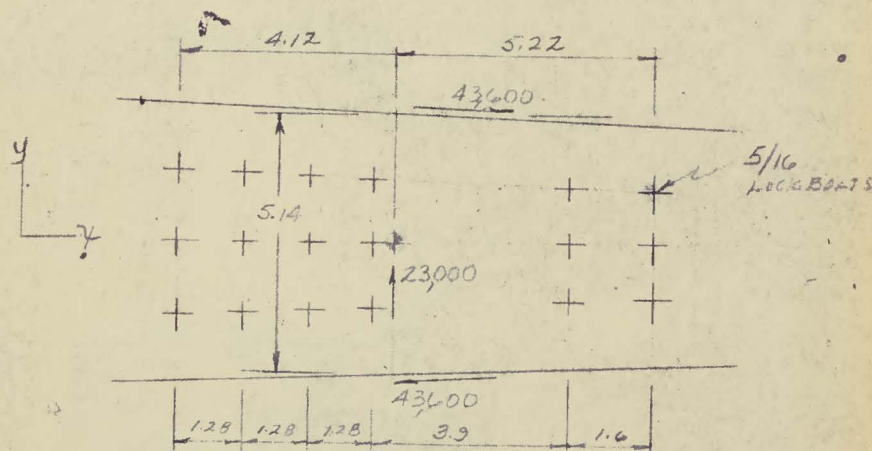
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FIN REAR SPAR ATTACHMENT FITTING
ATTACHMENT TO UPPER SURFACE

FIN SKIN SHEAR = 7500 #/IN. (PG. 1.03)

$$\text{SHEAR LOAD} = \frac{7500 \times 11.6}{2} = 43,600 \text{ LBS.}$$

SIDE SHEAR FROM FIN SPAR = 23,000 LBS (7/05/52/16, PG. 1.08)

$$\text{MOMENT ON ATTACHMENTS} = 43,600 \times 5.14 = 224,000 \text{ IN. LBS.}$$

$$I_p = 221.3$$

LOAD ON CRIT. BOLT FROM MOMENT

$$\text{"Y" COMPONENT} = \frac{224,000 \times 5.22}{221.3} = 5,300 \text{ LBS.}$$

$$\text{"X" COMPONENT} = \frac{224,000 \times 1.02}{221.3} = 1,030 \text{ LBS.}$$

LOAD PER BOLT FROM SIDE SHEAR = 23,000 / 18 = 1,280 LBS.

$$\text{RESULTANT LOAD} = \left[(5,300 + 1,280)^2 + 1,030^2 \right]^{1/2}$$

$$= 6,650 \text{ LBS.}$$

ALLOW. LOAD PER BOLT = 7,300 LBS.

$$\text{M.S.} = \frac{7,300}{6,650} - 1 =$$

10 M.S.

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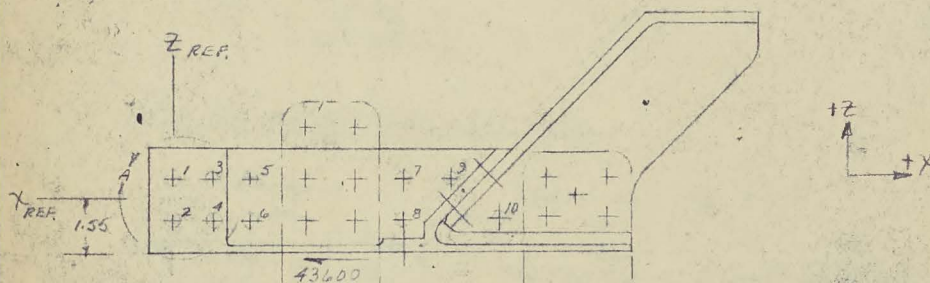
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FIN ATTACHMENT
TO FUSELAGE

FIN REAR SPAR ATTACHMENT FITTING
ATTACHMENT TO FIN SKIN



$$\text{MOMENT @ CTRD} = 1.55 \times 43,600 \\ = 67,500 \text{ IN. LBS.}$$

Bolt	X	Z	$Y_{NA} = \bar{Y} - X$	Y_{NA}^2	Z^2	VERT. COMP. $M/I \cdot Y_{NA}$	HORR. COMP. $M/I \cdot Z + \Delta P_x$	RESULTANT $(\frac{-Y^2}{VERT} + \frac{-Z^2}{HORR})^{1/2}$
1	0	.52	3.36	11.3	.27	2450	-3380	4650
2	0	-	3.36	11.3		2450	-4740	5250
3	1.05		2.31	5.3		1680	-3380	4160
4	1.05	-	2.31	5.3		1680	-4740	5030
5	2.02		1.34	1.8		980	-3380	4080
6	2.02	-	1.34	1.8		980	-4740	4830
7	5.98		-2.62	6.9		-1910	-3380	4400
8	5.98	-	-2.62	6.9		-1910	-4740	5100
9	7.20		-3.84	14.8		-2800	-3380	4850
10	8.31	-.52	-4.95	24.5	.27	-3610	-4740	5950
Σ	33.61			89.9	2.7			

$$\bar{X} = 33.61/10 = 3.36$$

$$I_{NA} = \Sigma Y_{NA}^2 + \Sigma Z^2 = 89.9 + 2.7 = 92.6$$

$$M/I = 67500/92.6 = 730$$

$$\Delta P_x = -43650/10 = -4360$$

A.V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT

C-105

FIN ATTACHMENT
TO FUSELAGE

REPORT NO. _____

SHEET NO. 516 36

PREPARED BY:

MCCABE

DATE:

3-4-55

CHECKED BY:

DATE:

FIN REAR SPAR ATTACHMENT FITTING
ATTACHMENT TO FIN SKIN (CONT'D)

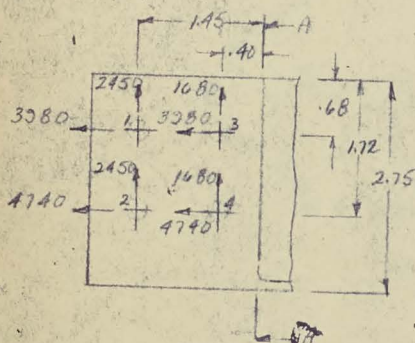
CRITICAL BOLT #10 = 5,950 LBS. (PG. 515)

ALLOW. LOAD $5/16$ LOCKBOLT = 7300 LBS

$$M.S. = \frac{7300}{5950} - 1 =$$

.23 M.S.

BENDING OF FITTING VERT. LEG



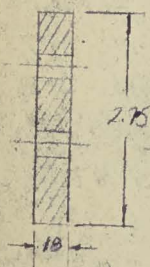
LOADS REF. PG. 515

$$\begin{aligned} \text{MOMENT (A-A)} &= 2(1680 \times .40 + 2450 \times 1.45) \\ &+ 2 \times 4740 \times .34 - 2 \times 3380 \times .69 \\ &= 2(672 + 3550 + 1610 - 2790) \\ &= 6180 \text{ IN. LBS.} \end{aligned}$$

$$S_b = 6180 / .228 = 27000 \text{ PSI}$$

$$f_c = \frac{2(4740 + 3380)}{.36} = 48500 \text{ PSI}$$

DETAIL A (PG. 515)



$I/c = .228$

$A = .36$

SEC. A-A

$M/I = 108,000$ REF. AERO. DESIGN MANUAL

ALLOW. MOM. = $108,000 \times .228 = 24,600$

$$R_b = 6180 / 24600 = .25$$

$$F_t = 73000$$

$$R_t = 48500 / 73000 = .63$$

$$M.S. = \frac{1}{(.25 + .63)} - 1 =$$

.114 M.S.

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO.

SHEET NO

518

AIRCRAFT:

C-105

PREPARED BY

DATE

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FIN REAR SPAR ATTACHMENT FITTING

MOMENT OF INERTIA OF FLANGE BOLTS 1-7
ABOUT X-AXIS (REF SKETCH PAGE 7)

	A	y	y ²	Ay ²
1	1.0	2.4	5.76	5.76
2	1.0	2.4	5.76	5.76
3	.65	2.4	5.76	3.74
4	.65	2.4	5.76	3.74
5	1.00	1.8	3.24	3.24
6	1.00	1.8	3.24	3.24
7	1.00	1.8	3.24	3.24
Σ	6.3			28.72

$$I = 2 \times 28.72 = 57.5$$

LOADS - FLANGE BOLTS

$$\text{MOMENT} = 5123,000 = 115,000 \text{ IN. LBS}$$

$$M/I = 115,000 / 57.5 = 2000 \text{ LBS}$$

	A	y	LOAD = M/A	
1	1.0	2.4	4800	
2	1.0	2.4	4800	
3	.65	2.4	3120	
4	.65	2.4	3120	Σ(1-4) = 15,840 LBS.
5	1.00	1.8	3600	
6	1.00	1.8	3600	
7	1.00	1.8	3600	Σ(5-7) = 10,800 LBS.

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT:

C-105

REPORT No.

SHEET No. 5.19

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1-13-56

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FIN REAR SPAR ATTACHMENT FITTING
FLANGE ATTACHMENT LOADS

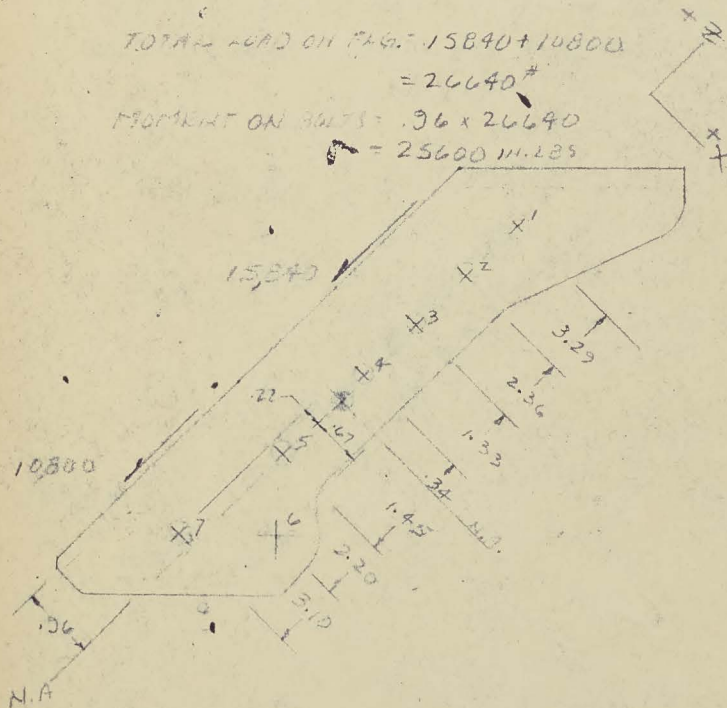
TOTAL LOAD ON FLG. = 15840 + 10800

= 26640#

MOMENT ON BOLTS = .96 x 26640

= 25600 IN. LBS

(REF PG. 5.18)



1	2	3	4	5	6	7	8	9	10	11
	A	Z	X	AZ ²	AX ²	LOAD COMPONENTS HORIZ. VERT.	SHORE VERT. =	TOTAL VERT. =	RESULT	
						M.B.H E.B.H	M.X.A E.X.A	(PG 5.18)	8 + 9	(7 + 10) / 2
1	1.0	3.29	.05	1080	-	-2440	-	-4800	-4800	5560
2	1.0	2.36		558	-	-1750	-	-4800	-4800	5100
3	.65	1.33		1.15	-	-640	-	-3120	-3120	3150
4	.65	.34		.08	-	-163	-	-3120	-3120	3120
5	1.0	1.45	.05	2.10	-	1070	-	-3600	-3600	3760
6	1.0	2.20	.67	485	.45	1630	430	-3600	-3110	3500
7	1.0	3.10	-.22	9.60	.05	2230	-163	-3600	-3763	4400
Σ	6.3			34.16	.50					

I = 34.16 + .50 = 34.66

M = 26640 = 740

I = 34.66

BULGE CORRECTION

M = 7300 - 1

5560

3177.5

A. V. ROE CANADA LIMITED
MALTON, ONTARIO
TECHNICAL DEPARTMENT (Aircraft)

REPORT No. _____

SHEET No. 522

AIRCRAFT:

C-105

PREPARED BY

DATE

Mc CABE

1-13-56

CHECKED BY

DATE

FIN REAR SPAR ATTACHMENT FITTING

SEC. B-3 STRESSES (PG 5.21)

$$\text{MOMENT} = 4.8 \times 15,800 - 2.8 \times 23,000 \\ = 76,000 - 65,000 = 11,000 \text{ IN. LBS.}$$

$$I/c = \frac{4.3 \times .25}{6} = .776$$

$$S_b = \frac{11,000}{.776} = 14,176 \text{ PSI. (NOT CONT.)}$$

SEC B-10

$$S_s = \frac{23,000}{4.3 \times .25} = 21,400 \text{ PSI.}$$

$$F_s = 45,000$$

$$M.S. = \frac{45,000}{21,400} - 1$$

HIGH M.S.



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT:

C-105

REPORT NO.

SHEET NO. C.03

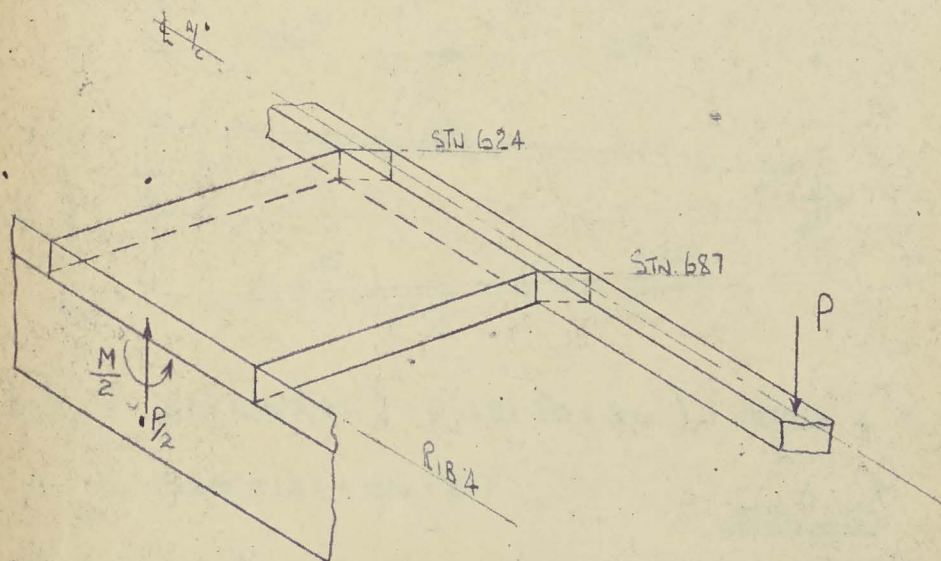
PREPARED BY

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DATE

WING INERTIA LOADS - FIN BOX STRUCTURE - STA 624 - STA 687



THE ASSUMED WING TORQUE BOX FOR DISTRIBUTION OF LOAD P FROM STRUCTURE AFT OF REAR SPAR IS LOCATED BETWEEN STATIONS 624 AND 687, THE CENTRE REAR SPAR AND REAR SPAR RESPECTIVELY. THE LOAD P AND MOMENT M ARE REACTED HALF TO EITHER SIDE OF A/C. HENCE ONLY ONE HALF WILL BE CONSIDERED.

AS THIS ANALYSIS IS PURELY TO DETERMINE THE APPLICATION OF P AND M TO THE FIN BOX AND THE IMMEDIATE WING BOX REACTIONS, NO ACCOUNT WILL BE TAKEN OF THE SLOPE OF THE WING BOX AND ITS VARIATION IN SECTION CONSTANTS BETWEEN THE FIN BOX AND RIB 4.



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TECHNICAL DEPARTMENT (Aircraft)

REPORT NO.

SHEET NO. 604

AIRCRAFT:

C-105

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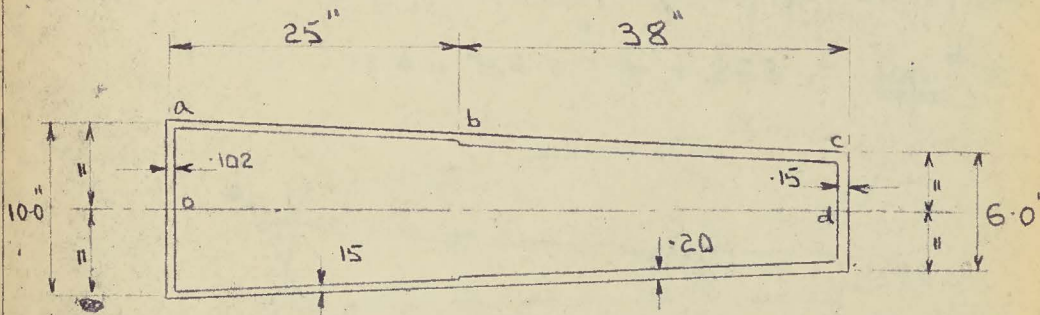
P. McGenzie

DATE

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AIR & WEIGHT LOADS FIN BOX STRUCTURE - STA 674 STA 687
SECTION THROUGH WING TORQUE BOX AT ITS JOINT
WITH FIN BOX



$$I_{od} = 2(25 \times 15 \times 4.6^2) + 2(38 \times 20 \times 3.6^2) + \frac{102 \times 10^3}{12} + \frac{15 \times 6^3}{12}$$

$$= 1585 + 197 + 8.5 + 2.7 = \underline{\underline{366.7 \text{ ins}^4}}$$

FOR $V = 1000 \#$

$$\frac{V}{I} = \frac{1000}{366.7} = \underline{\underline{2.73}}$$

FLEXURAL SHEAR FLOW

ASSUME $q_a = 0$

$$q_a = 0 + \frac{V}{I} \sum_0^a Z A$$

$$q_a = 0 + (2.73 \times 102 \times \frac{10}{2} \times \frac{10}{4}) = \underline{\underline{4.0 \#/\text{in}}}$$

$$q_b = 4.0 + (2.73 \times 15 \times 25 \times 4.6) = \underline{\underline{51.0 \#/\text{in}}}$$

$$q_c = 51 + (2.73 \times 20 \times 38 \times 3.6) = \underline{\underline{125.6 \#/\text{in}}}$$

$$q_d = 125.6 + (2.73 \times 15 \times \frac{6}{2} \times \frac{6}{4}) = \underline{\underline{127.45 \#/\text{in}}}$$



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TECHNICAL DEPARTMENT (Aircraft)

REPORT NO.

SHEET NO. 605

AIRCRAFT:

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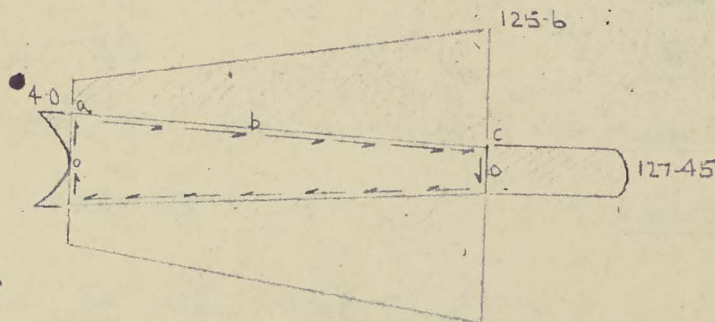
DATE

AIR. INERTIA LOAD ~ FIN BOX STRUCTURE - STAG 24 - STAG 87

CHARACTER:

$$\sum Vz = 1000 = -\left(\frac{4 \times 10}{3}\right) + (125.6 \times 6) + \left(\frac{2 \times 1.85 \times 6}{3}\right) + (2 \times 0.31 \times 64.8 \times 63)$$

$$= -13.4 + 754 + 7.4 + 253 = \underline{1001} \# \checkmark$$



TOTAL SHEAR STRAIN DUE TO THIS SHEAR FLOW.

TAKING G AS CONSTANT AND EQUAL TO UNITY

$$\sum \sigma = \sum \frac{qL}{t}$$

$$= 2 \left[\frac{4 \times 5}{3 \times 102} + \frac{28 \times 25}{15} + \frac{88 \times 38}{20} + \frac{125.6 \times 3}{15} + \frac{1.85 \times 3 \times 2}{3 \times 15} \right]$$

$$= 2 [65.4 + 4670 + 16710 + 2512 + 23] = \underline{47960}$$

FOR CONDITION OF NO TWIST A CONSTANT SHEAR FLOW $q \#/\text{in}$ WHICH WILL PRODUCE A SHEAR STRAIN OF -47960 MUST BE APPLIED TO BOX. VIZ.

$$\sum \sigma_L = 2 \left[\frac{q \times 5}{102} + \frac{q \times 25}{15} + \frac{q \times 38}{20} + \frac{q \times 3}{15} \right] = -47960$$

$$\therefore q = \frac{-47960}{812.2} = \underline{\underline{-59.0 \#/\text{in}}}$$



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. _____

SHEET NO. 6-06

PREPARED BY

DATE

P. M. Morgan

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AIRCRAFT:

C-105AIR & INERTIA LOADS - FIN BOX STRUCTURE - STA 624 - STA 632
RESULTANT SHEAR FLOW

$$q_D = 0 - 59 = -59.0 \text{ #/"}$$

$$q_{u'} = 4 - 59 = -55.0 \text{ #/"}$$

$$q_b = 51 - 59 = -8.0 \text{ #/"}$$

$$q_L = 125.6 - 59 = +66.6 \text{ #/"}$$

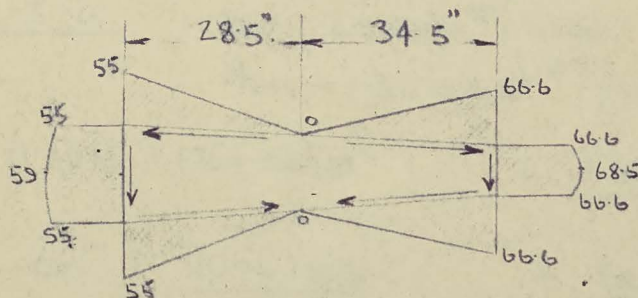
$$q_d = 127.45 - 59 = +68.5 \text{ #/"}$$

FOR LOCATION OF ZERO SHEAR FLOW.

$$\left(\frac{4.0 + 121.6x}{63} \right) = 59$$

$$x = \frac{63 \times (59 - 4)}{121.6} = 28.5"$$

FLEXURAL SHEAR FLOW FOR 1000 # APPLIED AT SHEAR CENTRE



$$\text{SHEAR CENTRE} = \frac{\sum M_o}{\sum V} \quad \text{MOMENTS ABOUT O}$$

$$= \frac{67.8 \times 6 \times 63 + 66.6 \times 34.5 \times 5 - 55 \times 28.5 \times 5}{1000}$$

$$= \frac{29250}{1000} = 29.25"$$



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT No.

SHEET No. 607

AIRCRAFT:

C-105

PREPARED BY

DATE

P. McGowan

CHECKED BY

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AIR & INERTIA LOADS - FIN BOX STRUCTURE - STA 624 - STA 657
CHECK ON TOTAL SHEAR

$$\begin{aligned}\Sigma V &= 1000 = (678 \times 6) + (58 \times 10) + (.0318 \times 66.6 \times 345) - (.0318 \times 54 \times 285) \\ &= 409 + 580 + 64.7 - 56 = \underline{997.7} \# \checkmark\end{aligned}$$

FUSELAGE STATION AT SHEAR CENTRE

$$x = 624 + 2925 = \underline{653.25}$$

FROM REPORT NO.

PP

THE SHEAR AND MOMENT FOR VARIOUS CASES ARE
QUOTED AT STATION 655. THEREFORE THE MOMENT
AT STATION 650.9

$$= M_{655} + 4.1 V_{655}$$

CASE 11-3-a

$$V_{655} = 20000 \# / \text{SIDE}$$

$$M_{655} = 1.3075 \times 10^6 \#'' / \text{SIDE}$$

$$M_{SC} = (1.3075 + 1.75 \times .02) 10^6 = \underline{13425 \times 10^6}$$

$$\text{AREA OF BOX} = \frac{(10+6)63}{2} = \underline{504}''^2$$

$$\text{BATHO SHEAR FLOW } q = \frac{T}{2A}$$

$$= \frac{1.3425 \times 10^6}{2 \times 504} = \underline{1335} \# / \text{IN}$$



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 113

SHEET NO. 6.08

AIRCRAFT:

C-105

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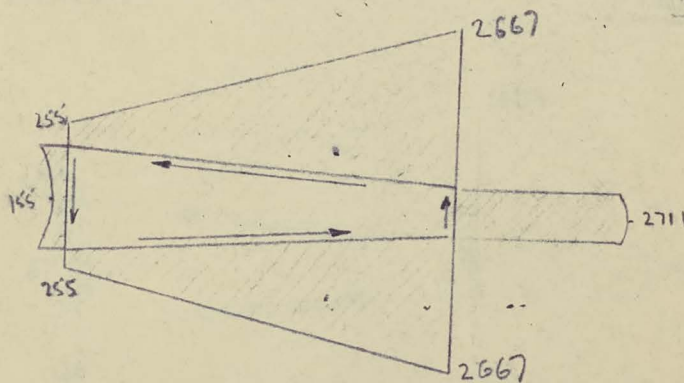
P. M. Jensen

CHECKED BY

DATE

AIRCRAFT INERTIA LOADS - FIN BODY STRUCTURE STA 624 - STA 637
 TOTAL SHEAR FLOW - CASE 11-3-a

$$\begin{aligned}
 q_a &= 59 \times 20 - 1335 &= -155 \text{ #/in} \\
 q_b &= 54 \times 20 - 1335 &= -255 \text{ #/in} \\
 q_c &= -666 \times 20 - 1335 &= -2667 \text{ #/in} \\
 q_d &= -688 \times 20 - 1335 &= -2711 \text{ #/in}
 \end{aligned}$$



R.B.D CASE 22

$$V_{655} = 13000 \text{ #/in} \text{ / SIDE}$$

$$M_{655} = 8106 \times 10^6 \text{ #/in} \text{ / SIDE}$$

$$M_{s.c} = (-8106 + 175 \times 0.13) 10^6 = 8639 \times 10^6 \text{ #/in}$$

$$\begin{aligned}
 q &= \frac{-8344 \times 10^6}{1008} &= 828 \text{ #/in}
 \end{aligned}$$



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. _____

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AIRCRAFT:

C-105

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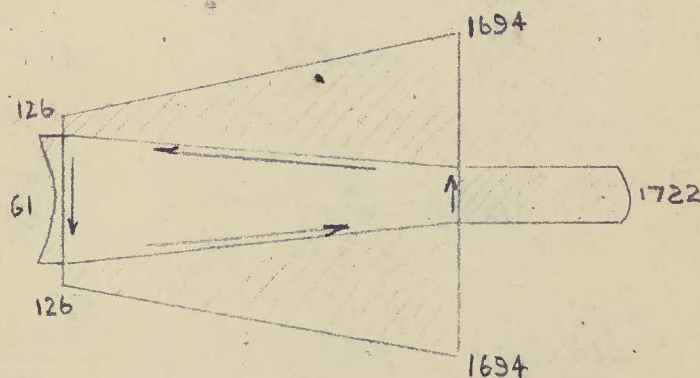
AIR F INERTIA LOADS - FIN BOX STRUCT. - STA 624 - STAGE
TOTAL SHEAR FLOW RPD CASE 22

$$q_D = 59 \times 13 - 828 = -61.0 \text{ #/"}$$

$$q_A = 54 \times 13 - 828 = -126.0 \text{ #/"}$$

$$q_C = -666 \times 13 - 828 = -1694.0 \text{ #/"}$$

$$q_D = -688 \times 13 - 828 = -1722.0 \text{ #/"}$$



DISTRIBUTION OF SHEAR AND MOMENT TO FIN BOX

SHEAR AND MOMENT IS APPLIED TO THE FIN BOX BY THE INFINITELY STIFF FIN SKIN AND IS REACTED BY THE ELASTIC TORSION BOX WHICH HAS A VARYING FLEXURAL STIFFNESS BETWEEN SPARS. FOR THE PURPOSE OF MAXIMUM FINGER PLATE LOADING A TRIANGULAR STIFFNESS OF WING BOX IS ASSUMED



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO.

SHEET NO.

412

AIRCRAFT

C-105

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AIR & INERTIA LOADS - FIN BOX STRESS STA 624-STA 687
R.P. Q. CASE 22

$$\text{SHEAR AT MID BOX (Sta. 655.5)} = 13000 \# / \text{SIDE}$$

$$\text{MOMENT " " " " } = (8106 - 0.13 \times 5) 10^6 = 8041 \# \times 10^6$$

LOADING / INCH FROM MOMENT

$$W_M = \frac{M \times L^2 \times \alpha}{I} = \frac{8041 \times 10^6 \times 577 \times X^2}{408800} = 1135 X^2$$

LOADING / INCH FROM SHEAR

$$W_V = \frac{4VX}{L^2} = \frac{4 \times 13000 \times X}{69^2} = 10.9 X$$

1	2	3	4	5
X	X ²	1135X ²	10.9X	(3) + (4)
34.5	1190	+ 1350	376.0	+ 1726 - 974
30.5	930	+ 1055	333.0	+ 1388 - 722
25.5	650	+ 738	278.0	+ 1016 - 460
20.5	430	+ 488	224.0	+ 712 - 264
15.5	240	+ 272	169.0	+ 441 - 103
10.5	110	+ 125	115.0	+ 240 - 10
5.5	30	+ 34	60.0	+ 96 + 26

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO.

SHEET NO.

615

AIRCRAFT:

C-105

PREPARED BY

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10-5-55

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AIR & INERTIA LOADS - FIN. BOX STRUCTURE - STA. 624 - STA. 627

CASE 11 (E) 9

VERTICAL SHEAR

1.	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)
STA.	g #/IN.	d	AVG. SHEAR LOAD (20+10) / 2	LOAD VERT. COMPONENT 2 (4) 2 63	FINGER LOAD 20 IN.	LOAD @ SPARS	TOTAL LOAD = 5 + 6 + 7	SHEAR (2 + 8)
	PG. 6.14	PG. 6.14			PG. 6.14	PG. 6.14		
624.05	255				8550	-1880	6670	6670
630.23	463	5.44	1950	124	4020		4144	10810
635.73	709	5.44	3180	202	2020		2222	13032
641.17	878	5.44	4310	274	820		1034	14126
646.62	1086	5.44	5340	333	110		443	14575
652.28	1305	5.66	6760	430			430	15005
657.94	1520	5.66	8000	508	-480		28	15033
663.60	1735	5.66	9200	584	-1200		-416	14417
669.20	1965	6.10	11300	717	-3850		-3133	11284
674.01	2210	6.31	13120	836	-6330		-2034	5190
681.87	2435	5.86	13610	865	-10100		-3235	-4095
687.83	2667	5.36	15200	965	-13100	16100	3065	0

VERTICAL SHEAR LOADS ARE COMPUTED BY SUMMING
UP SHEAR FLOW VERT. COMPONENT, FINGER LOADS,
& SHEAR AT SPARS. THE FINAL SHEAR LOADS
IN COL. (9) ARE PLOTTED ON PG. 6.14

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO.

SHEET NO. 6-16

AIRCRAFT:

C-105

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10-5-55

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AIR FINESTIA LOADS FIN BOX STRUCTURE-STAG 24-STAG 27
CASE 11 (B)-4
MOMENTS

STA.	(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)
		PG. C. 14	PG. C. 24	PG. C. 15			HORIZ. SHEAR 4000	LOADS IN X Y	LOADS IN X Y	TOTAL 6 X 10
64.85				6670		0	PG. C. 15	0	0	0
67.23	5.44	3.65		10810	70200	31200	1750	18900	-19000	17400
68.595	5.44	3.31		13632	58800	35000	3180	29900	-48400	46600
69.17	5.44	2.36		14126	70300	105300	4310	38600	-87000	78300
69.62	5.84	8.42		14575	76900	242800	5340	43000	-133000	107800
69.88	5.66	8.26		15005	82500	325300	6740	53800	-188800	124500
69.84	5.26	7.00		15083	85000	410300	8000	63200	-207000	158300
69.200	5.66	7.54		14417	85100	435400	9200	70300	-321400	174000
69.270	6.10	7.16		11284	86000	583900	11300	81000	-402400	181000
69.601	6.31	6.26		5720	71100	639500	13100	83100	-433500	163400
69.87	2.78	6.38		-4045	30400	639500	13410	84300	-578400	106500
69.83	2.78	6.00		0	24100	665800	15200	91200	-603400	0

FIN BOX BENDING MOMENTS ARE COMPUTED FROM SHEAR
FLAWS & FIN BOX WINGS SHOWN IN SKETCH ON PG. C. 14
APPLIED FIN MOMENT DIAGRAM PLOTTED FROM LOADS IN CASE 10
IS ALSO INCLUDED ON PG. C. 14

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT:

C-105

REPORT NO.

SHEET NO.

617

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DATE

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10-6-55

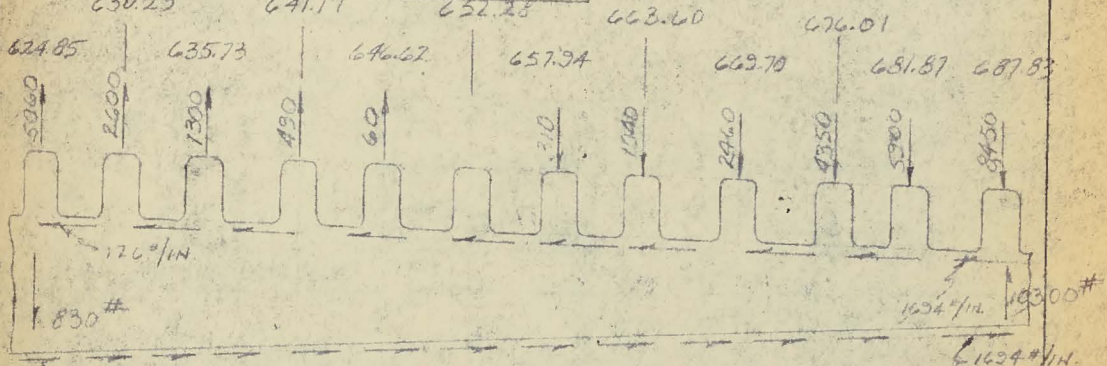
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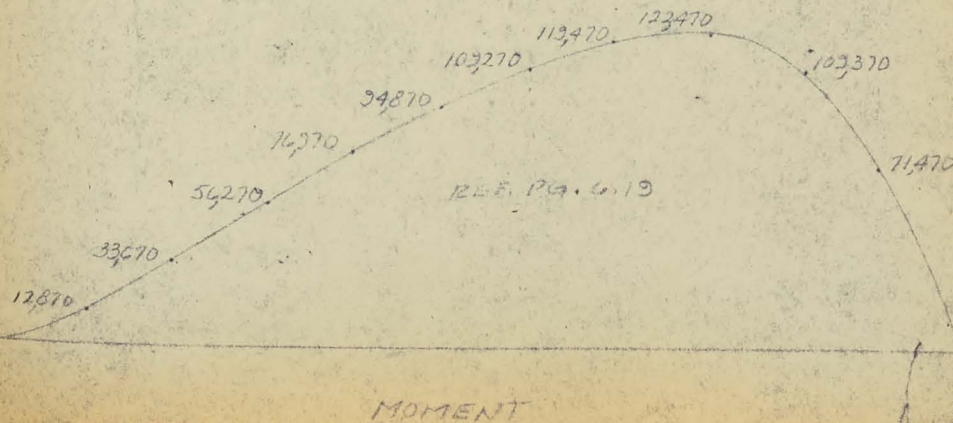
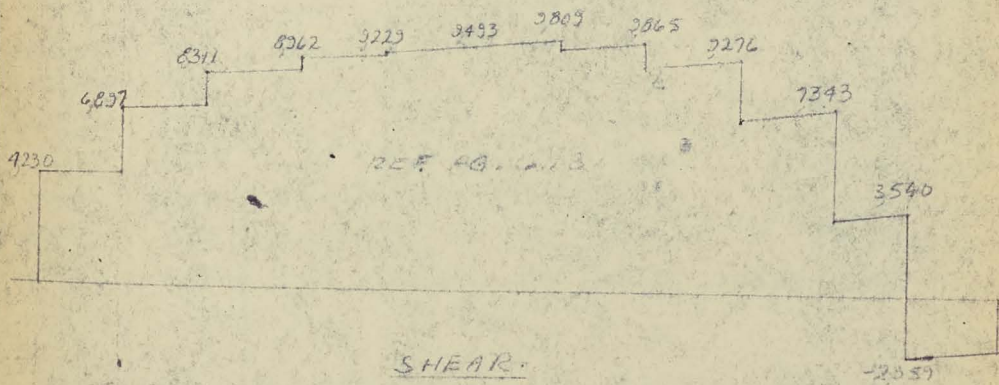
AIR & INERTIA LOADS - FIN BOX STRUCTURE - STA 624 - STA 687

CASE 22

SHEAR & MOMENT DIAGRAM



LOADS FROM CURVE PG. 6.13 ARE PLotted TO FIN BOX FINGERS. 1364 in. FOR FIN BOX SHOWN REF. PG. 6.12



TECHNICAL DEPARTMENT (Aircraft)

REPORT NO.

SHEET NO. 618

AIRCRAFT:

C-105

PREPARED BY

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10-13-55

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DATE

AIR INERTIA LOADS - FIN BOX STRUCTURE - STA. 424 - STA. 487
CASE 22
VERTICAL SHEAR

(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)
STA.	#/IN.	d	AVG SHEAR LOAD = (8190)d 2	LOAD VERT COMPONENT 2 4 2 63	FINGER LOADS	LOAD @ SPARS	TOTAL LOAD 546.17	SHEAR ± 18
	PG. 6.09	PG. 6.14			PG. 6.17	PG. 6.17		
624.85	126				5700	-830	4230	4230
630.29	261	5.44	1050	67	2600		2667	6897
635.73	327	5.44	1730	114	1300		1414	8311
641.17	533	5.44	2525	161	400		651	8962
646.62	666	5.44	3260	207	60		267	9229
652.28	808	5.66	4150	264			264	9493
657.94	949	5.66	4970	316	-310		0	9493
663.60	1090	5.66	5760	366	-1040		-674	8825
669.20	1238	6.10	7100	451	-3460		-1009	6816
676.01	1337	6.31	8360	527	-4850		-3223	2993
681.81	1546	5.86	8630	548	-5300		-5232	2359
687.83	1694	5.26	3650	614	-8450	10300	2461	0

VERTICAL SHEAR LOADS ARE COMPUTED BY SUMMING
UP SHEAR FLOW VERT. COMPONENT, FINGER LOADS,
& SHEAR AT SPARS. THE FINAL SHEAR LOADS IN
COL 9, ARE PLOTTED ON PG. 6.17

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT:

C-105

REPORT NO.

SHEET NO.

0.19

PREPARED BY

McCabe

DATE

10-13-55

CHECKED BY

DATE

AIR & INERTIA LOADS - FIN BOX STRUCTURE - STA. 624-581.87

CASE 22

MOMENTS

(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)
STA.	d	h	VERT. SHEAR	ΔM_1	$\Sigma \Delta M_1$	NOSE SHEAR LOAD	ΔM_2	$\Sigma \Delta M_2$	TOTAL MOMENT
				$d \times (4)_{n-1}$			$h \times (7)$		$6 \times (9)$
						Pg. C.18			
624.85			4230		0	Pg. C.18	0	0	0
630.29	5.44	9.65	6897	23000	23000	1050	10130	-10130	12870
635.73	5.44	9.31	8311	37500	60500	1790	16700	-24830	33670
641.17	5.44	8.96	8362	45200	105700	2525	22600	-42970	56270
646.62	5.44	8.62	9229	48900	154500	3260	28100	-77530	76970
652.28	5.66	8.26	3493	52200	206700	4150	34300	-111830	94870
657.34	5.66	7.90	3499	53600	260300	4270	37200	-151030	102270
663.60	5.66	7.54	8825	53700	314000	5760	43500	-194530	112470
669.70	6.10	7.16	6816	56800	362800	7100	50800	-245330	123470
676.01	6.31	6.76	2923	48000	410800	8300	56700	-304130	102370
681.87	5.86	6.38	-2359	12500	423300	8480	55000	-356430	71470
687.83	5.36	6.00		14050	414350	3660	55000	-414430	0

FIN BOX BENDING MOMENTS ARE COMPUTED FROM SHEAR FLOWS & FINGER LOADS SHOWN IN SKETCH ON PG. C.17
A BENDING MOMENT DIAGRAM PLOTTED FROM LOADS IN COL. (10) IS ALSO INCLUDED ON PG. C.17

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT:

C-105

REPORT NO.

SHEET No. 6.21

PREPARED BY

MC CABE

DATE

10-14-55

CHECKED BY

DATE

AIR & INERTIA LOADS - FIN BOX STRUCTURE - STA 624 - STA 687
LOWER SURFACE ATTACHMENTS
OUTER PLATE INCLUDING SKIN

CRITICAL CASE 11-B-2

MAX. SHEAR = 15617 LBS. (PG. 6.14)

VQ/I SHEAR

AT STA 669.7, SEC. A-A,
 $I = 87.4$

$$8) .526 (7.406 - 3.82) = 1.89$$

$$3) .809 (7.326 - 3.82) = 2.84$$

$$Q = 4.73$$

PG. 6.20

SHEAR PER IN. = VQ/I

$$= \frac{15617 \times 4.73}{87.4} = 850 \text{ "/IN.}$$

ATTACHMENT IS BY 2 ROWS $1/4"$ BOLTS @ $1.45"$ AVG.

$$\text{LOAD PER BOLT} = \frac{850 \times 1.45}{2} = 616 \text{ \#}$$

IN ADDITION TO LOADS FROM FIN BOX
BENDING, BOLTS ARE LOADED BY WING
SKIN END LOADS.

WING SKIN END LOAD STA. 669 = 10400 #/IN (PG. 3.04)

$$\text{LOAD PER BOLT SHEAR FACE} = \frac{10400 \times 1.45}{2 \times 2}$$

$$\text{RESULTANT LOAD} = \frac{(3770^2 + 616^2)^{1/2}}{2} = 3800 \text{ LBS.}$$

$$\text{ALLOW. LOAD } 1/4" \text{ IN. 188} = 4650 \text{ LBS.}$$

$$F.S. = \frac{4650}{3800} - 1$$

1.22 F.S.

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. _____

SHEET No. 4.22

AIRCRAFT:

C-105

PREPARED BY

DATE

McLARE

10-14-55

CHECKED BY

DATE

AIR FLEXIBILITY LOADS - FIN BOX STRUCTURE - STAG 24 STAG 27
LOWER SURFACE ATTACHMENTS
INNER PLATE TO WING SKIN

CRITICAL CASE 11.3.A

MAX. SHEAR = 15617 LBS (PG. 4.14)

VQ/I SHEAR

AT STA. 669.7, SEC. A-A,

$I = 87.4$

1) $.750(706 - 3.82) = 1.83$

3) $.546(242 - 3.82) = 2.84$

3) $.809(706 - 3.82) = 2.86$

$Q = 7.53$

PG. 4.21

SHEAR PER IN. = VQ/I

$= \frac{15617 \times 7.53}{87.4} = 1360 \text{ LBS/IN.}$

ATTACHMENT IS BY 2 ROWS $\frac{1}{4}$ " BOLTS @ 1.45" A/C.

LOAD PER BOLT = $\frac{1360 \times 1.45}{2} = 990 \text{ LBS.}$

LOAD ON BOLT FROM WING SKIN END LOAD
= 3770 LBS. (PG. 4.21)

RESULTANT = $(3770^2 + 990^2)^{1/2}$
= 3900 LBS.

ALLOW. LOAD $\frac{1}{4}$ " BOLT IN .125 = 4200 LBS.

M.S. = $\frac{4200 - 1}{3900}$

1.10175

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. _____

SHEET No. 6.23

AIRCRAFT:

C-105

PREPARED BY

DATE

MCCABE

10-14-55

CHECKED BY

DATE

AIR & INERTIA LOADS - FIN BOX STRUCTURE - STA 624 - STA 627
LOWER SURFACE ATTACHMENT
EXTRUDED TEE TO INNER PLATE

CRITICAL CASE 11.3.4

MAX. SHEAR = 15617 LBS. (PG. 6.14)

VQ/I SHEAR

AT STA 6027, SEC. A-A,

$I = 87.4$

6. .887 (6.92 - 3.82) = 2.75

1. .750 (7.06 - 3.82) = 1.83

8. .526 (7.40 - 3.82) = 2.84

9. .809 (7.30 - 3.82) = 2.86

$Q = 10.34$

PG. 6.20

SHEAR PER IN. = VQ/I

$= \frac{15617 \times 10.34}{87.4} = 1860 \text{ \#/IN.}$

87.4

SINCE TEE IS DISCONTINUED AT EACH
DIAPHRAGM, ACTUAL LOAD ON TEE WILL
BE HIGHER THAN LOAD COMPUTED ABOVE.

DIAPH. SPACING = 11.5

TEE LENGTH = 8.0

ACTUAL LOAD ON TEE = $\frac{1860 \times 11.5}{8} = 2740 \text{ \#/IN.}$

ATTACHMENT IS BY 2-ROWS $1/4"$ BOLTS @ 1.1 IN.

LOAD PER BOLT = $\frac{2740 \times 1.1}{2} = 1510 \text{ LBS.}$

ALLOW. LOAD $1/4"$ BOLT IN .125 = 4280 #

M.S. = $\frac{4280 - 1}{1510}$

HIGH M.S.

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT:

C-105

REPORT NO.

SHEET NO.

C-24

PREPARED BY

DATE

MCCABE

10-14-55

CHECKED BY

DATE

AIR INERTIAL LOADS - FIN BOX STRUCTURE - STA 24 - STA 87
LOWER SURFACE ATTACHMENTS
EXTRUDED TEE TO FIN BOX SIDE PLATES

CRITICAL CASE 11.3.4

MAX. SHEAR = 15617# (PG. 6.14)

VQ/I SHEAR

AT STA 605.7, SEC. A+A,

$I = 87.4$

6 .897 (6.92 - 3.82) = 2.75

7 .750 (7.06 - 3.82) = 2.43

8 .526 (7.406 - 3.82) = 2.84

9 .809 (7.312 - 3.82) = 2.56

12 .334 (6.410 - 3.82) = 2.42

$Q = 13.30$

PG. 6.20

SHEAR PER IN. = VQ/I

= $\frac{15617 \times 13.30}{87.4} = 2380 \text{ #/in.}$

SINCE TEE IS DISCONTINUED AT EACH DIAPHRAGM, ACTUAL LOAD ON TEE WILL BE HIGHER THAN LOAD COMPUTED ABOVE.

DIAPH. SPACING = 11.8 IN.

TEE LENGTH = 8.0 IN.

ACTUAL LOAD PER IN. = $\frac{11.8 \times 2380}{8} = 3500 \text{ #/in.}$

IN ADDITION TO LOADS FROM FIN BOX BENDING, ATTACHMENTS ARE LOADED BY VERTICAL COMPONENT OF WING SKIN KINK LOAD.

WING SKIN VERT. COMPONENT DUE TO KINK LOAD = 1200 #/in. (PG. 4.10)

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. _____

SHEET No. 6.25

AIRCRAFT:

C-105

PREPARED BY

MCCABE

DATE

10-17-55

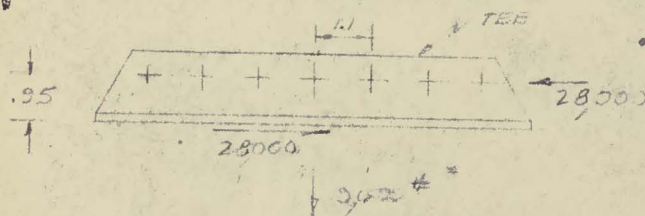
CHECKED BY

DATE

AIR INERTIA LOADS - FIN BOX STRUCTURE - STAG 24 - STAG 27
LOWER SURFACE ATTACHMENTS
EXTRUDED TEE TO FIN BOX SIDE PLATES

LOAD ON TEE FROM BENDING $8 \times 3500 = 28,000 \#$ (Pg. 6.24)

LOAD ON TEE FROM RINK $10 \times 900 = 9,000 \#$ (Pg. 6.24)



MOMENT ON ATTACHMENTS $= 28,000 \times 9.5 = 266,000 \text{ IN. LBS.}$
 $I_p = 33.8$

LOAD ON CRIT. BOLT FROM MOMENT
 $= \frac{266,000 \times 3.3}{33.8} = 2,600 \text{ LBS.}$

LOAD PER BOLT FROM SHEAR

VERT. COMPONENT $= 9,000 / 7 = 1,270 \#$

HORIZ. COMPONENT $= 28,000 / 7 = 4,000 \#$

RESULTANT $= \sqrt{4,000^2 + (1,270)^2}^{1/2}$
 $= 5,650 \text{ LBS.}$

ALLOW. LOAD $= 9,300 \text{ LBS.}$

M.S. $= \frac{9,300}{5,650} - 1 =$

.65 M.S.

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. _____

SHEET No. 6.26

AIRCRAFT:

C-105

PREPARED BY

DATE

MCCABE

10-17-55

CHECKED BY

DATE

WING INERTIA LOADS - FIN BOX STRUCTURE - STA. 624 - STAGE 7

UPPER SURFACE ATTACHMENTS

EXTRUDED ANGLES TO FIN BOX SIDE PLATES

CRITICAL CASE 11.3.9

MAY. SHEAR = 15617 LBS. PG. 6.14

VQ/I SHEAR

AT STA. 669.7, SEC. A-A,

$I = 87.4$

1 .755 (3.82 - .102) = 2.81

2 .449 (3.82 - .212) = 1.62

3 .510 (3.82 - .437) = 1.73

4 .449 (3.82 - .666) = 1.41

5 .755 (3.82 - .556) = 2.46

10 .674 (3.82 - 1.010) = 1.83

$Q = 11.91$

PG. 6.20

SHEAR PER IN ON ATTACHMENTS = VQ/I

$$= \frac{15617 \times 11.91}{87.4} = 2130 \text{ \#/IN.}$$

INNER ANGLE IS DISCONTINUED AT EACH DIAPHRAGM, CONSEQUENTLY, ACTUAL LOAD PER IN WILL BE HIGHER THAN LOAD COMPUTED ABOVE.

DIAPHRAGM SPACING = 11.8 IN.

ANGLE LENGTH = 8.0 IN.

ACTUAL LOAD PER IN. PER ANGLE = $\frac{11.8 \times 2130}{2 \times 8.0}$

$$= 1550 \text{ \#/IN.}$$

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO.

SHEET NO.

428

AIRCRAFT:

C-105

PREPARED BY

McLAREN

DATE

10-18-55

CHECKED BY

DATE

AIR & INERTIA LOADS - FIN BOX STRUCTURE - STA 624 - STA 627
UPPER SURFACE ATTACHMENTS STA 629
OUTER PLATE JOINED TO DIAPHRAGM

CRITICAL CASE P.P.O.

$$\text{MOMENT STA 629} = 122,470 \text{ (PG. 6.17)}$$

$$I = 67.7$$

$$c = 3.730 - 102 = 3.688 \text{ } \left. \begin{array}{l} \text{PG. 6.20} \end{array} \right\}$$

STRESS IN PLATE FROM FIN BOX BENDING

$$= \frac{122,470 \times 3.688}{67.7}$$

$$= 6,670 \text{ PSI}$$

$$\text{PLATE AREA OVER DIAPH. (WIDTH} = 2 \times 7.55 \text{ (PG. 6.20))}$$

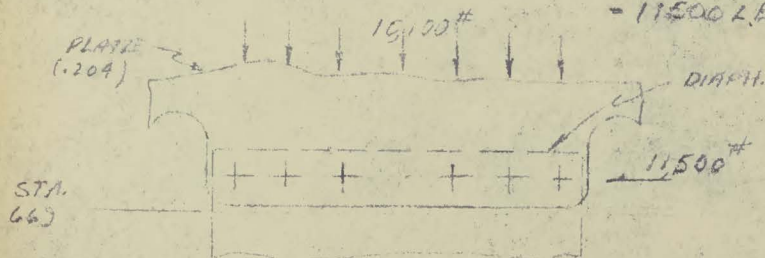
$$= 1510 \text{ IN.}^2$$

LOAD IN PLATE CAUSED BY DIAPH. ATTACHMENTS

$$= 1510 \times 6,670 = 10,100 \text{ LBS.}$$

IN ADDITION TO ABOVE LOAD, ATTACHMENTS
ARE LOADED BY DIAPH. LOADS DUE TO FIN ROLL.

$$\text{LOAD IN PLATE FROM DIAPH.} = 45000/4 \text{ (PG. 2.05)} \\ = 11,500 \text{ LBS.}$$



$$\text{RESULT. LOAD PER ATTACHMENT} = \left[\frac{(10,100)^2}{6} + \frac{(11,500)^2}{6} \right]^{1/2} \\ = 25,600 \text{ LBS.}$$

ALLOW. LOAD = 46,500 LBS.

$$\text{M.S. } \frac{46,500}{25,600} = 1.82$$

.82 M.S.

A. V. ROE CANADA LIMITED
MALTON - ONTARIO
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. _____

SHEET NO. 629

AIRCRAFT:

C-125

PREPARED BY

DATE

MCCABE

10-12-55

CHECKED BY

DATE

AIR FUEL TANK LOADS - FIN BOX STRUCTURE - STA 624 STA 657
UPPER SURFACE ATTACHMENTS STA. 669
OUTER PLATE ATTACH TO WING SKIN

CRITICAL CASE 11.3.9

MOMENT STA. 669 = 18,000 IN. LBS. (PG. 619)

$I = 67.7$

$C = (3.79 - .212) = 3.578" \quad \left. \vphantom{C = (3.79 - .212) = 3.578"} \right\} 6.20$

STRESS IN PLATE FROM FIN BOX BENDING

$= \frac{18,000 \times 3.578}{67.7}$

$= 9400 \text{ PSI.}$

PLATE AREA AT ATTACHMENT TO WING SKIN

$= .449 \text{ IN.}^2 \text{ (PG. 620)}$

LOAD IN PLATE $= 9400 \times .449 = 4220 \text{ LBS.}$

ASSUMED LOAD IN PLATE IS TRANSFERRED TO WING SKIN THROUGH 4-1/4" ATTACHMENTS, THEN, LOAD PER ATTACHMENT

$= 4220 / 4 = 1050 \text{ LBS.}$

IN ADDITION TO ABOVE LOAD, BOLTS ARE LOADED BY WING SKIN AND LOAD.

WING SKIN END LOAD = 10400 LBS./IN. (PG. 3.04)

PLATE ATTACHMENT TO WING SKIN IS BY

2-ROW 1/4" BOLTS @ 1.42 IN. C. ON SKIN.

LOAD PER BOLT $= \frac{10400 \times 1.42}{2} = 7400 \text{ LBS.}$

LOAD PER SHEAR FRT $= 7400 / 2 = 3700 \text{ LBS.}$

RESULTANT LOAD $= (3700^2 + 1050^2)^{1/2}$
 $= 3850 \#$

ALLOWABLE LOAD = 4500

M.S. $= \frac{4500}{3850} - 1 =$

.21 M.S.

