AVRO AIRCRAFT

TECHNICAL REPORT

S Thurston

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

ANALYZED

AIRCRAFT: C.105

REPORT No: 7-0558-38

FILE No:

No OF SHEETS:

TITLE: CUT-OUTS IN DUCT

~~CONFIDENTIAL~~

Classification cancelled / Changed to UNCLASS
By authority of AVRS
Date 30 Sept 96
Signature R.Bully
Unit / Rank / Appointment AVRS 5

PREPARED BY SEE OVER. DATE

CHECKED BY DATE

SUPERVISED BY DATE

APPROVED BY DATE

ISSUE No	REVISION No	REVISED BY	APPROVED BY	DATE	REMARKS
					15865849

FORM 1316A

A. V. ROE CANADA LIMITED
MALTON ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT:		REPORT No.	
		SHEET No.	
		PREPARED BY	DATE
		CHECKED BY	DATE

CONFIDENTIAL

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FORM 1319A

2.1 DOUBLER AROUND HOLE.

CUT OUT MAJOR AXIS 7·00 INS DOUBLER 13·10"
MINOR AXIS 4·70 INS (LOAD DIRECTION) DOUBLER. 9·10"

$p = 16{\cdot}65$ PSI
$R = 24{\cdot}65$ INS
$t = {\cdot}016$ INS (24 STA)
$t_R = {\cdot}025$ INS (24 STA).

USING R AES DATA SHEET 02.04.03.

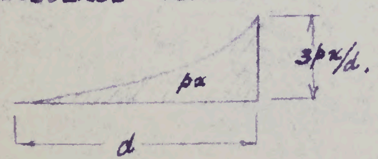
$D/d = 1{\cdot}93$
$t_r/t = 1{\cdot}56$.
$\therefore f_{MAX}/f = 1{\cdot}60$

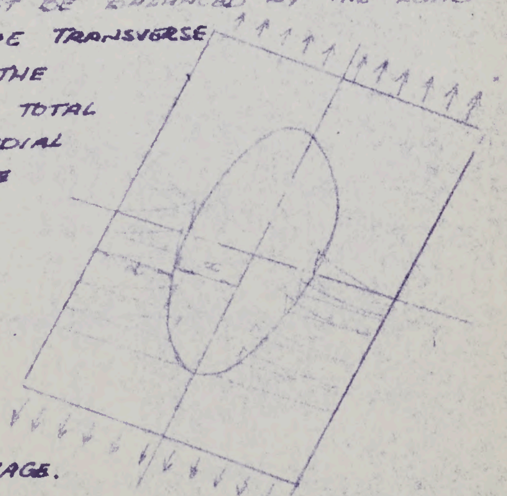
$f = \dfrac{16{\cdot}65 \times 24{\cdot}65}{{\cdot}016}$
$= 25{,}700$ PSI
$\therefore f_{MAX} = 41{,}100$ PSI

F_{TU} 24STA AT 300°F $= {\cdot}80 \times 56{,}000$
$= 44{,}800$ PSI
$1{\cdot}364\ F_{TY}$ $= 1{\cdot}364 \times 34{,}000 \times {\cdot}87$
$= 40{,}400$ PSI

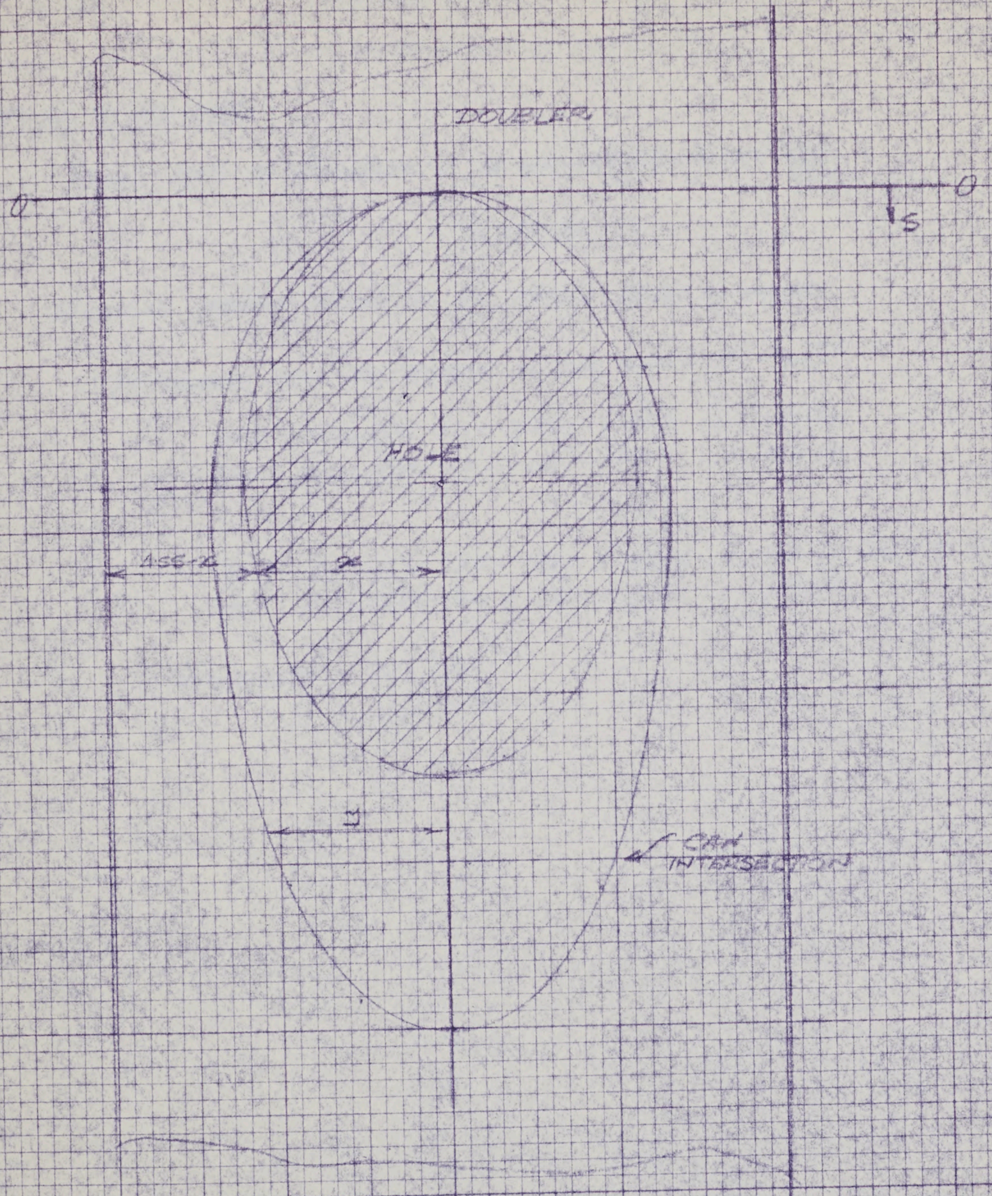
RF = 1·0

LOCAL BENDING OF DOUBLER. RADIAL LOAD REQUIRED TO BALANCE HOOP TENSION T/R lb/INS². AND THIS MUST BE BALANCED BY THE LOAD AT THE EDGE OF THE FAIRING. LET THE TRANSVERSE DISTRIBUTION OF THE EXCESS LOAD IN THE DOUBLER BE PARABOLIC SO THAT THE TOTAL LOAD / SIDE OF DOUBLE ACTING IN A RADIAL DIRECTION = px lb/in. AND HENCE THE TRANSVERSE LOADING DISTRIBUTION IS





Mr. MANNING AGREES THAT THIS EFFECT MAY BE IGNORED AT THIS STAGE.



DEVELOPED VIEW OF DOUBLER (HALF SCALE)

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

REPORT NO.	SHEET NO. 12.	
AIRCRAFT: C.105	REAR OUTBD ENG MOUNT ACCESS DOOR.	PREPARED BY K. J. IDDON. DATE 29 SEPT 55. CHECKED BY DATE

RIVETING OF DOUBLER.

TOTAL LOAD TO DOUBLER = $16.65 \times 24.65 \times 9.10 \times \frac{.025}{.041}$

= 2,280 lb.

RIVETING ABOVE HOLE IS 25 AD3's AND 6 AD4's

ALLOWABLE FOR AD RIVETS IN .016 24ST4 AT 300°F

AD3 SHEAR = .83 × .780 × 217 = 141 lb
AD3 BEARING = .90 × 1.10 × 153 = 152 lb

AD4 SHEAR = .83 × .350 × 388 = 113 lb (MUST BE > 140 lb)
AD4 BEARING = .90 × 1.10 × 205 = 200 lb

∴ STRENGTH = 31 × 141
= 4,380 lb

RF = 1.92

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

REPORT NO.		
SHEET NO.	23	
AIRCRAFT: C105	ACCESS DOOR - FRONT ENGINE MOUNT - OUTB'D	
PREPARED BY	G M COILEY	DATE 16-9-55
CHECKED BY		DATE

RIVETING OF DOUBLER AND TUNNEL FLANGE TO SHROUD SKIN.

DOUBLER TO SHROUD SKIN:-

Stress in doubler & shroud skin acting together $= \frac{45,000 \times .016}{.041} = 17,600$ lb/□"

∴ Load/in to doubler $= 17,600 \times .025 = \underline{440 \text{ lb/in}}$.

$\frac{1}{8}$ dia AN470-AD4 rivets in .016 75S-T6

Shear strength at temp. $= 314$ lb
Bearing " " " $= 211$ lb

∴ Rivet pitch reqd $= \frac{211}{440} = \underline{0.48 \text{ in}}$.

TUNNEL FLANGE TO DOUBLER AND SKIN :-

F_{TU} for 61S-T6 at temp $= 36,100$ lb/□"

∴ Load in flange $= 36,100 \times 1.7 \times .025 = \underline{1540 \text{ lb}}$

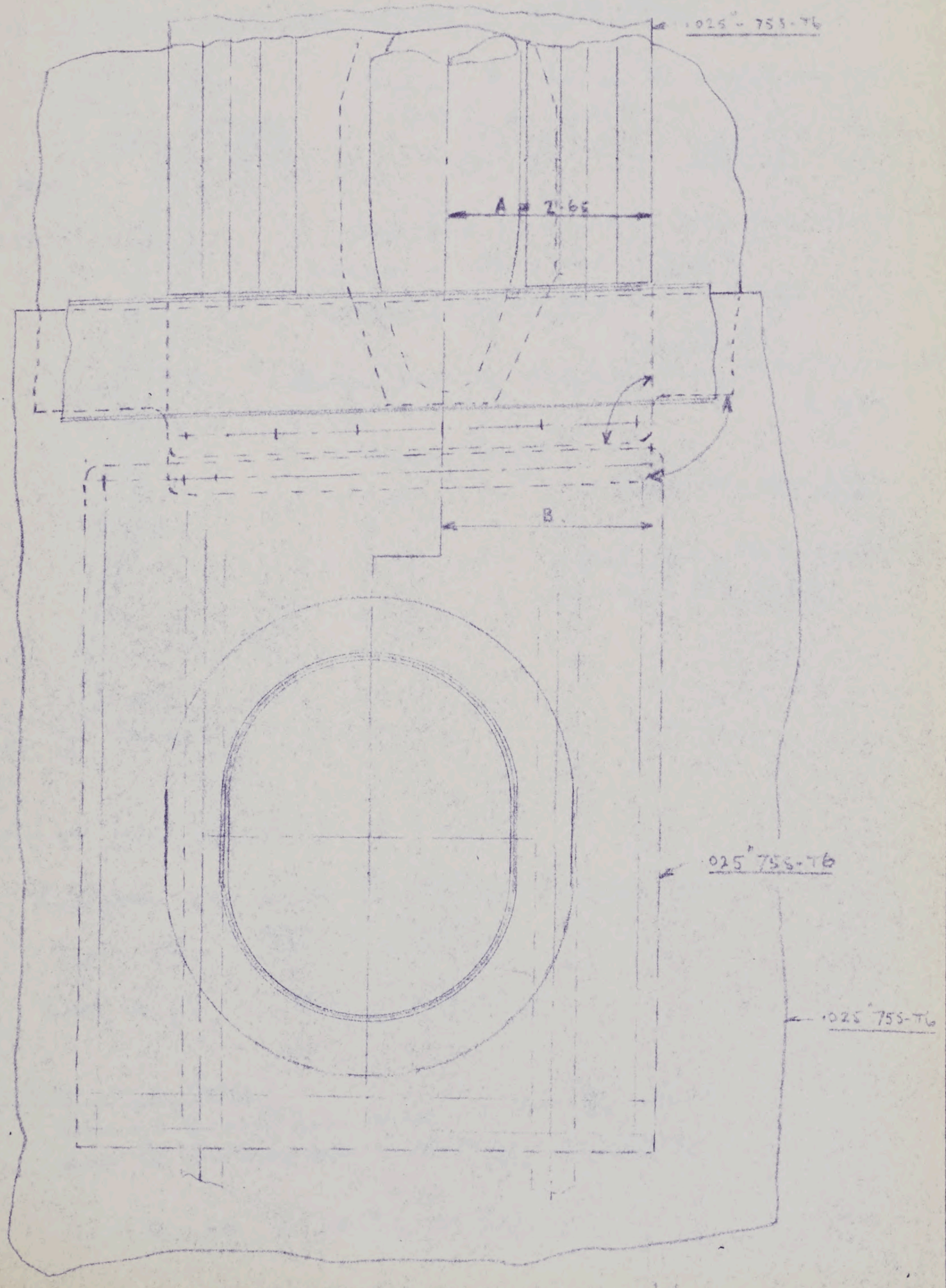
$\frac{1}{8}$ dia AN470-AD4 rivets in .025 61S-T6.

Shear strength at temp. $= 388 \times .936 \times .90 = 327$ lb.
Bearing " " " $= 321 \times .88 \times .86 = 243$ lb

No of rivets reqd $= 7$

ORM 1319A

A. V. ROE CANADA LIMITED MALTON ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO.	
		SHEET NO. 24	
AIRCRAFT:	ACCESS DOOR - FRONT	PREPARED BY	DATE
C105	MOUNT - OUTB'D	G.M COILEY	16-9-55
		CHECKED BY	DATE



VIEW ON SHROUD - LOOKING INB'D

FORM 1319A

A. V. ROE CANADA LIMITED MALTON - ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO.	
		SHEET NO. 38 d	
		PREPARED BY	DATE
AIRCRAFT: C-105	~~REAR CENTRE BLG. MTG. ACCESS~~ SHROUD ~ STA 711.58	MORGAN	FEB. 28, 1956
		CHECKED BY	DATE

FRAMES 707.3, 709.8, 712.3 & 714.8 STA $\theta_1 = 81°$

MOMENT ~ ULT

FROM REACTIONS ~ R_1 & H_1

$$M_R = 78.4 \times 24.65(1-\cos 81°) - 49.5 \times 24.65 \times \sin 81°$$
$$= 24.65\{78.4 \times .844 - 49.5 \times .986\}$$
$$= 430 \text{ IN. LB}$$

FROM SKIN SHEAR q_1 $\quad \theta_1 = 81° \quad \phi = 14°$

$$M_S = P r_1 \cdot 2.5\left(\frac{1}{7.8261} + \frac{\cos(7.8261\phi) - \cos\phi}{2(7.8261+1)} + \frac{\cos(7.8261\phi) - \cos\phi}{2(7.8261-1)} - \frac{\cos(7.8261\phi)}{7.8261}\right)$$
$$= 210 \times 24.65 \times 2.5\left[.12778 - \frac{.3340 + .970}{17.6522} - \frac{.3340 + .970}{13.6522} + \frac{.3340}{7.8261}\right]$$
$$= 210 \times 24.65 \times 2.5[.12778 - .0737 - .0955 + .0426]$$
$$= 210 \times 24.65 \times 2.5 \times .0012 = 15.7 \text{ IN. LB}$$

RESULTANT MOMENT ~ ULT

$$M = 430 + 15.7 = 445.7 \text{ IN LB.}$$

SHEAR ~ ULT

FROM REACTIONS ~ P LOAD

$$P_S' = R_1 \cos 9° + H_1 \times \sin 9°$$
$$= 78.4 \times \cos 9° - 49.5 \times \sin 9°$$
$$= 77.4 - 7.8 = 69.6 \text{ LB}$$

FROM SKIN SHEAR q_1

$$P_S'' = \sin 7°\left\{-\frac{P \times 2.5}{\cos(\psi/2)}\left(\frac{1}{7.8261} + \frac{\cos(7.8261\phi) - \cos\psi}{2(7.8261+1)} + \frac{\cos(7.8261\phi - \cos\phi)}{2(7.8261-1)}\right) - \frac{P \times 2.5}{7.8261 \times \cos(\psi/2)}[\cos(7.8261\phi) - 1]\right\}$$
$$= .1220\left\{-\frac{210 \times 2.5}{.99255}\left(.12778 - \frac{.3340 + .970}{17.6522} - \frac{.3340 + .970}{13.6522} + \frac{.3340}{7.8261}\right) - \frac{210 \times 2.5}{7.8261 \times .99255}[-.3340 - 1]\right\}$$
$$= .1220\{528.9(.12778 - .0737 - .0955 + .0426) + 67.6 \times 1.334\}$$
$$= .1220\{-528.9 \times .0012 + 67.6 \times 1.334\}$$
$$= .1220 \times 89.54 = 10.9 \text{ LB}$$

RESULTANT SHEAR ~ ULT

$$P_S = 69.6 + 10.9 = 80.5 \text{ LB}$$

FORM 1319A

FRAMES 707.3, 709.8, 712.3 & 714.8 ~ STA $\theta_1 = 81°$

AXIAL LOAD ~ ULT

FROM REACTIONS ~ P LOAD

$P_A' = -R_1 \sin 9° + H_1 \cos 9°$

$= -78.4 \times \sin 9° - 49.5 \cos 9°$

$= -12.3 - 48.9 = -61.2$ LB

FROM SKIN SHEAR q_1

$P_A'' = -\cos 9° \times 89.54$

$= -88.5$ LB

FROM HOOP TENSION

$P_A''' = 16.65 \times 24.65 \times 9.6 = 3940$ LB

RESULTANT AXIAL LOAD ~ ULT

$P_A = 3940 - 88.5 - 61.2 = 3790.3$ LB

STRESS ~ $\theta_1 = 81°$

TENSION

$f_t = (445.7 \times .1219 / .002362 \times 4) + (3790.3 / 4 \times .26438)$

$= 5750 + 3585 = 9335$ P.S.I ULT OR 6990 P.S.I. LIMIT

ALLOWABLE 24S-T4 CLAD ~ YIELD

$F_{ty} = 34000 \times .86 = 29200$ P.S.I

MARGIN OF SAFETY

M.S. = $(29200/6990) - 1 = +3.22$ +3.22

COMPRESSION ~ ULT

$f_c = (445.7 \times (.442 - .039) / .002362 \times 4) - 3790.3 / 4 \times .26438)$

$= 19000 - 3585 = 15415$ P.S.I

ALLOWABLE 24S-T4 CLAD

$F_{cc} = 3.62 \times 10^7 / 42.9^2 = 19150$ P.S.I $b/t = (.75 - 4 \times .016)/.016 = 42.9$

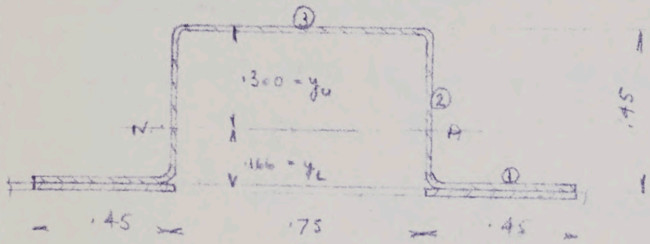
MARGIN OF SAFETY

M.S. = $(19150/15415) - 1 = +.24$ +.24

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

AIRCRAFT: C105	GEAR BOX DRIVE SEAL	REPORT No.
		SHEET No. 40
		PREPARED BY: G.M. COILEY. — DATE: 29-9-58
		CHECKED BY: — DATE:

STRENGTH OF HAT SECTIONS CUT OFF AT TOP OF SEAL



SCALE – 2/1
MAT^L – Ti
$t = 0.016''$
I_{NA} (INCLUDING 30t OF SKIN)
$= 0.001886 \text{ in}^4$

TO FIND CRIPPLING STRESS FOR ITEM (3) :-

$$F_{cc} = \frac{C_e\sqrt{F_{cy}E}}{\left(\frac{b'}{t}\right)^{0.75}}$$

$$= \frac{0.366\sqrt{1{,}363{,}000}}{(32.5)^{0.75}}$$

$$= \frac{0.366 \times 1{,}170}{13.6} = 41.1 \text{ k.p.in}^2$$

where $C_e = 0.366.$
$F_{cy} = 110 \times x \cdot 80 = 88 \text{ k.p.i}^2$
$E = 15{,}500 \text{ k.p.i}^2$
$b' = (\cdot 74 + \cdot 3)\frac{1}{2} = \cdot 52 \text{ in.}$
$t = 0.016 \text{ in}$

TO FIND CRIPPLING STRESS FOR ITEM ① :-

$$F_{cc} = \frac{\cdot 342 \times 1{,}150}{(19.1)^{0.75}}$$

$$= \frac{\cdot 342 \times 1{,}170}{9.11}$$

$$= 43.8 \text{ k.p.in}^2$$

$C_e = \cdot 342$
$F_{cy} = 88 \text{ k.p.in}^2$
$E = 15{,}500 \text{ k.p.i}^2$
$b' = (\cdot 45 + \cdot 16)\frac{1}{2} = \cdot 305 \text{ in}$
$t = \cdot 016 \text{ in}$

BENDING STRENGTHS :-
COMP^N AT TOP, TEN^N AT BOTTOM

$$\text{Bending Strength} = \frac{F_{cc_u} \times I}{y_u} = \frac{41.1 \times \cdot 001886}{\cdot 3} = \cdot 258 \text{ k-in} \quad \underline{\cdot 258 \text{ k-in}}$$

$$\text{or} \quad \frac{F_{TU_L} \times I}{y_L} = \frac{120 \times \cdot 76 \times \cdot 001886}{\cdot 166} = 1.037 \text{ k-in} = \underline{1.037 \text{ k-in}}$$

FORM 1319A

AIRCRAFT		PREPARED BY	DATE
C105	GEAR BOX DRIVE SEAL	G. M. COILEY	29-4-55
		CHECKED BY	DATE

TEN^{N} AT TOP COMP^{N} AT BOTTOM:-

$$\text{Bending Strength} = \frac{F_{cc_L} I}{y_L} = \frac{43.8 \times 0.001886}{.166} = 0.499 \text{k-in} = \underline{499 \text{ k-in}}$$

$$\text{OR}, \quad \frac{F_{TU_u} I}{y_u} = \frac{120 \times .76 \times .001886}{.3} = 0.574 \text{ k-in} = \underline{570 \text{ k-in}}$$

THE PANEL JUST ABOVE THE TITANIUM PANEL IS LIGHT ALLOY SO CONSIDERING THE STRENGTH OF LIGHT ALLOY PANEL AS BEING EFFECTIVE:-

CRIPPLING STRESS FOR ITEM 3 :-

$$F_{cc} = \frac{C_e \sqrt{F_{cy} E}}{\left(\frac{b'}{t}\right)^{.75}}$$

$C_e = 0.366$
$F_{cy} = 64 \times .70 = 44.8 \text{ k/in}^2$
$E = 10,500 \times .91 = 9,550 \text{ k/in}^2$
$b' = (.37 + .3)\frac{1}{2} = .335$
$t = .016 \text{ in}$

$$= \frac{.366 \sqrt{\{428\,000\}}}{9.78}$$

$$= \underline{24.5 \text{ k/in}^2}$$

CRIPPLING STRESS FOR ITEM 1 :-

$C_e = .342$
$b' = .45 + .16 \cdot 3 = .305 \text{ in}$

$$F_{cc} = \frac{.342 \times \sqrt{\{428\,000\}}}{9.11}$$

$$= \underline{24.6 \text{ k/in}^2}$$

BENDING STRENGTHS :-

TEN^{N} AT BOTTOM.

$$\text{Bending Strength} = \frac{F_{cc_u} \times I}{y_u} = \frac{24.5 \times .001886}{.3} = \underline{.154 \text{ k-in}}$$

$$\text{OR} = \frac{F_{TU_L} \times I}{y_L} = \frac{70 \times .74 \times .001886}{.166} = \underline{.588 \text{ k-in}}$$

COMP^{N} AT BOTTOM,

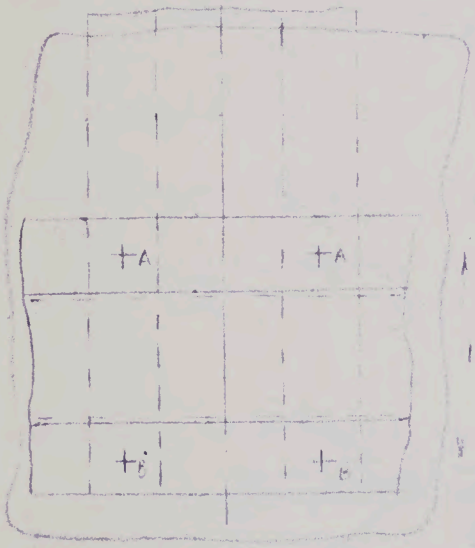
$$\text{Bending Strength} = \frac{F_{TU_u} \times I}{y_u} = \frac{70 \times .74 \times .001886}{0.3} = \underline{.325 \text{ k-in}}$$

$$\text{OR} = \frac{F_{cc_L} \times I}{y_L} = \frac{24.6 \times .001886}{.166} = \underline{.279 \text{ k-in}}$$

A. V. ROE CANADA LIMITED MALTON · ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. SHEET NO. 42	
		PREPARED BY	DATE
AIRCRAFT: C105	GEAR BOX DRIVE SEAL	C. M. COILEY	29-9-57
		CHECKED BY	DATE

THIS CONNECTION TO CUT OFF HAT SECTIONS MUST BE GOOD FOR 154 lb-in.

REACTING BM AT RIVETS A & B:-



LOAD ON EACH RIVET $= \pm \dfrac{279}{2 \times 1.25} = \underline{111.5 \text{ lb}}$

For $\dfrac{t}{d} = \dfrac{.016}{.125} = .128$

NACA TN 930 GIVES:-

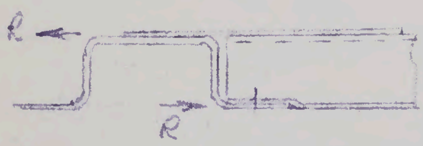
$\dfrac{\text{Tensile strength of rivet}}{\text{" " " shank}} = 0.26$

Tensile strength of rivet

$= \dfrac{.26 \times \pi \times .125^2 \times 56}{4} = \underline{.174 \text{ lb}}$

Bending of the rivet heads will take place due to bending of the flange. The method of attachment is not considered safe.

REACTING BM AT TOP AND BOTTOM OF SECTION



$R = \dfrac{154}{.45} = \underline{342 \text{ lb}}$

Cherry (CR 156, CR 157) 1/8 dia. Rivets in 0.016" 75S-T6

Shear Strength $= 395 \times .9 \times .8 = 284$ lb

Bearing " $= 203 \times 1.44 \times .77 = \underline{231 \text{ lb}}$

∴ 2 rivets are req'd for rivet attachment and 2 will be sufficient for lower flange, because load is already in stressed skin.

SHEAR STRESS IN TORSION MEMBER:-

$$q = \frac{154}{2 \times .44 \times .74 \times .016} = \underline{14{,}900 \text{ lb/in}^2}$$

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

REPORT NO.		
SHEET NO. 43		
PREPARED BY: G. M. COILET	DATE: 2-10-55	
CHECKED BY	DATE	

AIRCRAFT: C105 — GEAR BOX DRIVE SEAL

f_b for top of cut. (sheared skin under section is stiffened by doubler):-

$$= 4.85 \times 10{,}500 \times .91 \times \left(\frac{.016}{.62}\right)^2$$

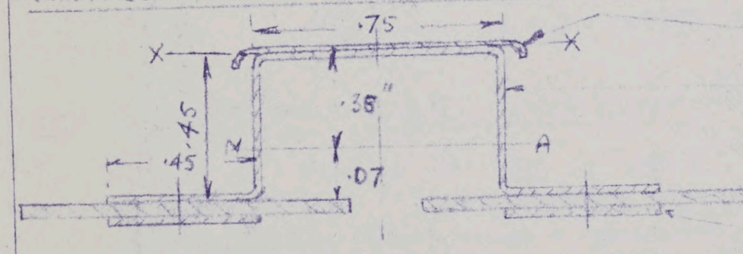
$$= \underline{30.8 \text{ k.p.s.i.}} \qquad \text{Also } F_{su} = 84 \times .86 = \underline{72.2 \text{ k.s.i.}}$$

Riveting of tension member to skin:

Shear/in $= 14{,}900 \times .016 = 247$ lb/in.

Rivet bite req'd $= \dfrac{231}{247} = \underline{0.935 \text{ in.}}$

REINFORCING OF HAT SECTIONS ADJACENT TO CUT-OUT :-



E for 75S-T6 $= 10{,}500 \times .91 = \underline{9{,}550 \text{ k.p.s.i.}}$

E for Ti. $= \underline{15{,}800 \text{ k.s.i.}}$

Areas of Ti will be multiplied by $\dfrac{15{,}800}{9{,}550} = \underline{1.654}$

1st Moment of Area about XX :-

$.016 \times .90 \times .225 = .00324$

$.016 \times .90 \times .44 = .00634$

$1.654 \times .016 \times .96 \times .49 = .01245$

$1.654 \times .032 \times 1.92 \times .47 = \underline{.04780}$

TOTAL $= \underline{.06983 \text{ in}^3}$

Area

$.016(.9 + .75 + 1.8) = .0552$

$.032 \times 1.92 \times 1.654 = .1015$

$.016 \times .96 \times 1.654 = \underline{.0254}$

TOTAL $= \underline{0.1821 \text{ in}^2}$

$$\bar{x} = \frac{.06983}{.1821} = \underline{0.383 \text{ in}} \qquad \underline{\text{Say } 0.38 \text{ in}}$$

FORM 1319 A

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

AIRCRAFT: C105	GEAR BOX DRIVE SEAL	REPORT NO.	
		SHEET NO. 44	
		PREPARED BY: G. M. COLLET	DATE: 2-9-55
		CHECKED BY	DATE

2nd moment of area about N.A.:-

$.016 \times .9 \times .39^2 = .00219$
$.016 \times .75 \times .37^2 = .00164$
$.016 \times .38^3 \times \frac{1}{3} = .00029$
$.016 \times .07^3 \times \frac{1}{3} = .00000$
$.016 \times .9 \times .06^2 = .00052$
$.016 \times .96 \times .110^2 \times 1.654 = .00031$
$.032 \times 1.92 \times .086^2 \times 1.654 = .00075$

$.00570 \text{ in}^4$

2nd moment of area of normal Aluminium Alloy section
$= .001886 \text{ in}^4$

$\therefore$ 2nd moment req'd $= .003772 \text{ in}^4$ OK.

TORSION MEMBER - REINFORCED MEMBER CLEAT.

Shear to be carried $= \frac{154}{.45} = 342$ lb

Say moment on cleat rivets $= 342 \times .6 = 205$ lb-in.

$\therefore$ Load on rivets due to moment $= \frac{205}{1.0} = \pm 205$ lb

Direct load on each rivet $= \frac{342}{3} = 114$ lb.

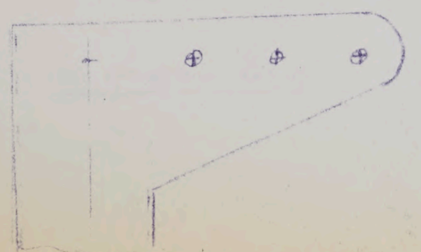
Total load on worst loaded rivet $= 319$ lb. no good

Limiting moment given by Mr K. Iddon is 121 lb-in

Load on worst loaded rivet $= \frac{319 \times 121}{154} = 250$ lb.

Allow this load for section is near simply supported edge of shroud & full strength of member will probably not be used.

Centre Cleats:-



Compression stress $= \frac{342}{.036 \times 1.1} = 8{,}640 \text{ lb/in}^2$

$f_b = .385 \times 9{,}550 \text{ K} \times \left(\frac{.036}{.55}\right)^2 = 15{,}700 \text{ lb/in}^2$

RM 1319A

A. V. ROE CANADA LIMITED MALTON - ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO.	
		SHEET NO. 45	
		PREPARED BY	DATE
AIRCRAFT: C105	GEAR BOX DRIVE SEAL	G.M. COILEY	30-9-55
		CHECKED BY	DATE

HOOP TENSION IN SHROUD.

TENSION / IN = pr = 25.55×30.5 = 779 lb/in FACTORED.

$\therefore$ STRESS IN 0.016" SKIN = $\frac{779}{.016}$ = 48700 lb/□"

F_{TU} FOR Ti (COLD ROLLED) = $120 \times .76$ = 91,100 lb/□".

STRESS CONCENTRATION FACTOR AT TOP OF CUT-OUT :-

Using R.Ae.S. Data Sheet 02.04.03 :-

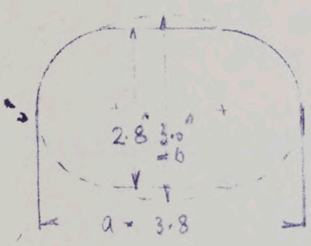
Plate thickness t = 0.016"
Thickness of reinforcement t_r = .032"

$\therefore \frac{t_r}{t} = 2.0$

Say d = 3.8 in
and D = 7.58 in

then $\frac{D}{d} = 2.0$

From Data Sheet f_{max}/f = 1.64.



The top of the door however is not reinforced. Using the stress concentration factors given in ROARK page 345 for an ellipse

For a circular hole :- ($a = b$)

$$k = 1 + 2\frac{a}{b} = 3.0$$

For an elliptical hole :- ($a = 3.8$, $b = 3.0$)

$$k = 1 + \frac{3.8 \times 2}{3.0} = 3.54.$$

Therefore multiplying the factor for a circular hole by $\frac{3.54}{3.0}$ gives f_{max}/f = 1.94

$\therefore f_{max}$ = $1.94 \times 48,700$ lb/□"
= 94,400 lb/□"

Then RF on F_{TU} = $\frac{91,100}{94,400}$ = 0.966

MR. K. IRONS has given OK for this R.F.

A. V. ROE CANADA LIMITED MALTON - ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT No.	
		SHEET No. 46	
AIRCRAFT: C105	GEAR BOX DRIVE SEAL	PREPARED BY	DATE
		G. M. COLLEY	2?-?-5?
		CHECKED BY	DATE

RIVETING AT TOP OF ·032 ADAPTER.

Consider one former pitch. (2·40")

Area of shroud skin = 2·4 × ·016 = ·0384 □"

Area of stiffener = ·03814 □"

Load per pitch = 779 × 2·40 = 1870 lb

∴ Load in shroud skin per pitch = $\frac{1870 \times ·0384 \times 1·654}{(1·654 \times ·0384 + ·03824)}$

= $\underline{1170 \text{ lb}}$

Load in stiffener = 1870 − 1170 = $\underline{700 \text{ lb}}$

Strength of $\frac{1}{8}$ dia AN470-AD4 rivets in ·016"

Shear strength = 388 × ·9 × ·9 = 314 lb
Bearing " = 208 × 1.2 × ·94 = $\underline{235 \text{ lb}}$. } Rivet Mat^l taken as for 24S.

STIFFENER – DOUBLER RIVETING

All the load must be taken out of the stiffener

∴ Rivets req'd = $\frac{700}{235}$ = 2·98 say 4.

SHROUD SKIN – DOUBLER RIVETING

Load in shroud skin when acting with doubler.

= $\frac{1870 \times ·016}{·016 + ·032}$ = 625 lb.

∴ Load to doubler = 1170 − 625 = 545 lb./pitch.

∴ Rivets req'd = $\frac{545}{235}$ = 2·32 say 3·0 per stiffener pitch.

SHROUD SKIN AND DOUBLER – TORSION BOX RIVETING

Width applied = 7·58"

∴ Total load = $\frac{1870 \times 7·58}{2·40}$ = 5900 lb

Strength of AN470-AD4 rivets in 0·025 in (Torsion Box skin)

Shear strength = 388 × ·936 × ·90 = $\underline{328 \text{ lb}}$.
Bearing " = 463 × 1.2 × ·94 = 522 lb } Rivets mat^l taken as for 24S.

∴ No of rivets req'd = $\frac{5,900}{328}$ = 18.

A. V. ROE CANADA LIMITED MALTON · ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**	REPORT NO.	
	SHEET NO.	47
AIRCRAFT: C105 — GEAR BOX DRIVE SEAL	PREPARED BY: G. M. COILEY	DATE: 3-10-55
	CHECKED BY	DATE

RIVETING AT BOTTOM OF ADAPTER TO COMPLETE TORSION BOX:-

Torque in torsion Box:—

STN.	ASSYMMETRIC CASE	SYMMETRIC CASE
591.65	56,340 lb.in	55,579 lb.in
606.05 } 610.85 } 615.65	37,830 lb.in equally	39,930 lb.in equally.
625.25	17,070 lb.in	16,079 lb.in
644.43	61,440 lb.in	61,074
TOTALS.	172,680 lb.in	172,662 lb.in

T = Torque uniformly applied $= -\frac{172,680}{52.78} = -3270$ lb.in/in.

ASSYMMETRIC CASE

STN. x_n	$x_n - x_{n-1}$	$T(x_n - x_{n-1})$	S	$\int T(x_n - x_{n-1})\,\&\,S\,dx$
				0
591.65			56,340	+56,340
	14.40	-47,100		+9,240
606.05			12,630	+21,870
	4.80	-15,700		+6,770
610.85			12,630	+19,410
	4.80	-15,700		+3,710
615.65			12,630	+16,340
	9.60	-31,400		-15,060
625.25			17,070	+2,010
	19.18	-62,600		-60,590
644.43			61,440	NEGLIGIBLE

				0
591.65			55,579	+55,579
		-47,100		+8,479
606.05			13,300	+21,779
		-15,700		+6,079
610.85			13,300	+19,379
		-15,700		+3,679
615.65			13,300	+16,9[illegible]9
		-31,400		-14,421
625.25			16,079	-1,658
		-62,600		-64,158
644.43			61,074	NEGLIGIBLE

Reduced area of torsion box $= 34.17 - 3.24$

$= 30.93\ \square''$

Over the bays under consideration the torque varies between 21,870 lb.in and 6,770 lb.in, the higher value will be used

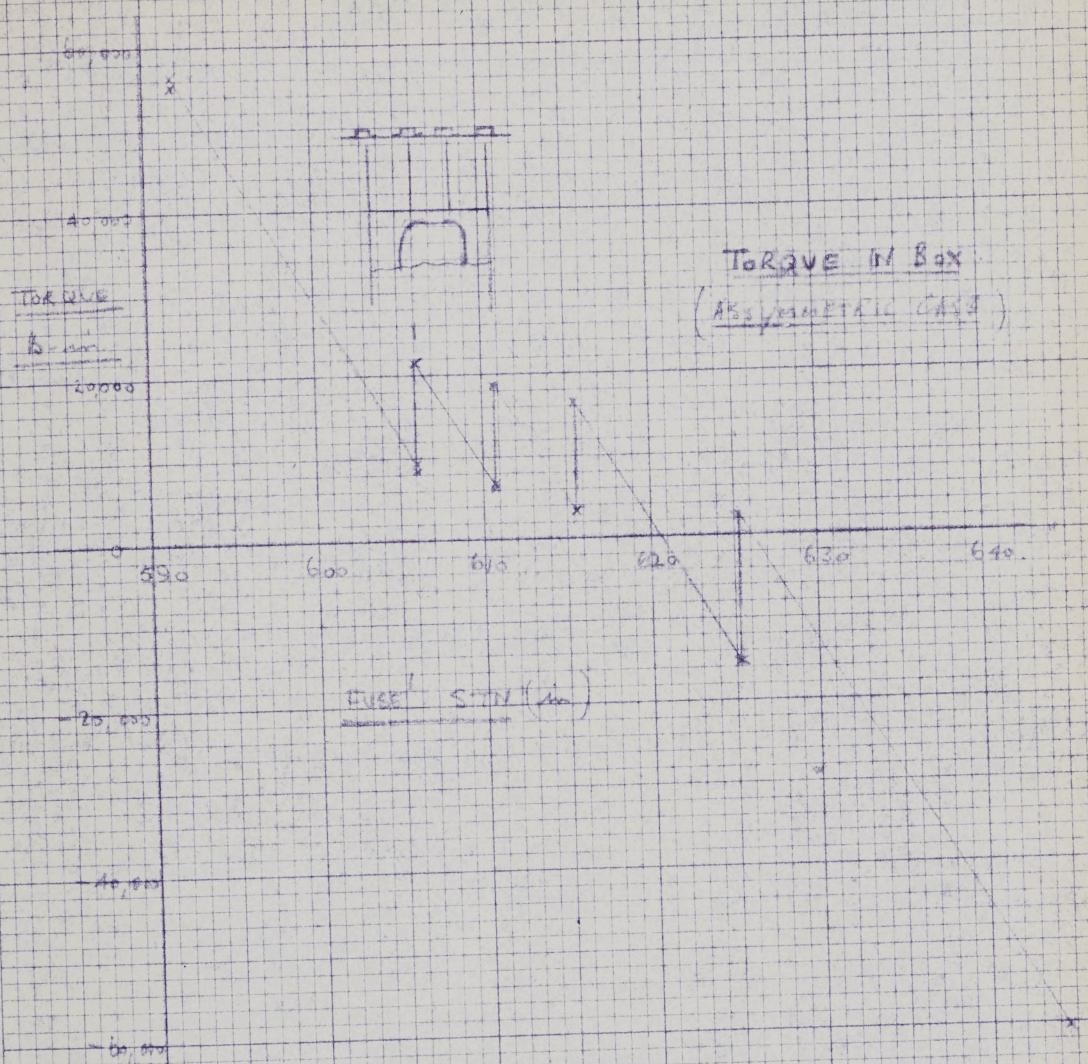
$$S/\text{in} = \frac{21,870}{2 \times 30.93} = 355 \text{ lb/in}$$

$$\therefore \text{Pitch req} = \frac{328}{355} = 0.924 \text{ in}$$

FORM 1319A

RAFT C105 COMPONENT GEAR BOX DRIVE SEAL SHEET No. 48 REPORT No.

W. DATE 3-10-55. PREP. BY G.M. COLEY



FORM 1746

AIRCRAFT:		PREPARED BY	DATE
C105	GEAR BOX DRIVE SEAL	G. M. COILEY	30-9-57
		CHECKED BY	DATE

CAP - SHELL RIVETING

Load/in on rivets due to pressure on cap:-

$= 25.55 \times \frac{3.8}{2} = \underline{48.6 \text{ lb}}$ (Shear)

Load/in on rivets due to pressure on shell:-

$= 25.55 \times \frac{3.0}{2} = \underline{38.3 \text{ lb}}$ (tension)

∴ Load on each rivet at .62 pitch = $\begin{cases} 30.1 \text{ lb (shear)} \\ 23.7 \text{ lb (tension)} \end{cases}$

$\frac{1}{8}$ dia AN470 - AD4 rivets in .020" T[illegible] in tension:-

(NACA T.N. 930) $\frac{\text{Sheet thickness}}{\text{Rivet dia.}} = \frac{.02}{.125} = .16$

Then:- $\frac{\text{Tensile strength of rivet}}{\text{" " " shank}} = 0.4$

∴ Tensile strength of rivet $= 0.4 \times \frac{\pi}{4} \times .125^2 \times 55000 = \underline{269 \text{ lb}}$ OK

SHELL - ADAPTER RIVETING:- Loading and riveting as per Cap - Shell riveting.

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

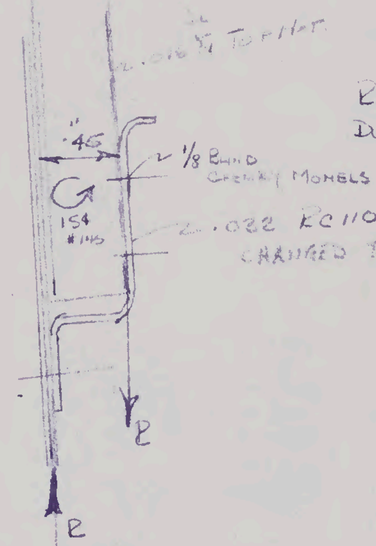
AIRCRAFT: C105	GEAR BOX DRIVE SEAL	REPORT No 7-0558-28
		SHEET No 52
		PREPARED BY A. FERENC · DATE DEC/55
		CHECKED BY · DATE

CHECK CONNECTION AT STRINGER TERMINATION

STRINGER MOMENT CAPACITY = 154 IN LBS REF SHT 42

$R = \frac{154}{.45} = 354$ LBS

REF SECTION AA
DWG 7-0159-308/3



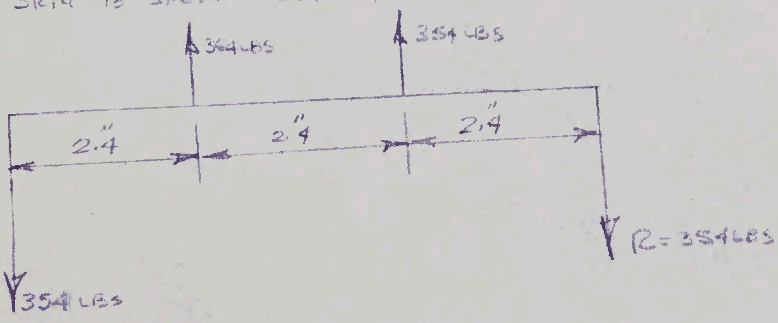
ALLOWABLE S.S REF ANC-5 PAGE 54
= 775 LBS

ALLOWABLE BEARING STRENGTH
IN .016 Ti AT TEMP
$= .125 \times .016 \times 170000 \times .75 =$
$= 259$ LBS

IN .016 R 7506
$= .125 \times .016 \times 133000 \times .82 = 218$ LBS

TOTAL CAPACITY = 2 X 259 = 519 LBS , $R_F = \frac{519}{354}$ 1.46

THESE LOADS ARE BEAMED OUT BY THE LIPPED 'Z' SECTION TO THE TWO EXTENDED STRINGERS ADJACENT TO THE HOLES. THE REACTION ADJACENT TO SKIN IS SHEARED OUT BY SKIN TO THE TWO STRINGERS.



B.M MAX AT CENTER = 354 (3.6 - 1.2) = 850 LBS INS

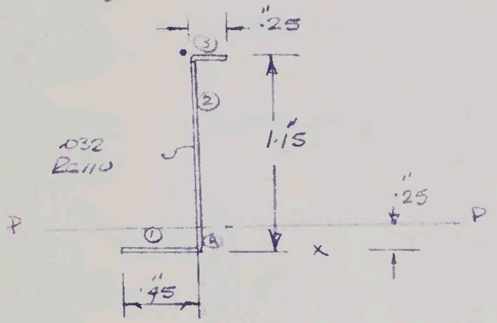
B.M AT .9" FROM END = .9 X 354 = 318 LBS INS

AT .9" FROM EITHER END THE LIPPED Z IS CUT AWAY LOCALLY TO CLEAR STRINGER

A. V. ROE CANADA LIMITED MALTON · ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT No. 7-0358-28 SHEET No. 53	
		PREPARED BY	DATE
AIRCRAFT: C105	GEAR BOX DRIVE SEAL	A. FERENC	DEC/55
		CHECKED BY	DATE

SECTION BENDING PROPERTIES

ALL RADII SQUARED OFF FOR CONVENIENCE



BENDING SECTION AT CENTER IS AS SHOWN
AT .9" FROM END BENDING SECTION IS ABOVE LINE PP.

ITEM	b	t	A	y	Ay	Ay²	Icg
1	.45	.032	.0144	.016	.0002305	NEG	NEG
2	1.118	.032	.0378	.559	.0211500	.011850	.00438
3	.25	.032	.0080	1.134	.0090700	.010280	NEG
Σ			.0602	.507	.0304505	.022130	.00438

$$I_{CG} = .022130 + .00438 - .0602 \times .507^2 = .02651 - .01545 = .01006 \text{ INS}^4$$

$$Z_T = \frac{.01006}{.643} = .01565 \text{ INS}^3 \qquad Z_C = \frac{.01006}{.507} = .01985 \text{ INS}^3$$

BENDING SECTION AT .9" FROM ENDS

ITEM	b	t	A	y	Ay	Ay²	Icg
Σ 1-3			.0602	.507	.0304505	.022130	.00438
①			$\overline{.0144}$	.016	$\overline{.0002305}$	NEG	NEG
4	.25	.032	$\overline{.008}$	.125	$\overline{.001}$	$\overline{.000125}$	NEG
			.0378	.718	.0271455	.022005	.00438

$$I_{CG} = .022005 + .00438 - .718^2 \times .0378 = .026385 - .0195 = .006885 \text{ INS}^4$$

$$Z_T = \frac{.006885}{.432} = .0155 \text{ INS}^3 \qquad Z_C = \frac{.006885}{.468} = .01423 \text{ INS}^3$$

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

REPORT NO. 7-0558-38		
SHEET NO. 54		
AIRCRAFT: C105	GEAR BOX DRIVE SEAL	PREPARED BY: A. FEENS — DATE: DEC/55
		CHECKED BY — DATE

CHECK BENDING CAPACITY OF .032 STIFFENER (LIPPED Z SECTION)

fb AT CENTER

$f_{bc} = \frac{850}{.01985} = 42800 \text{ LBS/INS}^2$

$f_{bT} = \frac{850}{.01565} = 54300 \text{ LBS/INS}^2$

ALLOWABLE F_{cy} AT TEMP = 110000 x .58 = 63800 LBS/INS²

ALLOWABLE F_{TU} AT TEMP = .75 x 120,000 = 90000 LBS/INS²

$R_{F\,TENS} = \frac{90000}{54300}$

$R_{F\,COMP} = \frac{63800}{42800}$

SINCE LOADING MAY BE REVERSED MIN $R_{F\,COMP} = \frac{63800}{54300}$ * 1.175

CHECK LOCAL CRIPPLING

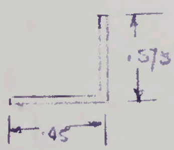
* SEE SHT 54

USING NEEDHAMS FORMULA REF GEN/1030/321-1

E AT TEMP (COMP. LONG.) REF PAR TN MET. 184
APR 28/1955
= 15.9 x 10⁶

ONE EDGE FIXED
ONE FREE

$$F_{cc} = \frac{.342\left(63800 \times 15.9 \times 10^6\right)^{1/2}}{\left(\frac{.575 + .25}{2 \times .032}\right)^{.75}} = \frac{.342 \times 1,008,000}{6.14} = 56700 \text{ LBS/INS}^2$$



BOTH EDGES FIXED

$$F_{cc} = \frac{.366 \times 1008000}{\left(\frac{.45 + .575}{2 \times .032}\right)^{.75}} = \frac{.366 \times 1008000}{8.0} = 46100$$

$R_{F\,CR} = \frac{46100}{54300} = .847$

CHANGING GAGE TO .040

$$F_{cc} = \frac{.366 \times 1008000}{6.72} = 54800$$

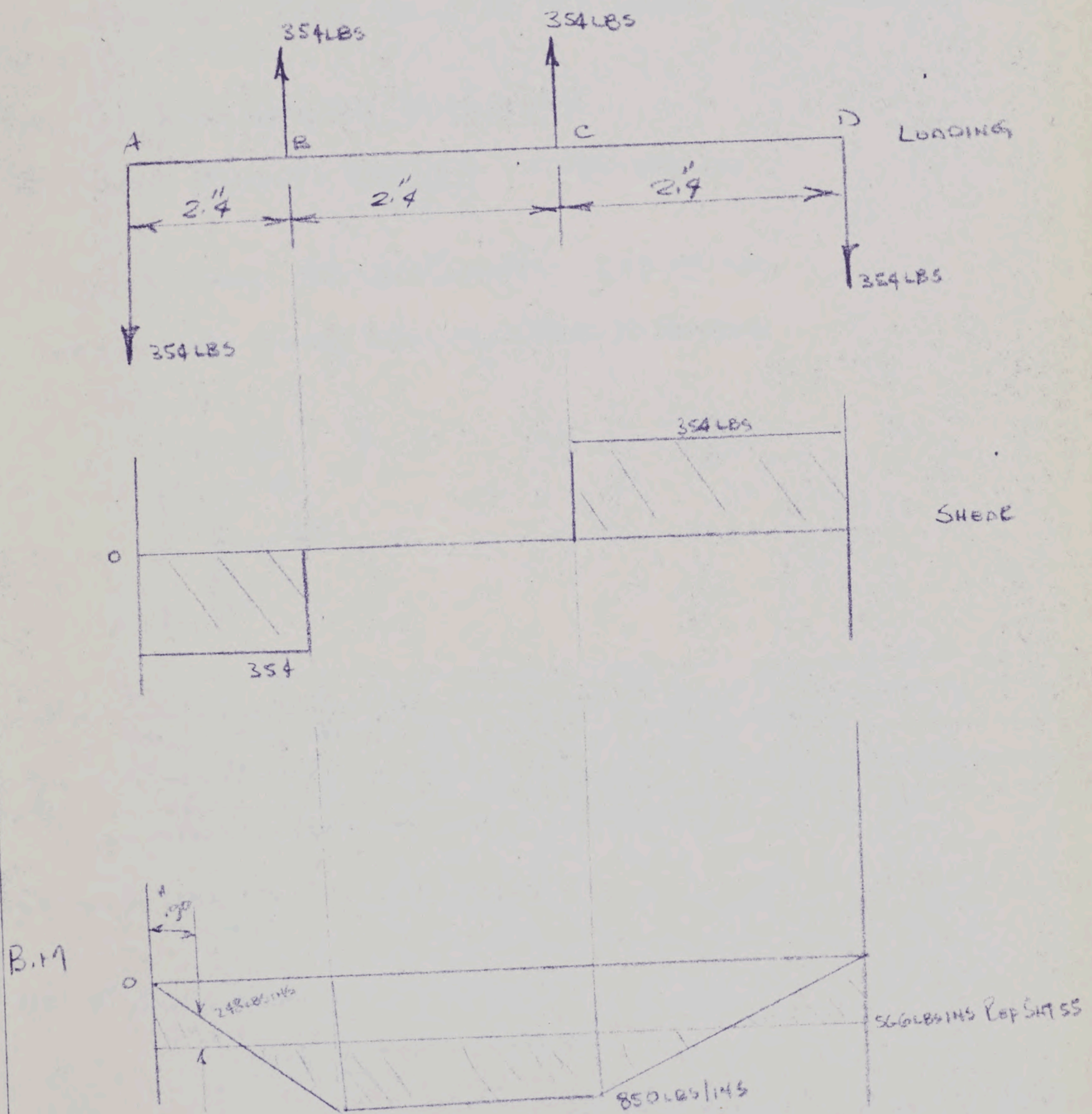
$R_F = \frac{54800}{54300} = 1.008$

SHEAR IN THIS AREA = 0

FORM 1319 A

A. V. ROE CANADA LIMITED MALTON - ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7-0558-33 SHEET NO. 56	
		PREPARED BY	DATE
AIRCRAFT: C105	GEAR BOX DRIVE SEAL	A. FURENC	DEC/55
		CHECKED BY	DATE

SKETCH OF SHEAR & B.M DUE TO LOADING ON PAGE 52



MAX AV SHEAR STRESS AT B = $\frac{354}{1.15 \times .40}$ = 7700 LBS/INS^2

MAX PRINCIPLE COMP STRESS = $\frac{54300}{2} + \left(\frac{54300}{2}^2 + 7700^2\right)$

= 27650 + 28700 = 56350 LBS/INS^2

PRINCIPLE S.S = 28700 LBS/INS^2

RF COMP = $\frac{63800}{56350}$

1.132

THIS IS ASSUMING THAT THE BENDING COMP STRESS IS THE SAME

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

REPORT NO. 7-0558-38
SHEET NO. 57

AIRCRAFT:		PREPARED BY	DATE
C105	GEAR BOX DRIVE SEAL	A. FERENC	DEC/55
		CHECKED BY	DATE

SECTION AT .9" AWAY FROM END

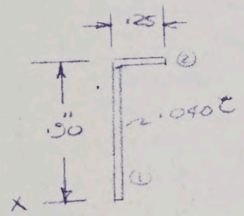
THE WORST B.M MAY OCCUR AT THIS SECTION BY ASSUMING FULL FIXITY AT ENDS.

FROM THE B.M DIAGRAM OF SHT 56

THE FIXING MOMENT $= \frac{850 \times 4.8}{7.2} = 566$ LBS INS

$\therefore$ BM AT .9" AWAY $= 566 - 318$ (SHT 52) $= 248$ LBS INS

$\therefore$ THE CASE OF SIMPLY SUPPORTED BEAM IS CRITICAL

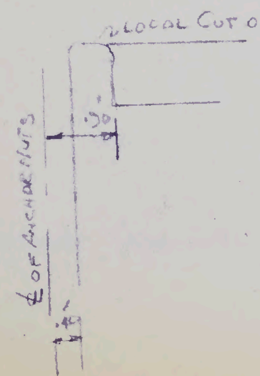


ITEM	b	t	A	y	Ay	Ay²	Icg
1	.86	.040	.0343	.43	.01475	.00635	.00220
2	.25	.040	.0100	.88	.00880	.00773	NEG
Σ			.0443	.53	.02355	.01408	.00220

$I_{NA} = .01408 + .00220 - .53^2 \times .0443 = .01628 - .0125 = .00378$ INS^4

$f_{b\,max} = \frac{318 \times .53}{.00378} = 44700$ LBS/INS^2

AT .4" AWAY FROM END MOMENT



B.M AT .40" AWAY $= 566 - .4 \times 354 = 424$ LBS INS

MOMENT $= \frac{424}{318} \times 44700 = 59500$ LBS/INS^2

RM 1319A

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

AIRCRAFT: C105	GEAR BOX SEAL DRIVE	REPORT NO. 7-0558-38	SHEET NO. 58
		PREPARED BY A. FERENS	DATE DEC/55
		CHECKED BY	DATE

CHECKING B.M ALONG TONGUE OF STIFFENER

FIXING MOMENT = 566 INS LBS

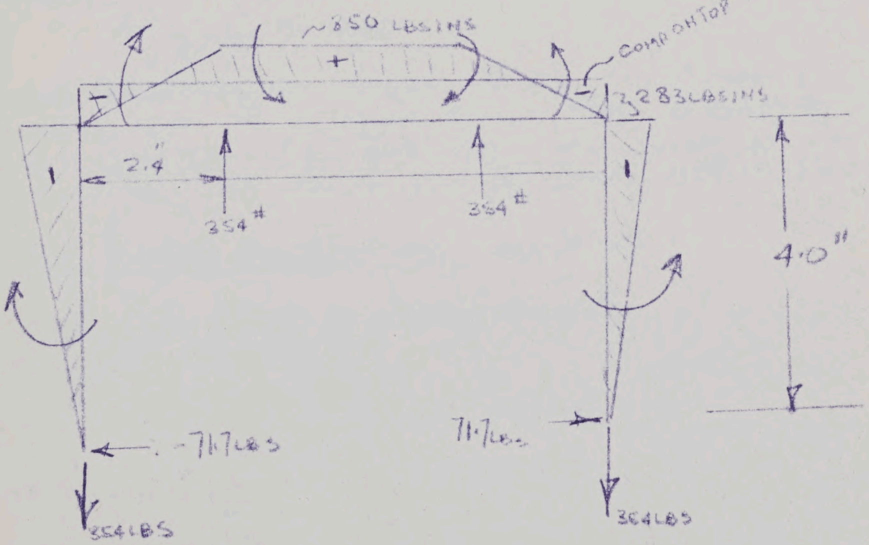
BENDING SECTION

.040 — .70" SCALED

$$Z = \frac{.040 \times .70^2}{6} = .00327 \text{ INS}^3$$

$$f_b = \frac{566}{.00327} = 173000 \text{ LBS/INS}^2.$$

IN A DISCUSSION WITH MR K. IODON IT WAS ASSUMED THAT CONSIDERING THE STIFFENER TO BE FULLY FIXED AT CORNERS WAS TOO SEVERE & THAT TAKING 1/2 FIXITY WAS MORE REALISTIC. THIS WAS DONE IN LIEU OF A STRAIN ENERGY ANALYSIS WHICH WAS CONSIDERED TO BE TOO LABORIOUS & TIME CONSUMING. THEREFORE OWING TO THIS, THE ABOVE ASSUMPTION WAS USED.



THE STIFFENER GAGE IS TO BE INCREASED TO .064"

NOTE STIFFENER ASSUMED FLAT

FORM 1319 A

A. V. ROE CANADA LIMITED MALTON - ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7-0558-38 SHEET NO. 59	
AIRCRAFT C105	GEAR BOX SEAL DRIVE	PREPARED BY A. FERENC	DATE DEC/55
		CHECKED BY	DATE

RECHECK ON B M AT 1.50 AWAY FROM END.

RECONSIDERING THE PORTION OF THE BEAM IT IS NOW OBVIOUS THAT CONSIDERING THE BEAM SIMPLY SUPPORTED AT THE ENDS WILL BE CRITICAL. THE SECTION PROPERTIES OF PAGE 57 WILL BE ASSUMED TO GO UP IN THE RATIO OF THE THICKNESSES.

$$\therefore I = .00378 \times \frac{.064}{.040} = .00615 \text{ INS}^4$$

$$f_b = \frac{318 \times .53}{.00615} = 27200 \text{ LBS/INS}^2$$

CHECK LOCAL CRIPPLING

REF BRUHN PAGE B.22

$$F_c = KE\left(T/b\right)^2 \qquad K = 3.63$$

$$F_c = 3.63 \times 15.5 \times 10^6 \,(\text{SHT } 54)\left(\frac{.064}{1.15}\right)^2 = 178500 \text{ LBS/INS}^2$$

$R_{F_{CY}}$ BY INSPECTION $>$ 2

BECAUSE OF THE HIGH VALUE OF LOCAL CRIPPLING NO ATTEMPT WAS MADE TO FIND $\bar{E}$ i.e. EFFECTIVE MODULUS

$\therefore$ F_{CY} WILL BE USED

THE PREVIOUS STRESSING OF PAGES 52-56 HAS ALREADY BEEN COVERED FOR A THINNER GAGE MTL THEREFORE WITH THE INCREASE TO .064 MTL WILL INCREASE THE R.F.'S. NO RECALCULATIONS OF R.F'S WILL BE MADE AT THIS POINT.

CHECKING OUT TONGUE OF STIFFENER REF SHT 58

$$f_b = \frac{283}{\frac{.064 \times .70^2}{6}} = \frac{283}{.00523} = 54500 \text{ LBS/INS}^2$$

$$\text{MAX SHEAR STRESS} = \frac{71.7}{.064 \times .70} = 1600 \text{ LBS/INS}^2$$

$$\text{DIRECT TENS STRESS} = \frac{354}{.064 \times .70} = 7930 \text{ LBS/INS}^2$$

NET COMP STRESS = 46570 LBS/INS^2

NET TENS STRESS = 62430 LBS/INS^2

$$\text{PRINCIPLE TENSION STRESS} = 31215 + \left(31215^2 + 1600^2\right)^{1/2} = 31215 + 31250 = 62465$$

ALLOW REF SHT 54 = 90000 LBS/INS^2 $RF = \frac{90000}{62465}$

PRINCIPLE COMP STRESS $\approx$ 46750 LBS/INS^2 (SHEAR CAN BE SEEN TO HAVE A NEGLIGIBLE EFFECT)

ALLOWABLE COMP STRESS REF SHT 56 = 63800 LBS/INS^2 1.022

SINCE LOADING CAN BE REVERSED $RF_{COMP} = \frac{63800}{62465}$

FORM 1319A

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

AIRCRAFT: C105	GEAR BOX SEAL DRIVE	REPORT NO. 7-0558-38	SHEET NO. 60
		PREPARED BY: A. FURCHE	DATE: DEC/55
		CHECKED BY	DATE

CHECK ON .064 Ti STIFFENER

VERT LOAD/ATTACHMENT = $\frac{354}{7}$ = 50.7 LBS

CRITICAL ATTACHMENT IS LOWER ONE

RESULTANT LOAD = $(50.7^2 + 71.7^2)^{1/2}$ = 87.9 LBS

RIVETS 1/8 BLIND MONEL CHERRY

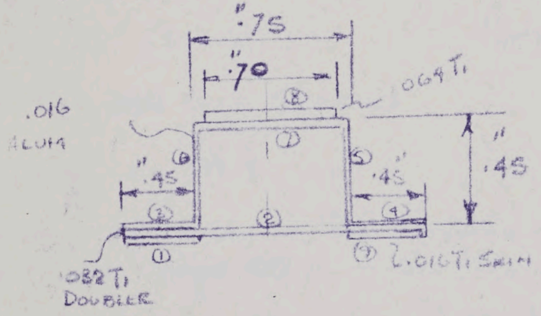
RIVET BEARING ALLOWABLE REF SHT 52 = ~~269~~ 213 LBS

R.F BY INSPECTION 72

CHECK ON ADJACENT STIFFENERS

SINCE THE ADJACENT STIFFENERS MUST BE ABLE TO CARRY THEIR OWN MOMENT PLUS THAT OF A CUT STIFFENER ITS I SHOULD BE TWICE AS GREAT AS THAT OF THE CUT STIFFENER

I OF CUT STIFFENER = .00188 INS^4 REF SHT 40



ON PAGE 43
t Ti WAS SAID TO BE BASED ON $\frac{E_{Ti}}{E_{AL}}$

IT IS FELT THAT
$t_{AL} = T_{Ti}\sqrt{\frac{E_{Ti}}{E_{AL}}} = T_{Ti} \times 1.287$

ITEM	b	t	FACTOR	AREA	y	Ay	Ay2	I cg
1 & 9	2X.45	.016	1.287	.0185	.008	.000148	NEG	NEG
2	1.618	.032	1.287	.0668	.032	.002140	.0000675	.000004
3 & 4	2X.45	.016	1.0	.0144	.064	.000920	.0000590	NEG
5 & 6	2X.418	.016	1.0	.01338	.273	.003660	.001000	.000246
7	.75	.016	1.0	.0120	.490	.005880	.002380	NEG
8	.70	.064	1.287	.0438	.530	.02320	.01230	NEG
				.16888	.213	.035948	.0162565	.000250

$I_{NA} = .0162565 + .000250 - .213^2 \times .16888 = .0165065 - .0076250 = .0088815$

I req'd = .003772 INS^4 REF SHT 44

BY INSPECTION RF $\frac{.0088815}{.003772}$ 72

FORM 1319A

A. V ROE CANADA LIMITED MALTON - ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT No. 7-0558-38 SHEET No. 71	
		PREPARED BY	DATE
AIRCRAFT: C105	HOLE FOR AIR CONDITIONING PIPE REF DRG 7-1058-1147	A. FERENC	Jan/58
		CHECKED BY	DATE

INDEX

M 1319A

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

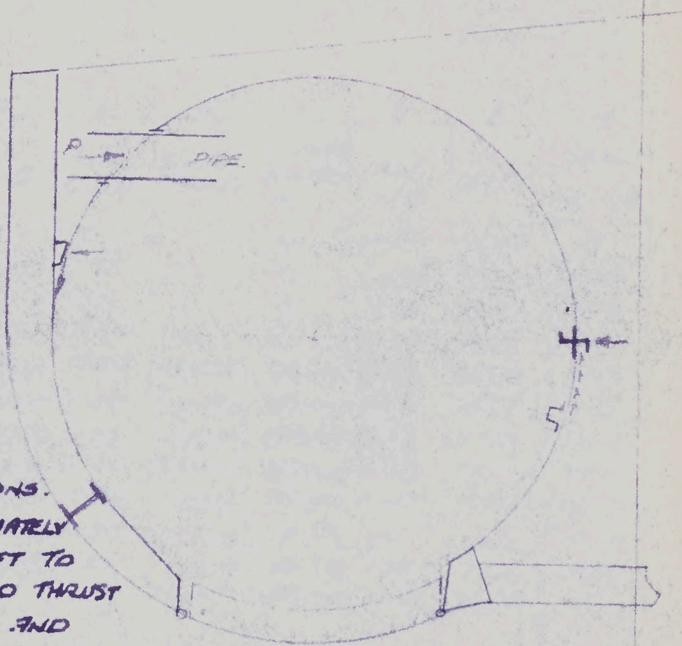
REPORT No 7-055-8-38
1. SHEET No 72

AIRCRAFT:		PREPARED BY	DATE
C.105	HOLE FOR AIR CONDITIONING PIPE.	K. J. IDDON.	16 DEC 55
		CHECKED BY	DATE

HOLE FOR AIR CONDITIONING PIPE.

DRAWING Nº 7-0158-326 ISS. 1.
HOLE ℄ STA 636·88.

1. SHROUD BENDING.



THE EFFECT OF THE HOLE PUT INTO THE SHROUD BY THE PIPE IS TO SUPERIMPOSE A LOAD 'P' UPON THE NORMAL HOOP TENSION CONDITION.

THIS LOAD P MUST BE REACTED INITIALLY BY THE INBD AND OUTBD DUCT BEAMS (ASSUME 50% ON EACH) TOGETHER WITH VERTICAL MOMENT REACTIONS.

THESE LOADS WILL ULTIMATELY TRAVEL ALONG THE AIRCRAFT TO BE REACTED BY THE ADDED THRUST ON THE ENGINE MOUNTING AND REACTIONS FROM THE PIPE AT BENDS ETC.

$$P = \frac{p.\pi D^2}{4}$$

$$= \frac{24{\cdot}85\,\pi \times 3{\cdot}0^2}{4}$$

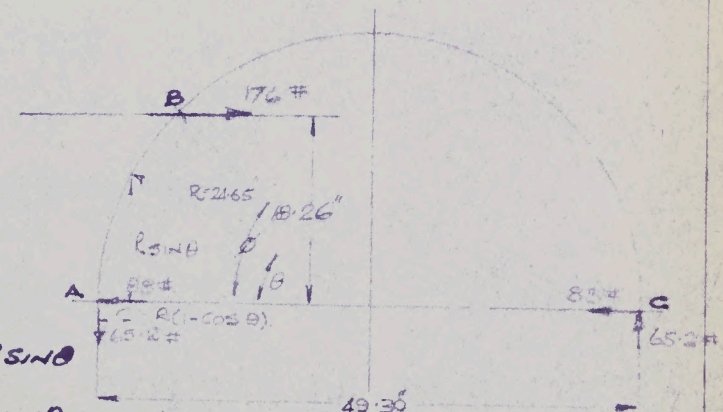
$$= 176 \text{ lb} \quad \text{AND}$$

$$R_H = \frac{176}{2}$$

$$= 88 \text{ lb}$$

$$R_V = \frac{176 \times 18{\cdot}26}{49{\cdot}30}$$

$$= 65{\cdot}2 \text{ lb}$$



$$M_{A\to B} = R_V.R(1-\cos\theta) - R_H.R\sin\theta$$

$$M_{B\to C} = R_V R(1-\cos\theta) - R_H R\sin\theta + PR(\sin\theta - \sin\phi)$$

$$\phi = \sin^{-1}\frac{18{\cdot}26}{24{\cdot}65}$$

$$= \sin^{-1}{\cdot}741 = 47° \, 48'$$

FORM 1319A

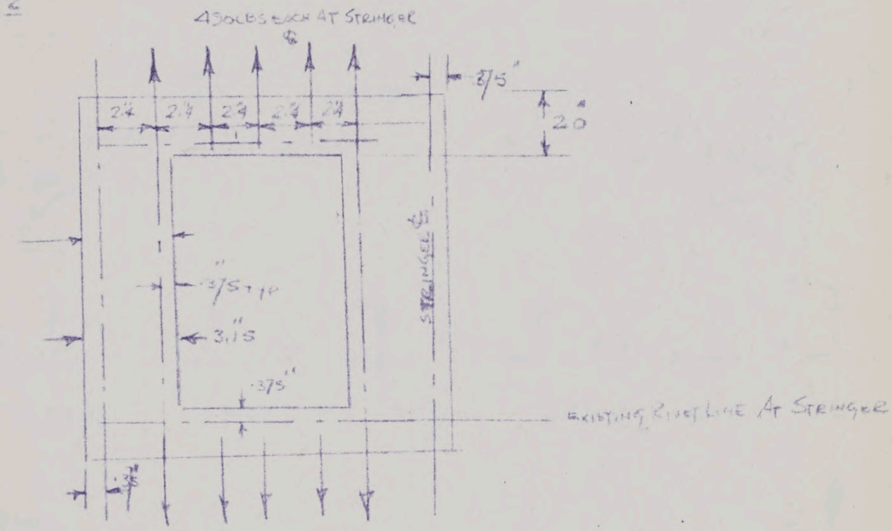
A. V. ROE CANADA LIMITED
MALTON - ONTARIO
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7-0558-38
SHEET NO. 81

AIRCRAFT:		PREPARED BY	DATE
C105	HOLE FOR AIRCONDITIONING PIPE	A. FERENC	DEC/55
		CHECKED BY	DATE

COVER PLATE (CONT'D)

CONSIDERING ITEM 2

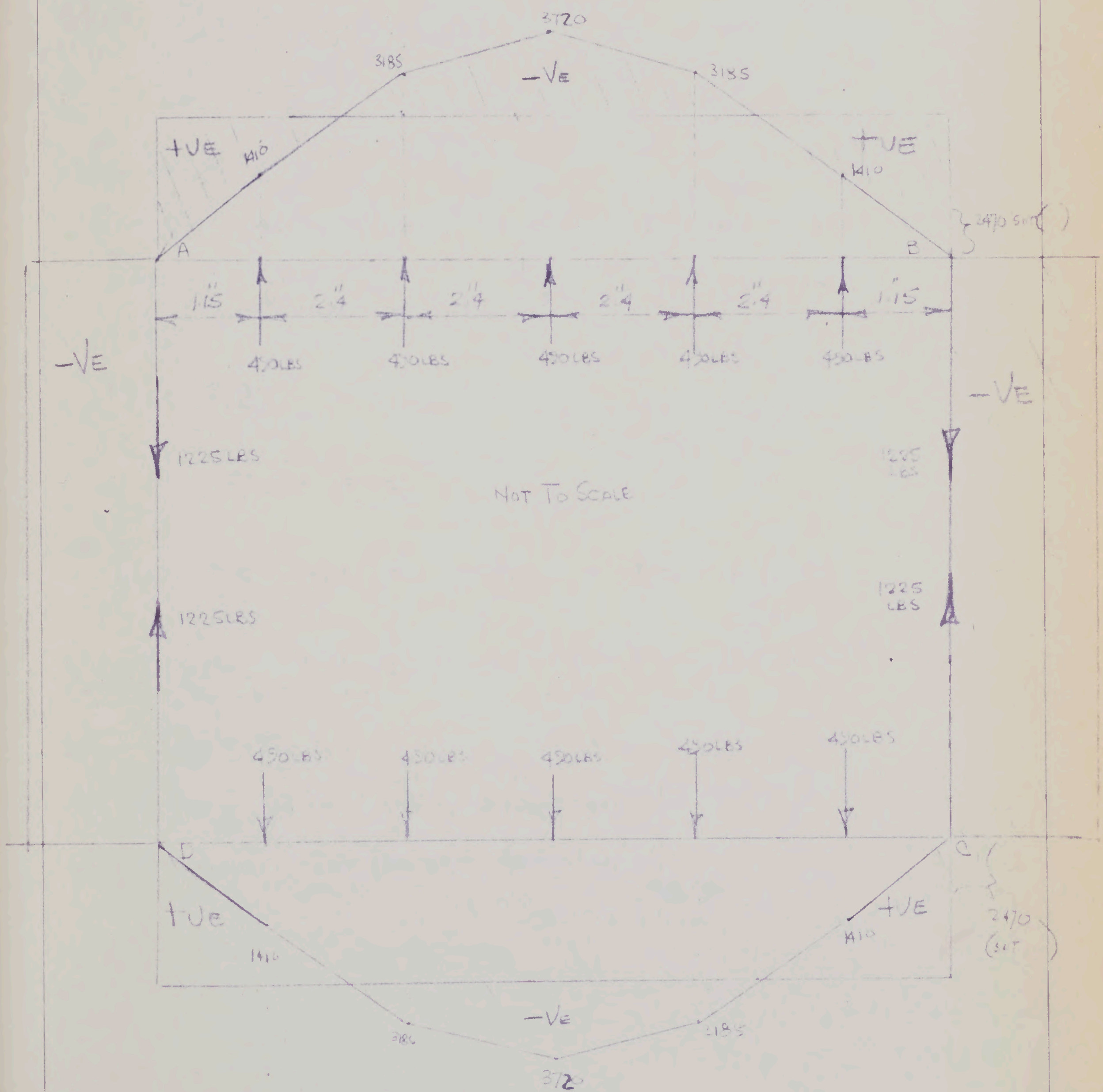


FORM 1319A

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

AIRCRAFT: C105	HOLE FOR AIRCONDITIONING PIPE	REPORT NO. 7-0553-38	SHEET NO 82
		PREPARED BY: A. Ferenc	DATE: Dec/55
		CHECKED BY	DATE

Cover Plate (Cont'd)



+Ve Moment Tension Inner Side

FORM 1319A

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

REPORT No. 7-0558-38
SHEET No. 33

AIRCRAFT		PREPARED BY	DATE
C105	HOLE FOR AIR-CONDITIONING PIPE	A. FERENC	DEC/55
		CHECKED BY	DATE

COVER PLATE (CONT'D)

$$M_o = \frac{\frac{M_x}{Z_x} - \frac{P}{2A_y}}{\frac{1}{Z_y} + \frac{1}{Z_x}}$$

USING ·040 GAGE

$$Z_x = \frac{2.0^2 \times .040}{6} = .02665 \text{ INS}^3 \qquad Z_y = \frac{3.15^2 \times .040}{6} = .0662 \text{ INS}^3$$

$$1/Z_x = 37.5/\text{INS}^3 \qquad 1/Z_y = 15.15/\text{INS}^3$$

$$A_y = 3.15 \times .040 = .126 \text{ INS}^2$$

$$\therefore M_o = \frac{\frac{3720}{.02665} - \frac{1225}{.126}}{37.5 + 15.15} = \frac{139500 - 9730}{52.65} = \frac{129770}{52.65} = 2470 \text{ LBS INS}$$

CHECK

$$f_{xx} = \frac{3720 - 2470}{.02665} = \frac{1250}{.02665} = 47000 \text{ LBS/INS}^2$$

$$f_{yy} = \frac{2470}{.0662} + \frac{1225}{.126} = 37400 + 9730 = 47130 \text{ LBS/INS}^2$$

MTL 75-S-T6

$$F_{cy} \text{ AT TEMP } 265° = 64000 \times .83 = 53100 \text{ LBS/INS}^2$$

$$F_{tu} \text{ AT TEMP} = .75 \times 72000 = 54000 \text{ LBS/INS}^2$$

$$R_F = \frac{53100}{47130} = 1.18$$

CHECK INTERRIVET BUCKLING ALONG LEG A-D

REF R.A.S. DATA SHT 02.01.05
% E OF ROOM TEMP AT [illegible] = ·96
SINCE F_{cy} AT TEMP = 53100 LBS/INS²
AS AN APPROXIMATION THE CURVES FOR 55 [illegible] MTL WILL BE USED

AT 47000 LBS/INS² $l/t = 21.2$

$$l = 21.2 \times .040 = .848$$

MAX PITCH TO BE CALLED ON DWG = ·75

FORM 1319A

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

AIRCRAFT: C105	HOLE FOR AIRCONDITIONING PIPE	REPORT No 7-0558-38	SHEET No 84
		PREPARED BY: A. FERENC	DATE: Dec/55
		CHECKED BY	DATE

COVER PLATE (CONT'D)

CHECK LOCAL CRIPPLING ALONG LEG A-D
RIVET EDGE DISTANCE = .35

$b/t = \frac{.35}{.040} = 8.77$

USING CURVE OF BRUHN PAGE B-5-5

$F_{cr} = 38755$ (ONE EDGE FREE) FOR MTL $F_{cy} = 46000$

F_{cy} FOR 75-S-T6 = 64000

AT TEMP $F_{cy} = 64000 \times .82$

ASSUMING F_{cr} GOES UP AS F_{cy}

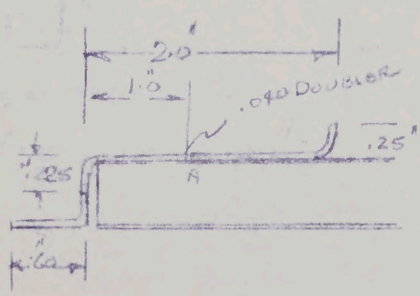
F_{cr} FOR 75-S-T6 $= \frac{.82 \times 64000}{46000} \times 38755 = 44250$

MAX COMP STRESS = 37400 − 9730 = 27670 LBS/IN^2

RF $= \frac{44250}{27670}$ 1.6

CHECK LEG A-B

MAX COMP STRESS = 47120 LBS/IN^2 SHT 83
BY INSPECTION OF SHAPE IT IS OBVIOUS THAT IT WILL FAIL IN LOCAL CRIPPLING
THE ENDS SHOULD HAVE FLANGES. THE UPPER & LOWER TOP HATS ARE TO BE REMOVED & THE DOUBLING PLATE IS TO BE EXTENDED AS SHOWN



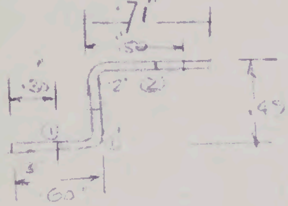
THE LOADING IS TO BE UNCHANGED ALTHOUGH THE SECTION PROPERTIES OF THE DOUBLER THRU XX HAVE INCREASED. THIS IS CONSERVATIVE. BY INSPECTION OF THE FORMULA ON PAGE 78 IT CAN BE SEEN THAT f_{xx} WILL BE DECREASED

FORM 1319A

A. V. ROE CANADA LIMITED MALTON · ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT No. 7-0558-38 SHEET No 35	
AIRCRAFT	HOLE	PREPARED BY	DATE
C105	FOR AIR CONDITIONING PIPE	A. FEBENC	
		CHECKED BY	DATE

CHECK LEG AT A-B CONT'D

ASSUMED COMP END



APPROX N.A. POSITION

$$= \frac{.25 \times .040 \times .020 + 2.0 \times .040 \times 1.0 + .41 \times .040 \times 1.33 + .60 \times .040 \times 2.3}{.25 \times .040 + 2.0 \times .040 + .41 \times .040 + .60 \times .040}$$

$$= \frac{.0002 + .08 + .0325 + .0553}{.01 + .08 + .0164 + .024} = \frac{.168}{.1304} = 1.29$$

USING NEEDHAMS FORMULA AT TEMP CONDITIONS.

$$F_{cc}\ \text{ITEM (1)} \quad \frac{.342 \times \left(53100 \times 10.15 \times 10^6\right)^{1/2}}{\left(\frac{.475}{.040}\right)^{.75}} = \frac{.342 \times 732200}{6.35} = 39400\ \text{LBS/INS}^2$$

$$F_{cc}\ \text{ITEM 2} - \frac{.366 \times 732000}{\left(\frac{.422}{.040}\right)^{.75}} = \frac{.366 \times 732000}{5.86}\quad 45700\ \text{LBS/INS}^2$$

THE ABOVE IS TOO CONSERVATIVE UP (SECTIONS ONLY)

THE BENDING COMP STRESS IS NOT UNIFORM OVER COMPLETE SECTION.

FOR ITEM 2'

$$F_{cc} = \frac{.342 \times 732000}{\left(\frac{.372}{.040}\right)^{.75}} = \frac{.342 \times 73200}{5.33} = 47200\ \text{LBS/INS}^2$$

$$\text{ITEM (1)}' = \frac{.366 \times 732000}{\left(\frac{.272}{.040}\right)} = \frac{.342 \times 73200}{4.21} = 63600\ \text{LBS/INS}^2$$

ITEM 3 REF BRUHN PAGE B-5-5

$$b/t = \frac{.30}{.040} = 7.5$$

$F_{cr} = 45600$ FOR 14TC $F_{cy} = 46000$

THIS VALUE IS CONSERVATIVE BECAUSE IT IS NOT FOR 75-S-T MATL
ASSUMING F_{cr} TO GO UP AS $\sqrt{F_{cy}}$ (REF NEEDHAMS FORMULA)
THEN FOR 75-S-T0 AT TEMP (E IS SLIGHTLY AFFECTED)

$$F_{cc} = 45600 \times \sqrt{\frac{53100}{46000}} = 49000\ \text{LBS/INS}^2$$

$$F_{cc\,AV} = \frac{47200 \times .372 + 63600 \times .272 + 49000 \times .30}{.372 + .272 + .30} = 53000\ \text{LBS/INS}^2$$

RF COVERED ON PAGE 83.

RM 1319A

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

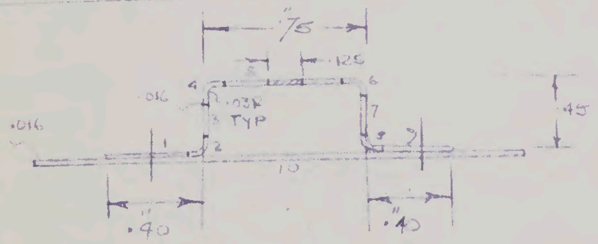
REPORT NO. 7-0558-39
SHEET NO. 86

AIRCRAFT:		PREPARED BY	DATE
C105	HOLE FOR AIR CONDITIONING PIPE	A. FERENC	DEC/55
		CHECKED BY	DATE

CHECK ATTACHMENT STRINGER TO COVER PLATE

LOAD 400 LBS SHT 76
R.F. COVERED BY SHT 75. AT LEAST 3 1/8" DIA RIVETS AT ATTACHMENT OF RING TO HAT.

CHECK BENDING OF TOP HAT AT RIVET HOLE IN CROWN



30t OF SKIN ASSUMED EFFECTIVE

Item	b	t	A	y	Ay	Ay²	Icg
1	.37	.016	.00592	.024	.000142	.0000034	NEG
2		↑	.000952	.0274	.0000261	.000000715	NEG
3	.358		.00573	.241	.00138	.000332	.0000612
4			.000952	.4386	.0000418	.0000183	NEG
5	.533		.00853	.458	.00391	.00179	NEG
6			.000952	.4386	.0000418	.0000183	NEG
7	.358		.00573	.241	.00138	.000332	.0000612
8			.000952	.0274	.0000261	.000000715	NEG
9	.37	↓	.00592	.024	.000142	.0000034	NEG
10	.96	.016	.01535	.008	.0001228	.000000983	NEG
			.051988	.139	.0072126	.00249813	.0001224

$$I_{cg} = .00249813 + .0001224 - .139^2 \times .05198 = .00262213 - .001005$$

$$= .0016173$$

$$\text{Say } .00162 \text{ INS}^4$$

TOTAL AREA INCLUDING 2.4" OF SKIN

$= .051988 + 1.44 \times .016 = .075038 \text{ INS}^2$

Say .075 INS²

CHECK BENDING OF TOP HAT AT RIVET HOLE IN CROWN

FOR LOADS REF SHT 74

TOP COMP

M = 220 LBS INS
P = 1375 LBS

$$f_C = \frac{220 \times \cdot 327}{\cdot 00162} - \frac{1375}{\cdot 075}$$

$= 44000 - 18350 = 25650$ LBS/INS^2

$f_{ALL} = 26800$ LBS/INS^2

$$f_T = \frac{220 \times \cdot 139}{\cdot 00162} + \frac{1375}{\cdot 075}$$

$= 18850 + 18350 = 37200$ LBS/INS^2

$f_{ALL} = 50700$

TOP TENS

M = 220 LBS INS
P = 1575 LBS

$$f_C = \frac{220 \times \cdot 139}{\cdot 00162} - \frac{1575}{\cdot 075}$$

$= 18850 - 21000 = -2150$ LBS/INS^2

$$f_T = \frac{220 \times \cdot 327}{\cdot 00162} + \frac{1575}{\cdot 075}$$

$= 44000 + 21000 = 65000$ LBS/INS^2

$f_{ALL} = 50700$

$$R_F = \frac{50700}{65000} \quad \cdot 78$$

THE MOMENT USED MAY BE LOWER AT THE PLACE IN QUESTION
THEREFORE VALUES OF PAGE 74 ARE REPLOTTED

SCALED DIST FROM HOLE ℄ TO 1ST RIVET HOLE THRU STRINGER AT
BOTTOM = 6" AT TOP 7.2"

R = 24.65"

$180° = \frac{\pi}{2} \times 24.65 \times 2$" INS ALONG CIRCUMFERENCE

$1° = \frac{\pi}{180} \times 24.65 = \cdot 43$"

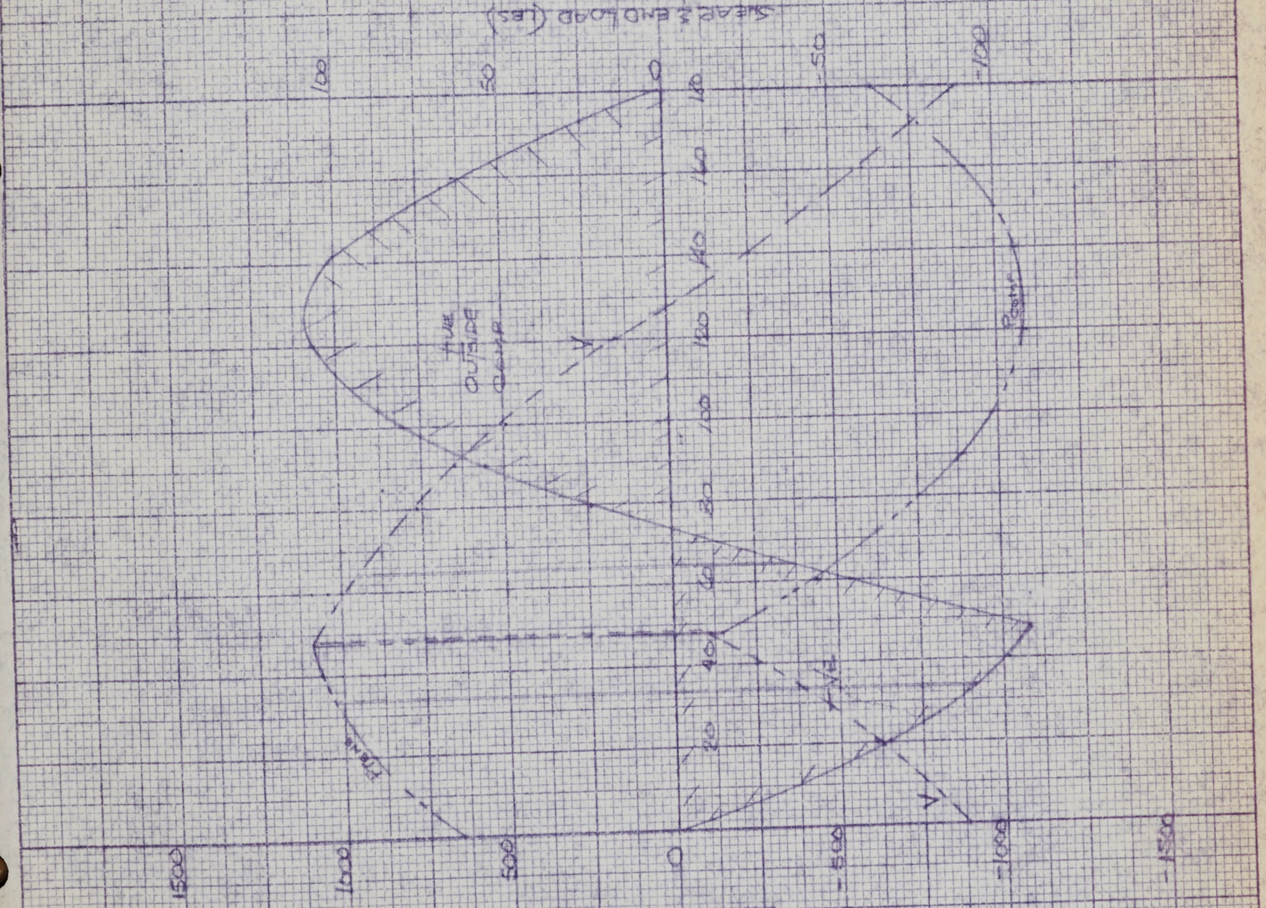
$\therefore 6" = \frac{6}{\cdot 43} = 13.95°$

$7.2" = \frac{7.2}{\cdot 43} = 16.75°$

ANGLE AT LOWER RIVET $= 47.8 - 13.95 = 33.85°$
ANGLE AT UPPER RIVET $= 47.8 + 16.75 = 64.55°$

PLOT OF SHEAR, END LOAD & B.M

REF PAGES 73 & 74



A. V. ROE CANADA LIMITED MALTON - ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**	REPORT NO. 7-0558-38 SHEET NO. 89	
AIRCRAFT: C105 — HOLE FOR AIR CONDITIONING PIPE	PREPARED BY: A. FERENC	DATE: DEC/55
	CHECKED BY:	DATE:

CHECK BENDING OF TOP HAT AT RIVET HOLE IN CROWN (CONT'D)

HOOP TENSION LOAD / PITCH = 1475 LBS REF SHT
FOR MOMENTS & END LOADS REF SHT 88

AT 33.85°

MOMENT / STRINGER = $\frac{900}{5}$ = 180 LBS INS (TENS CROWN)

TENS LOAD = $1475 + \frac{102}{5}$ = 1495 LBS

$f_{bc} = \frac{180 \times .139}{.00162}$ = 15450 LBS/INS² COMP

$f_{bt} = \frac{180 \times .327}{.00162}$ = 36300 LBS/INS² (TENS)

DIRECT TENS LOAD = $\frac{1495}{.075}$ = 19950 LBS/INS² (TENS)

TENSILE STRESS AT SKIN = 19950 − 15450 = 4500 LBS/INS²

TENSILE STRESS AT CROWN = 36300 + 19950 = 56250 LBS/INS²

f_{tu} ALLOW = 50700

AT 64.55°

= $\frac{375}{5}$ = 75 LBS INS (TENS CROWN)

TENS LOAD = $1475 - \frac{50}{5}$ = 1465 LBS

$f_{bc} = \frac{85 \times .139}{.00162}$ = 7300 LBS/INS²

$f_{bt} = \frac{85 \times .327}{.00162}$ = 17150 LBS/INS²

DIRECT TENS STRESS = $\frac{1465}{.075}$ = 19550 LBS/INS²

TENSILE STRESS AT SKIN = 19950 − 7300 = 12650 LBS/INS²

TENSILE STRESS AT CROWN = 17150 + 19950 = 37000 LBS/INS²

RF = $\frac{50700}{56250}$ .902

ACTUALLY LOWER LIMIT OF STRINGER GAGE WAS USED
USING NOMINAL GAGE .018

ASSUMING THAT ONLY THE THICKNESS OF ITEMS 1-9 OF SHT 86 IS INCREASED

$\therefore \Sigma A = (.05198 - .01535) \times \frac{.018}{.016} + .01535 = .04125 + .01535 = .0566$ INS²

ΣA INCLUDING 2.9" OF SKIN = .0566 + 1.49 × .016 = .0797 INS²

$\Sigma A_y = (.007126 - .0001228) \times \frac{.018}{.016} + .0001228 = .0082$ INS²

$\bar{y} = \frac{.0082}{.0566} = .145"$

$\Sigma A y^2 = (.00249813 - .000000983) \times \frac{.018}{.016} + .000000983 = .00280$

I_{cg} .000124 × $\frac{.018}{.016}$ = .0001395 =

I_{cg} = .0028 + .0001395 − .145² × .0566 = .002395 − .00119 = .0017495 SAY .00175 INS⁴

$f_{T MAX} = \frac{180 \times (.468 - .145)}{.00175} + \frac{1495}{.0797}$ = 33000 + 18750 = 51750 1.015

F_{TU} ALL AT 265°F = 70000 × .75 = 52500 LBS/INS² RF = $\frac{52500}{51750}$

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

REPORT NO. 7-0558-38
SHEET NO. 90

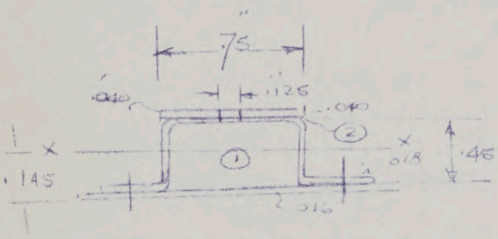
AIRCRAFT:		PREPARED BY	DATE
C105	HOLE FOR AIRCONDITIONING PIPE	A. FERRIS	DEC/55
		CHECKED BY	DATE

CHECK BENDING OF TOP HAT THRU PIPE HOLE IN TOP (CONT'D)

THE R.F AT LOWER LIMIT = .902
" " NOMINAL THICKNESS = 1.015

MR K. LODON WAS SATISFIED WITH THE ABOVE R.F'S ∴ STRINGERS WILL NOT BE STRENGTHENED LOCALLY.

CHECK STIFFNESS OF STIFFENER ADJACENT TO HOLE AT HOLE ℄



ITEM	b	t	A	y	Ay	Ay²	Icg.
1	SEE SHT		.0566	0	0	0	.00175
2	.625	.040	.0250	.341	.00853	.00291	NEG
			.0816	.1045	.00853	.00291	.00175

$I_{cg} = .00291 + .00175 - .0816 \times .1045^2 = .00466 - .000891 = .003769 \text{ INS}^4$

NOTE THE 2 Ti DOUBLERS HAVE NOT BEEN INCLUDED
SEE SHT 95

$I = .00594 \text{ INS}^4$

TO REPLACE STIFFNESS I SHOULD BE $2\frac{1}{2}$ TIMES AS GREAT AS ORIGINAL I

$\frac{.0065}{.00175} > 2\frac{1}{2}$

STIFFNESS IS REPLACED

ORM 1319A

ATTACHMENT OF TITANIUM DOUBLER TO SHROUD SKIN (CONT'D)

THERMAL STRESSES

AT TOP & BOTTOM IN LONGITUDINAL DIRECTION.

$$5.0\times10^{-6}(265-60)\,l + \frac{P_{Ti}\,l}{1.5\times.018\times15.5\times10^{6}} = 12.9\times10^{-6}(265-60)\,l - \frac{P_{AL}\,l}{1.5\times.018\times10.5\times10^{6}}$$

$$1025\times10^{-6} + \frac{P_{Ti}}{.418\times10^{6}} = 2640\times10^{-6} - \frac{P_{AL}}{.284\times10^{6}}$$

$$1025\times10^{-6} + 2.39\,P_{Ti}\times10^{-6} = 2640\times10^{-6} - 3.52\times10^{-6}P_{AL}$$

$$P_{AL} = P_{Ti} = P$$

$$-1615\times10^{-6} = -5.91\times10^{-6}\,P$$

$$P = \frac{1615}{5.91} = 273\text{ LBS}$$

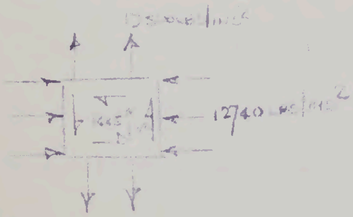
$$\delta = \left[5.0\times10^{-6}(205) + \frac{273}{1.5\times.018\times15.5\times10^{6}}\right]13.2$$

$$\delta = 13520\times10^{-6} + 8600\times10^{-6} = .022120\text{ INS}$$

$$\text{COMP STRESS IN AL} = \frac{273}{1.5\times.018} = 10120\text{ LBS/INS}^2$$

CHECK PRINCIPLE STRESSES IN AL

CRITICAL LOADS ARE AT $\theta = 73.4°$ REF SHT 91



PRINCIPLE STRESSES

$$\frac{19500-12740}{2} \pm \left(\left(\frac{19500+12740}{2}\right)^2 + 1445^2\right)^{1/2}$$

$$3383 \pm 16380$$

PRINCIPLE TENS = 19763 LBS/INS²

PRINCIPLE COMP = 12997

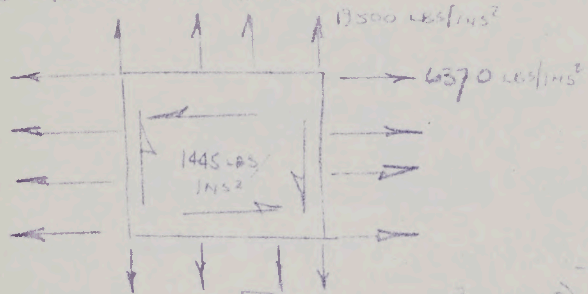
PRINCIPLE SHEAR = 16380 LBS/INS²

BY INSPECTION RF

A. V. ROE CANADA LIMITED MALTON · ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7-0558-38 SHEET NO. 34	
AIRCRAFT:		PREPARED BY	DATE
C105	Hole For AirConditioning Pipe	A. Ferenc	Dec/55
		CHECKED BY	DATE

Attachment Of Titanium Doubler To Shroud Skin (Cont'd)

Principle Stresses In Titanium



$$\frac{19500+6370}{2} \pm \left[\left(\frac{19500-6370}{2}\right)^2 + 1445^2\right]^{1/2}$$

12935 ± 6710

Max Principle Tens = 19645 LBS/INS²

Principle Shear Stress = 6710

By Inspection RF

72

Check Rivet Attachments

In An Attempt To Bring Up The Stiffness Of Section At The Doubler-Skin Splice The Rivets In Shear Will Be Worked To The Tension Capacity Of The Skin. This Load Is Greater Than That From Thermal Flexural & Hoop Tension Loads At The Top & Bottom Portions Of Splice

Ftu Of .018 AL At 265°F = 70000 x .75 = 52500 LB/INS²

Max Skin Tension Load/Pitch = 52500 x 2.4 x .018 = 2270 LBS

Present Rivets/Pitch Are 4 1/8" & 4 3/32 Rivets

Capacity Of 4 1/8 Rivets Ref SH 75 = 227 x 4 = 908 LBS

Allowable Bearing 3/32 In .018 75-S-T6 At 265°F = 173 x 146 x .82 = 219

" Shear = .754 x 203 x .84 = 168 LBS

Capacity Of 3/32 Rivets = 4 x 168 = 672 LBS

Total Capacity Of Rivets = 672 + 908 = 1580 LBS

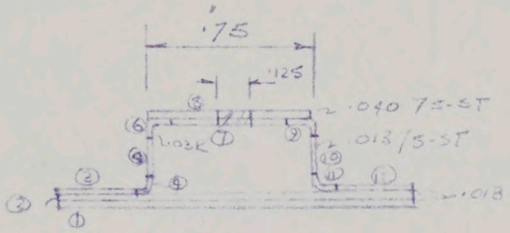
Doubler Therefore Is To Be Extended So That Lap Is 1/2" Instead Of Present .75 At Top & Bottom & Twice The Same Number & Type Of Rivets Are Included. ∴ Capacity = 2 x 1580 = 3160

$RF = \frac{3160}{2270}$ 1.39

FORM 1319A

CHECK STIFFENER ADJACENT TO HOLE AT HOLE ℄

REF PAGE 86 FOR DIMS



30t OF DOUBLER ASSUMED EFFECTIVE

AREA CONVERSION FACTOR OF Ti TO AL WILL BE BASED ON $\sqrt{E}$

IE $A_{AL} = A_{Ti}\sqrt{\dfrac{E_{Ti}}{E_{AL}}} = A_{Ti} \times \left(\dfrac{15.5\times10^6}{10.5\times10^6}\right)^{1/2} = A_{Ti} \times 1.215$

ITEM	b	t	A	AREA FACTOR	AREA	y	Ay	Ay²	ICG
1	1.08	.018	.01942	1.215	.0238	.009	.000214	NEG	NEG
2	1.08	.018	.01942	1.215	.0238	.027	.000617	.0000166	NEG
3	.352	.018	.00633	1.0	.00633	.045	.000285	.0000128	NEG
4		.018	.00109	1.0	.00109	.059	.0000643	.0000038	NEG
5	.354	.018	.00638	↓	.00638	.256	.001635	.000418	.0000665
6		.018	.00109		.00109	.4623	.000504	.000233	NEG
7	.529	.018	.00954		.00954	.477	.00455	.00217	NEG
8	.625	.040	.0250		.0250	.506	.01262	.00640	NEG
9		.018	.00109		.00109	.4623	.000504	.000233	NEG
10	.354	.018	.00638		.00638	.256	.001635	.000418	.0000665
11		.018	.00109	↓	.00109	.059	.0000643	.0000038	NEG
12	.352	.018	.00633	1.0	.00633	.045	.000285	.0000128	NEG
Σ			—		.1192	.205	.0229776	.0109218	.000133

$I_{CG} = .0109218 + .000133 - .1192 \times .205^2 = .0110548 - .00497 = .0063548 \text{ INS}^4$

FROM PAGE 87 M = 220 × 2.5 = 550 INS LBS TENS IN Ti DOUBLERS
P = 1575 × 2.5 = 3940 TENS

MAX TENS STRESS IN DOUBLERS $= \dfrac{550 \times .205}{.00635} + \dfrac{3940}{.1192} = 17700 + 33100 = 51300$ LBS/INS² TENS

MAX TENS STRESS IN LEGS OF STRINGER $= \dfrac{550 \times .187}{.00635} + 33100 = 48050$ LBS/INS² TENS

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

REPORT NO. 7-0558-38
SHEET NO. 96

AIRCRAFT:		PREPARED BY	DATE
C105	Hole for Air Conditioning Pipe	A. Ferenc	Dec/55
		CHECKED BY	DATE

CHECK STIFFENER ADJACENT TO HOLE

THERMAL STRESSES

1.5 WIDTH OF T_1 WITH STRINGER
REF SHT 92

$$5.0\times10^{-6}(265-60)l + \frac{P_{T_1}\,l}{1.5\times.036\times15.5\times10^6} = 12.9\times10^{-6}(265-60)l - \frac{P_{AL}}{.0393\times10.5\times10^6}$$

$$1025\times10^{-6} + \frac{P_{T1}}{.838\times10^6} = 2640\times10^{-6} - \frac{P_{AL}}{.4025\times10^6}$$

$$1025\times10^{-6} + 1.195\times10^{-6}P_{T1} = 2640\times10^{-6} - 2.48\times10^{-6}P_{AL}$$

$$P_{T1} = P_{AL} = P$$

$$-1615\times10^{-6} = -3.675P\times10^{-6}$$

$$P = \frac{1615}{3.675} = 440\text{ LBS}$$

THERMAL STRESS IN AL $= \frac{440}{.0393} = 11200$ LBS/INS^2 COMP

THERMAL STRESS IN $T_1 = \frac{440}{1.5\times.036} = 8150$ LBS/INS^2 TENS

MAX TENSILE STRESS IN $T_1 = 51300$ (SHT 95) $+ 8150 = 59450$ LBS/INS^2

ALLOW AT 265° $= .75\times120000 = 90000$ LBS/INS^2

RF $= \frac{90000}{59450}$ 1.51

CHECK RIVETS

SHEAR $= \frac{105}{2} = 52.5$ LBS

$$q = \frac{V}{I}\int y\,dA = \frac{52.5}{.0063543}(.187\times.0476) = 73.5\text{ LBS/INS}$$

RIVETS/INS 4 OFF 3/32 CAPACITY REF SHT 94 = 672 LBS

RF BY INSPECTION 72

THE THERMAL LOAD OF 440 LBS IS ASSUMED TAKEN IN LOCALLY AT THE CORNERS. RIVETING IS ADEQUATE BY INSPECTION.

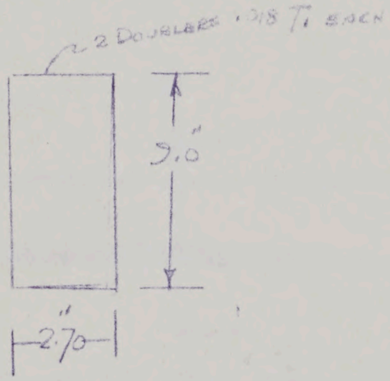
FORM 1319A

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

AIRCRAFT: C105	HOLE FOR AIR CONDITIONING PIPE	REPORT No 7-0553-38	SHEET No 97
		PREPARED BY: A. FERENC	DATE: DEC/55
		CHECKED BY:	DATE:

CHECK PRESSURE LOADS ON TITANIUM PLATE

IDEALIZED PLATE DIMS



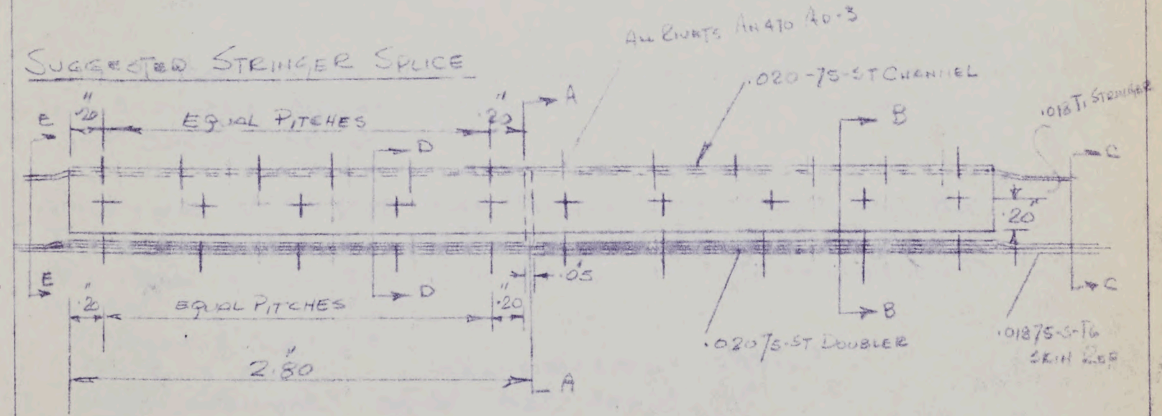
NORMAL PRESSURE = 24.85 LBS/INS^2

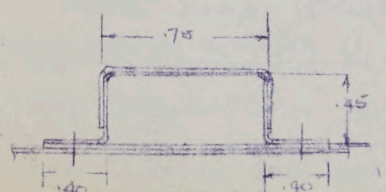
REF RAeS DATA SHT 02.09.01

$$\frac{P}{E}\left(b/t\right)^4 = \frac{24.85}{16.5\times10^6}\left(\frac{2.70}{.036}\right)^4 = 5.26$$

BY INSPECTION OF THE CURVES IT CAN BE SEEN THAT THE STRESSES WILL BE NEGLIGIBLE.

SUGGESTED STRINGER SPLICE





SECTION D-D

SECTION B-B SAME EXCEPT TOP HAT IS Ti

SECTION A-A IDENT EXCEPT TOP HAT IS NOT INCLUDED

FOR SECTION C-C & E-E REF SHT 86 & 89

NOTE ONLY AT CC TOP HAT IS TITANIUM

NOTE SHTS 97-113 ARE OBSOLETE FOR DETAILED STRESSING OF SPLICE REF SHTS 120-130

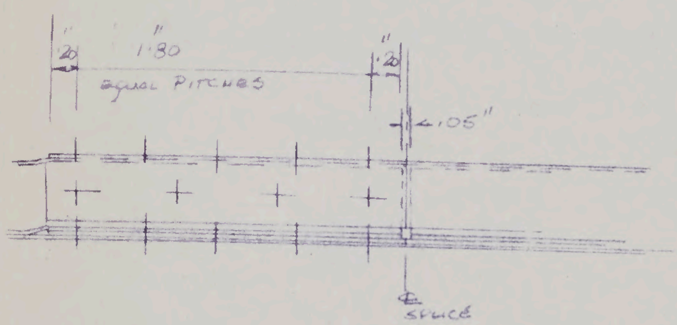
FORM 1319A

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

REPORT No. 7-0558-38
26 SHEET No. 98

AIRCRAFT:		PREPARED BY	DATE
C105	HOLE FOR AIR CONDITIONING DOOR	A. FERENC	DEC/55
		CHECKED BY	DATE

SUGGESTED STRINGER SPLICE CONT'D



THE STRINGER SPLICE WILL BE STRESSED TO MAXIMUM LOADS OF PAGE 88 PLUS THE ASSOCIATED HOOP TENSION LOAD OF 1475 LBS REF SHT 74

THE TWO MAX LOADING CONDITIONS ARE:

1)

M = 220 LBS INS (SKIN TENS)
P = 1475 + 110.5/5 = 1496 LBS TENS
V = 21 LBS INTO RING

2)

M = 220 LBS INS (SKIN COMP)
P = 1475 − 21 = 1464 LBS TENS
V = 0

BY INSPECTION OF THE LOADING CONDITIONS IT CAN BE SEEN THAT THE STRINGER SPLICE DIMENSIONS & RIVETTING CAN BE REDUCED TO THAT SHOWN ABOVE.

SINCE SKIN IS NOT CUT HOOP TENSION LOADS IN SKIN WILL BE TRANSFERRED BY SKIN IN TENSION. THE PROPORTION CARRIED BY SKIN & STRINGER WILL BE ASSUMED IN PROPORTION TO THE AREAS.

STRINGER PITCH = 2.4"
SKIN t = .016   ∴ AREA = 2.4 × .016 = .0384 INS²
AREA STRINGER = .04125 INS² SHT 89
TOTAL AREA = .04125 + .0384 = .07965 INS²

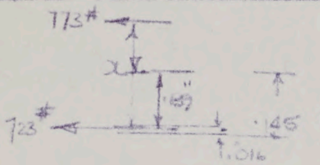
CONSIDERING LOADING 1)

TENS LOAD IN SKIN = $\frac{.0384}{.07965}$ × 1496 = 723 LBS

END LOAD IN STRINGER = 1496 − 723 = 773 LBS TENS
THESE ARE ASSUMED APPLIED AT CG OF SKIN & STRINGER.

THE MOMENT LOAD IS TAKEN AS A COUPLE AT TOP & BOTTOM
= 220/.45 = 490 LBS

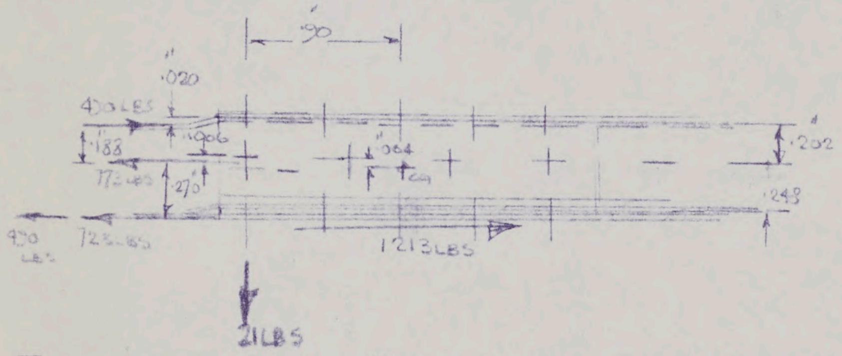
SUGGESTED STRINGER SPLICE (CONT'D)



$773(x-) = 723 \times 139$

$773x = 100500$

$x = \frac{100500}{773} = .1315"$

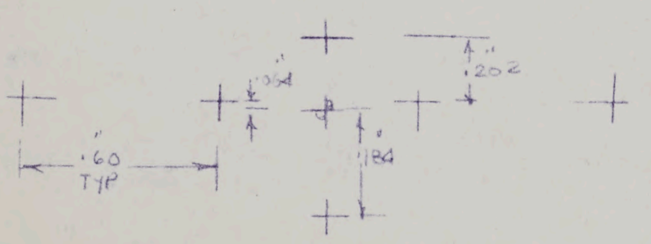


THE 1213 LB LOAD WILL REMAIN IN THE SKIN. THE REMAINDER OF LOAD IN STRINGER WILL BE TAKEN BY SKIN & DOUBLER CHANNEL WHICH WILL BE SPREAD OUT BY RIVET GROUP.

RIVET CG = $\frac{5 \times .45 + 8 \times .248}{23} = .184"$

DIST FROM RIVET LINE = .248 - .184 = .064

THE UPPER & LOWER RIVETS ARE EACH CONSIDERED AS ONE RIVET GROUP AS SHOWN BELOW.



$\Sigma AR^2 = 5 \times .286^2 + 10 \times .184^2 + 4 \times (.064^2 + .90^2) + 4(.064^2 + .30^2)$

$= .409 + .1475 + 3.25 + .376 = 4.1825$

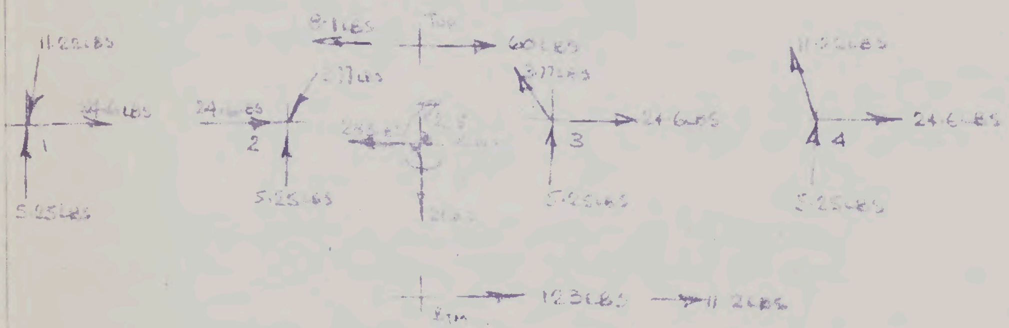
A. V. ROE CANADA LIMITED
MALTON – ONTARIO
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C105	Hole for Air Conditioning Pipe	REPORT NO. 7-0558-38	SHEET NO. 100
		PREPARED BY: A. Fergus	DATE: Dec/55
		CHECKED BY:	DATE:

Suggested Stringer Splice (cont'd)

Loads at Rivet C.G.

Tension = 773 − 490 = 283 lbs
Shear = 21 lbs
Moment = 490 × ·182 − 773 × ·058 − 21 × ·90
= 89.2 − 44.8 − 18.9 = 25.5 lbs ins



Assumptions: (1) All End Load Is Taken Equally By All Rivets In Splice
(2) The Moment Load Is Taken By All Rivets In Splice. The Rivets Thru Crown Of Stringer & Thru Skin & Doubler Are Considered As One Rivet
(3) Direct Shear Taken By Rivets Thru Upright Legs Only
(4) Portion Is Considered To Be Flat

Moment Loads

At Top = $\frac{5 \times ·266}{4.1825} \times 25.5$ = 8.1 lbs

At Bottom = $\frac{10 \times ·184}{4.1825} \times 25.5$ = 11.2 lbs

At ① & ④ = $\frac{2 \times ·923}{4.1825} \times 25.5$ = 11.22 lbs

At ② & ③ = $\frac{2 \times ·31}{4.1825}$ = 3.77 lbs

SUGGESTED STRINGER SPLICE
LOADING CONDITION (2)

SHEAR STRESS $= \frac{58.2}{2 \times .422 \times .020 \text{(SHT 31)}} = 3470 \text{ LBS/INS}^2$

f_s MAX AT TOP (CROWN) $= \sqrt{3470^2 + \left(\frac{48800}{2}\right)^2} = 24650 \text{ LBS/INS}^2$

MAX PRINCIPLE TENS $= 24650 + \frac{48800}{2} = 49050 \text{ LBS/INS}^2$

ALLOWABLE S.S AT TEMP $= 42000 \times .84 = 35300 \text{ LBS/INS}^2$

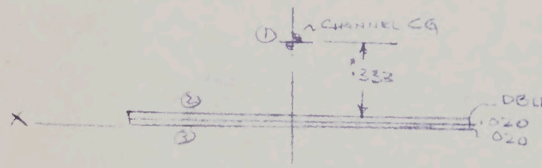
$RF_S = \frac{35300}{24650}$ 1.43

$F_{TU} = 52500 \text{ LBS/INS}^2$ REF SHT 94

$RF_{TENS} = \frac{52500}{49050}$ 1.07

FOR CONTINUITY THE STIFFNESS THRU SPLICE ℄ SHOULD BE $\geq$ STIFFNESS OF UNSPLICED STRINGER SKIN COMBINATION

SECTION PROPERTIES THRU SPLICE ℄



REF SHTS 97, 103, 99

30T OF SKIN & DBLR USED

ITEM	b	t	A	y	Ay	Ay²	I_{CG}
1			.0297	.373	.01108	.00413	.000524
2	1.2	.020	.024	.030	.00072	.0000216	NEG
3	1.2	*.020	.024	.010	.00024	.0000024	NEG
			.0777	.1432	.011166	.004154	.000524

$I_{CG} = .004154 + .000524 - .0777 \times .1432^2 = .004678 - .00159 = .003088 \text{ INS}^4$

I_{CG} REF SHT 89 $= .00175 \text{ INS}^4$

∴ CONTINUITY OF STIFFNESS MAINTAINED.

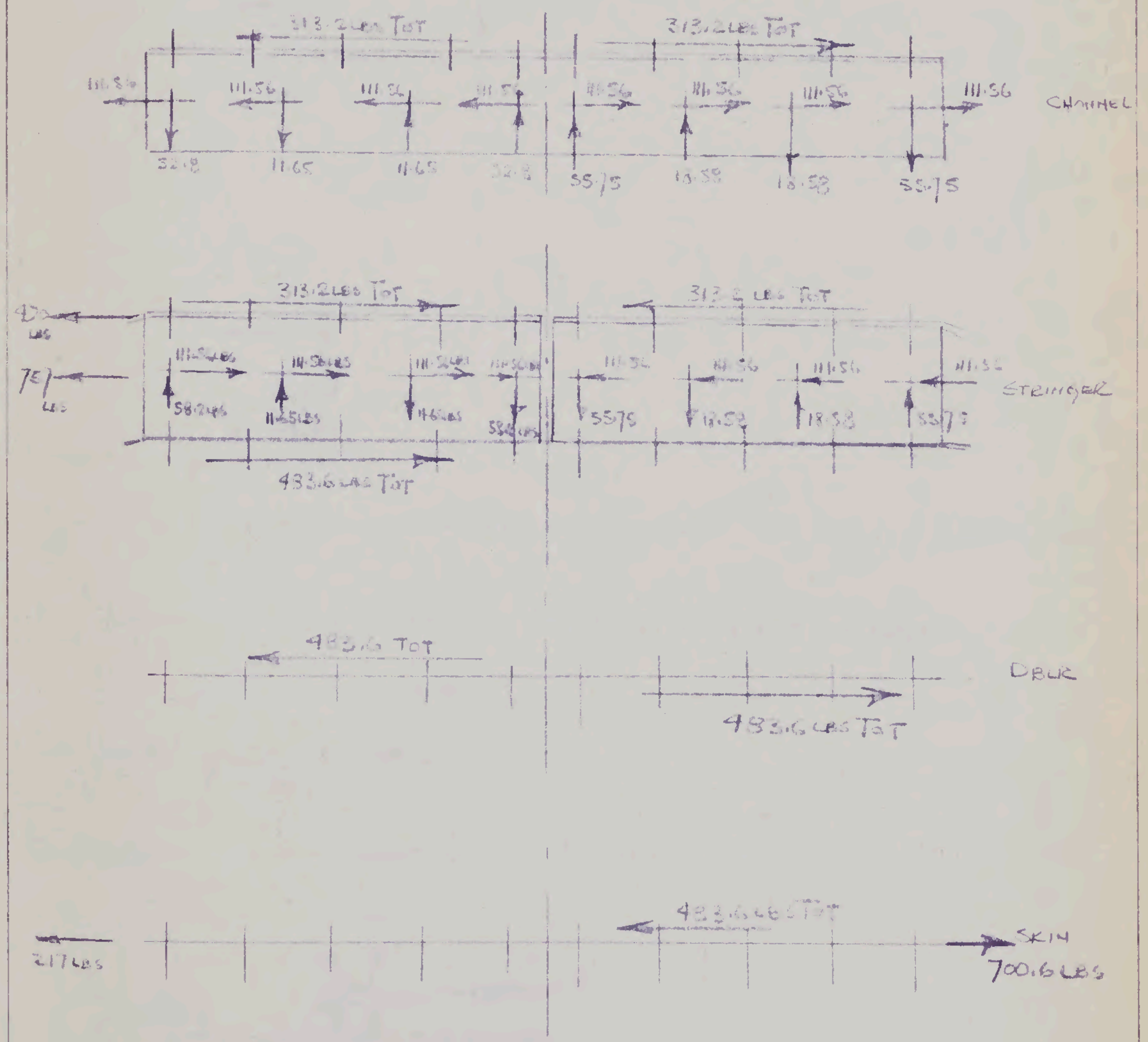
REMAINDER OF SPLICE PASSED BY INSPECTION SEE FOLLOWING SHT

* SKIN IS .018T BY INSPECTION IT CAN BE SEEN WITHOUT MAKING THE CORRECTION THAT I WILL BE > .00175 INS⁴

AVRO AIRCRAFT LIMITED
MALTON ONTARIO
TECHNICAL DEPARTMENT

REPORT NO	7-0558-38	
40 SHEET NO	112	
AIRCRAFT: C105	HOLE FOR AIRCONDITIONING PIPE	
PREPARED BY	A. FERENC	DATE JAN/55
CHECKED BY		DATE

LOADING ON ITEMS AT SPLICE FOR LOADING CONDITION (2)
FOR LOADING CONDITION (1) SEE SHT 106



NOT TO SCALE
ALL LOADING SHOWN
IS IN LBS WT

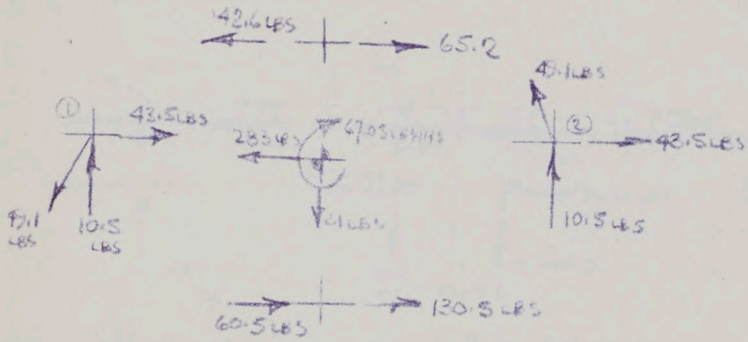
A. V. ROE CANADA LIMITED
MALTON - ONTARIO
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C105	HOLE FOR AIR CONDITIONING PIPE.	REPORT NO. 7-0558-08
		SHEET NO. 121
		PREPARED BY: A. FERENS — DATE: JAN/55
		CHECKED BY: — DATE:

SUGGESTED STRINGER SPLICE

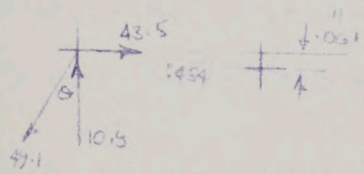
LOADING CASE 1

LOADS AT C.G. & REACTIONS



RESOLVING LOADS IN HORIZ & VERT

AT ①

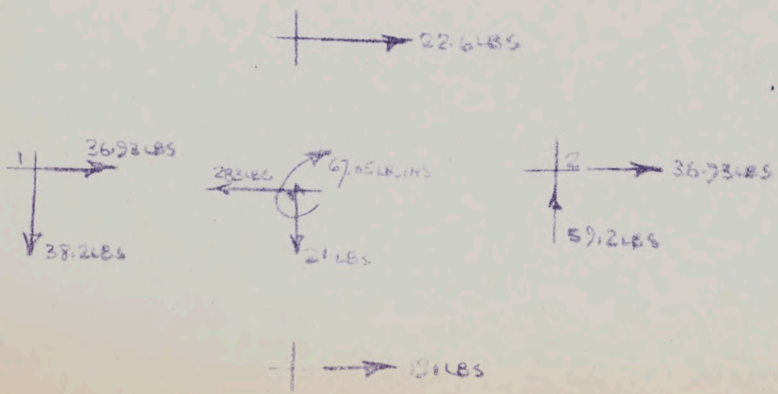


$\theta = \sin^{-1} \frac{.061}{.454} = 7.72^\circ$

HORIZ LOAD = 43.5 − 49.1 Sin 7.72° = 43.5 − 6.57 = 36.93 LBS →

VERT = 49.1 Cos 7.72° − 10.5 = 48.7 − 10.5 = 38.2 LBS ↓

AT 2 HORIZ LOAD = 43.5 − 6.57 = 36.93 LBS →
VERT = 48.7 + 10.5 = 59.2 LBS ↑

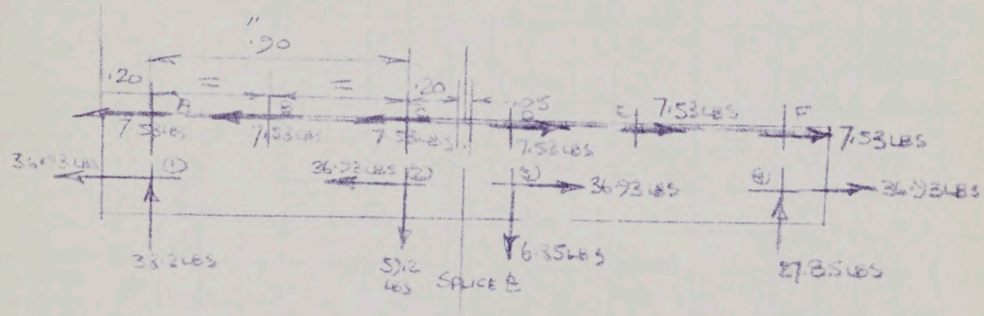


FORM 1319A

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

AIRCRAFT: C/05	HOLE FOR AIR CONDITIONING PIPE	REPORT No. 7-0558-28	SHEET No. 122
		PREPARED BY: A. FERENC	DATE: JAN/55
		CHECKED BY	DATE

SUGGESTED STRINGER SPLICE
LOADING CASE I
LOADS & REACTIONS ON SPLICE CHANNEL



MOMENT = 1.80 × 38.2 − .90 × 59.2 = 68.8 − 53.2 = 15.6 LBS INS

$\Sigma AR^2 = .45^2 + .45^2 = .405$

VERT REACTION NET AT 3 = $\overline{10.5} + \frac{15.6}{.90}$ = 17.35 − 10.5 = 6.85 LBS

VERT REACTION NET AT 4 = 10.5 + 17.35 = 27.85 LBS

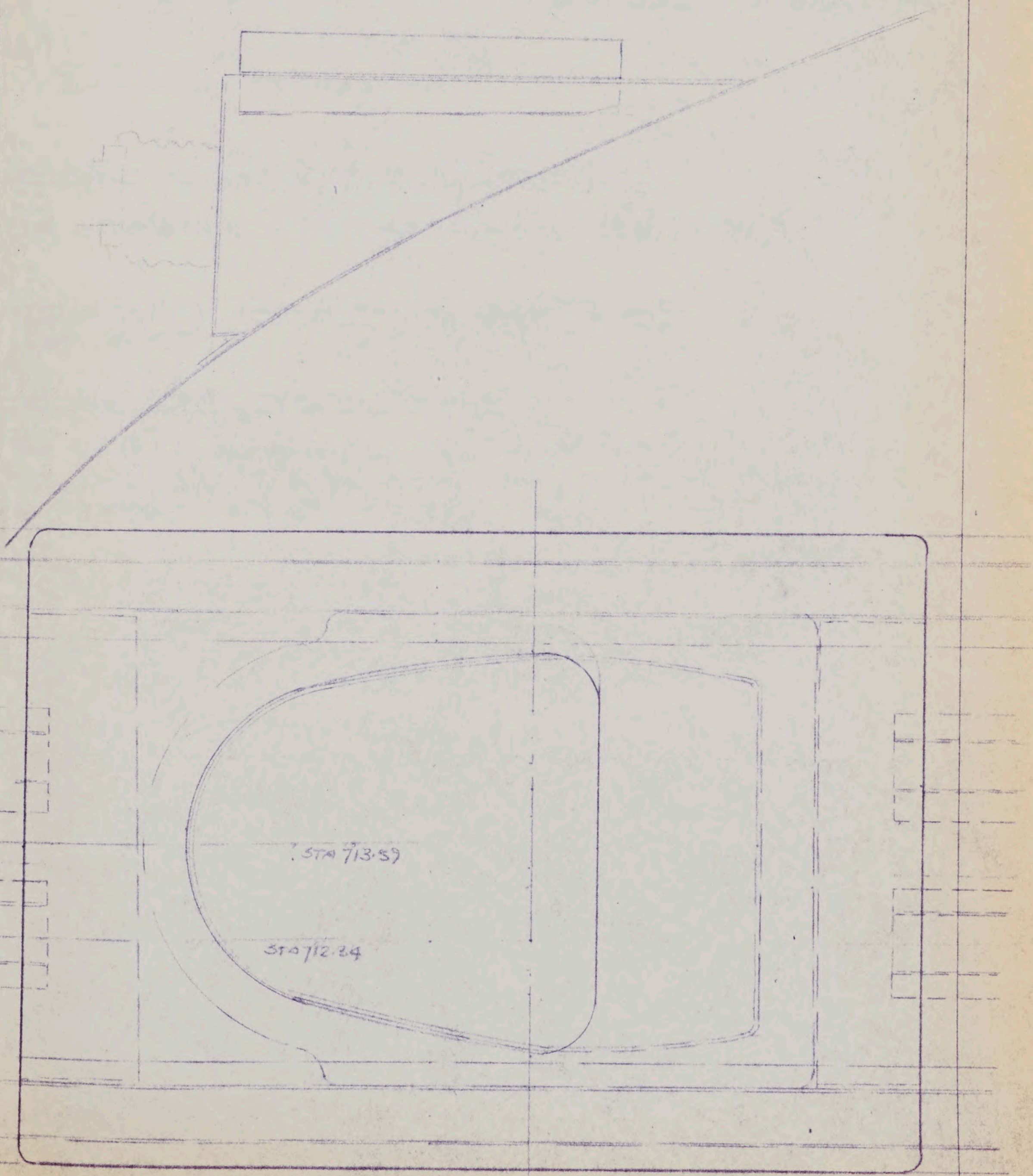
THE SHEAR END LOAD & B.M. DIAGRAM IS SHOWN ON PAGE 123

RM 1319A

AVRO AIRCRAFT LIMITED
MALTON - ONTARIO
TECHNICAL DEPARTMENT

AIRCRAFT: C105	REAR INB'D ENGINE MOUNTING ACCESS. REF DWG 7-1058-2115/6.	REPORT NO. 7-0558-38	SHEET NO. 131
		PREPARED BY A. FERENC	DATE FEB/56
		CHECKED BY	DATE

REAR INB'D ENGINE MOUNTING ACCESS
REF DWG - 7-0158 - 2115 - 2116 - 1



FORM 1319A

AVRO AIRCRAFT LIMITED
MALTON - ONTARIO
TECHNICAL DEPARTMENT

AIRCRAFT: C105	Rear Inb'd Mounting Access	REPORT No. 7-0558-38	SHEET No 132
		PREPARED BY: A. Ferenc	DATE: Feb/56
		CHECKED BY	DATE

Rear Inb'd Mounting Access.

Loading.

The Loading On Can Is A Pressure Load. The Pressures Are:
16.65 P.S.I. Ult (tens)
− 1.5 P.S.I. Ult (comp)
Temp Conditions 265° For 50 hrs.

Connection Of Cut Stiffener To Torsion Box

For Attachment & Tie Plate Stressing Refer To Sht 3

Attachment Of Torsion Box To Uncut Member
Ref Sht 5

Can Lid Under Internal Pressure

This Is Passed By Inspection On Comparison With The Stressing On The Rear Outb'd Engine Mount Access Ref Pages 5 - 9.

FORM 1319A

AVRO AIRCRAFT LIMITED
MALTON ONTARIO
TECHNICAL DEPARTMENT

REPORT No 7-0558-38

SHEET No. 143

AIRCRAFT:		PREPARED BY	DATE
C105	CUT OUTS FOR ENGINE RAIL BRACKETS	A. FERENC	FEB/56
		CHECKED BY	DATE

CUT OUT AT STA 644.43

HOOP TENS. LOAD = $30.4 \times 24.85 = 755$ LBS/INS
HOOP TENS. STRESS = $\frac{755}{.020} = 37800$ LBS/INS^2

SINCE THE TENSION LOAD IS PARALLEL WITH THE LONG SIDE IT WAS SUGGESTED BY K. IDDON TO ASSUME DIAMETERS CORRESPONDING TO THE SHORT SIDE DIMENSIONS

REF RAeS DATA SHT 02.04.03

$D = 2.1''$; $d = 1.1''$, $t = .020''$, $t_r = .051''$

$D/d = \frac{2.1}{1.1} = 1.91$; $\frac{t_r}{t} = \frac{.051}{.020} = 2.55$

FROM DATA SHT $\frac{f_{MAX}}{f} = 1.50$

$f_{MAX} = 1.50 \times 37800 = 56700$ LBS/INS^2

ALLOWABLE FTU FOR 75 S-T6 AT 265° FOR 50 HRS = $.78 \times 70000 = 54500$ LBS/INS^2

TOO HIGH:

ON THE FOLLOWING SHT A CROSS PLOT OF t_r/t Vs D/d : f/f_{MAX} FOR TWO VALUES OF t/t IS SHOWN.

AT t_r/t OF 2.55 : D/d OF 3.0 $f/f_{MAX} = 1.26$

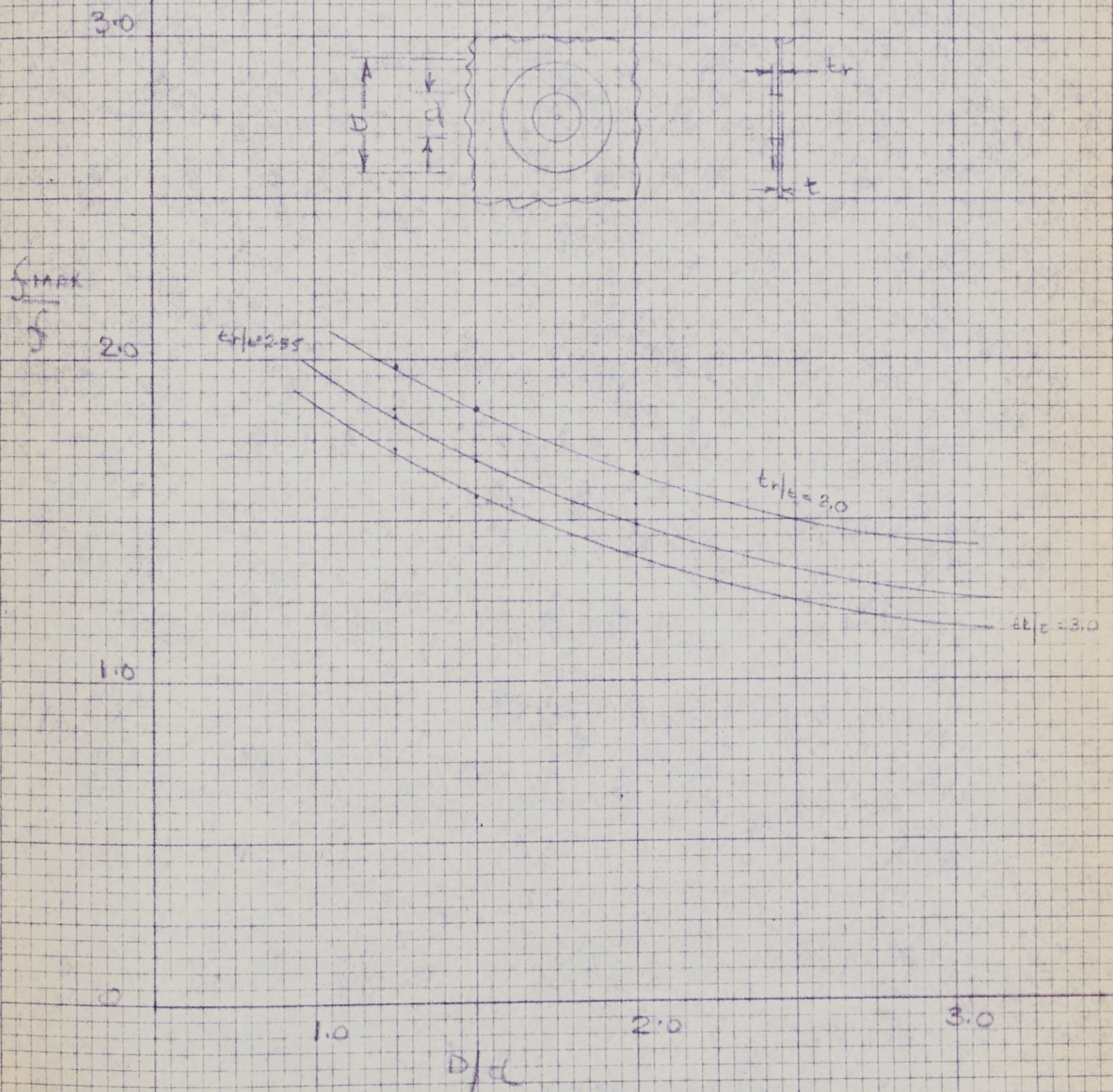
$\therefore f_{MAX} = 1.26 \times 37800 = 47600$ LBS/INS^2

R.F = $\frac{54500}{47600}$ 1.145

ACTUALLY THE D/d VALUE MUST BE INCREASED TO PICK UP ON EXISTING RIVETS IN BULB ANGLE, BUT R.F IS COVERED BY ABOVE.

FORM 1319 A

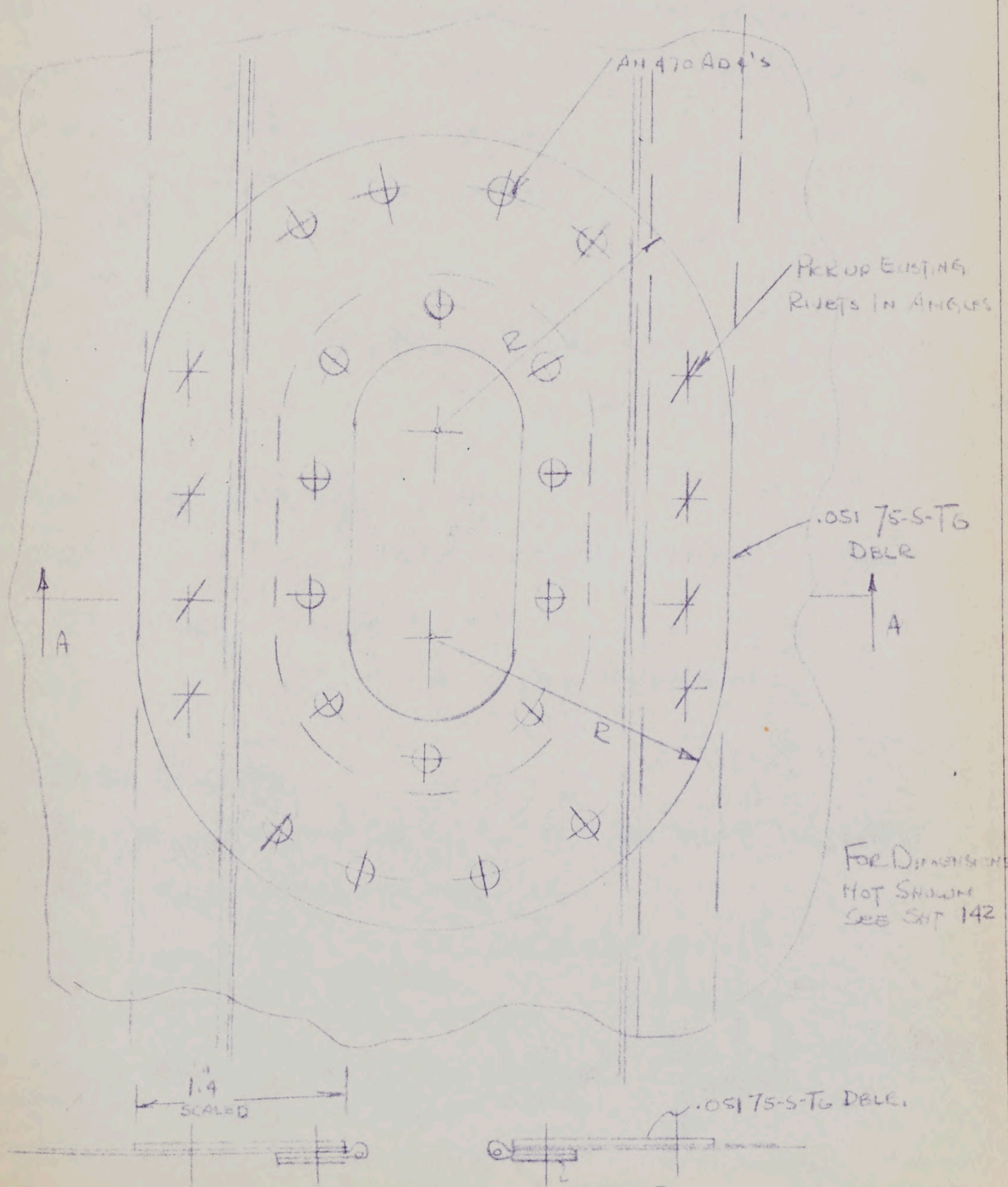
CROSS PLOT OF VARIOUS t_r/t VALUES VS D/d & f/f_{max}
REF RAeS DATA SHT 02.04.03



AVRO AIRCRAFT LIMITED MALTON - ONTARIO **TECHNICAL DEPARTMENT**		REPORT NO. 7-0558-38 5 SHEET NO. 145	
AIRCRAFT: C105	CUT OUTS FOR ENGINE RAIL SUPPORT BRACKETS	PREPARED BY A. FERENC	DATE Feb/56
		CHECKED BY	DATE

CUT OUT AT STA 644.43

SUGGESTED REINFORCING AROUND CUT OUT.



NOTE: THE ABOVE IS FOR STA 644.43
FOR STA 697.65 THE DOUBLER
& SEAL RETAINER ARE TO BE 24 S-T6

AVRO AIRCRAFT LIMITED
MALTON ONTARIO
TECHNICAL DEPARTMENT

AIRCRAFT: C105	CUT OUTS FOR ENGINE RAIL SUPPORT BRACKETS	REPORT NO. 7-0558-38	SHEET NO. 146
		PREPARED BY: A. FERRIC	DATE: FEB/56
		CHECKED BY	DATE

CUT OUT AT STA 644.43

CHECK RIVETTING

TENSION LOAD TO BE TRANSFERRED = 1.1 x 755 = 830 LBS

ALLOWABLE B.S OF 1/8 RIVETS IN .020 e/d = 2.0 AT 265° FOR 50 HRS IN 75-S-T6 CLAD
= .020 x .125 x .84 x 133000 = 333 LBS

S.S OF RIVET = 388 x .95 x .80 = 295 LBS
AT 265° FOR 50 HRS

NO OF RIVETS (MIN AT 1ST) = 4

TOTAL CAPACITY OF RIVETS = 4 x 295 = 1180 LBS

RF = $\frac{1180}{830}$ 1.425

LOAD ON DECK / SIDE = $\frac{830}{2}$ = 415 LBS

DECK AREA LESS HOLES = (1.4 x .051 − .25 x .051) = .051 (1.15) = .0587 INS^2

TENS STRESS = $\frac{415}{.058}$ = 7070 LBS / INS^2

SATISFACTORY BY INSPECTION.

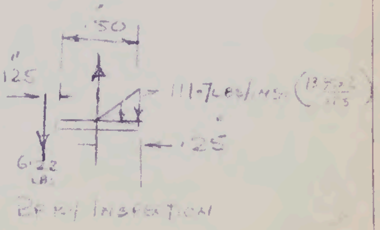
CHECK SEAL RETAINER

PERIMETER AROUND SEAL AT ℄ OF PORTION SUBJECTED TO PRESSURE

≃ 2 x 1.35 + π x .90 = 2.70 + 2.82 = 5.52 INS

WIDTH OF SEAL = .25" ∴ A = .25 x 5.52 = 1.38 INS^2
LOAD = 1.38 x 24.85 = 34.3 LBS
LOAD/INS = $\frac{34.3}{5.52}$ = 6.22 LBS/INS

TAKING AN INS WIDTH Z = $.051^2/6$ = .000433 INS^3
BEARING LOAD = $6.22 \times \frac{.375}{.25 \times .666}$ = 2.33/.1665 = 13.95 LBS



RIVET TENS LOAD = 13.95 + 6.22 = 20.17 LBS   RF BY INSPECTION 72

fb = 2.33/.000433 = 5420 LBS/INS^2

USING .032 RETAINER fb = $\frac{2.33 \times 6}{.032^2}$ = 13700 LBS/INS^2 RF 7 BY INSPECTION 72

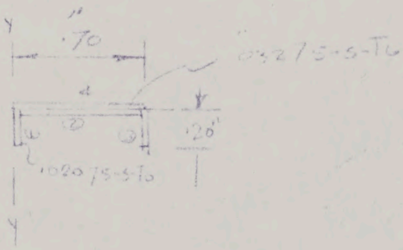
[illegible]

FORM 1319A

AVRO AIRCRAFT LIMITED
MALTON · ONTARIO
TECHNICAL DEPARTMENT

AIRCRAFT: C105	LOWER SHROUD	REPORT No 7-0558-38	SHEET No 157.
		PREPARED BY: A. FERENC	DATE: MAY/56
		CHECKED BY	DATE

CHECK STRINGER GUSSET BENDING
THE MOMENT OF 372 IN LBS WILL HAVE TO BE CARRIED IN BENDING BY GUSSET & STRINGER. ONLY THE CROWN PLUS .20" OF FLANGE WILL BE ASSUMED EFFECTIVE WITH GUSSET.



ITEM	b	t	A	x	Ax	Ax²	Icg
1	.20	.02	.004	.010	.00004	NEG	NEG
2	.66	.02	.0132	.35	.00461	.00161	.00048
3	.20	.02	.004	.69	.00276	.00190	NEG
4	.70	.032	.0224	.35	.00785	.00274	.000915
Σ			.0436	.35	.01526	.00525	.001395

$I_{cg} = .00525 + .001395 - .35^2 \times .0436 = .006645 - .00534 = .001305 \text{ ins}^4$

$Z_c = Z_T = \frac{.001305}{.35} = .00374$

$f_b = \frac{372}{.00374} = 99300 \text{ lbs/ins}^2$ TOO HIGH.

THE GUSSET IS TO BE FORMED DOWN OVER STRINGER WITH .25 FLANGES.

FORM 1319A

AVRO AIRCRAFT LIMITED
MALTON - ONTARIO
TECHNICAL DEPARTMENT

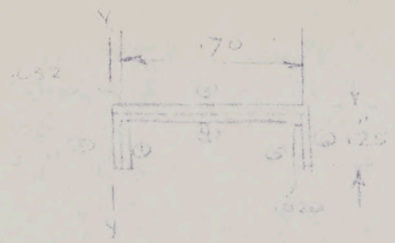
REPORT NO. 7-[illegible]-38
SHEET NO. 153

AIRCRAFT: C105

Lower Shroud

PREPARED BY	DATE
A. Ferenc	May/56
CHECKED BY	DATE

Check Stringer Gusset Bending.



Item	b	t	x	A	Ax	Ax²	I cg
1	.198	.020	.042	.00396	.000165	NEG	NEG
2	.218	.032	.016	.00697	.000111	NEG	NEG
3	.764	.032	.382	.02440	.009310	.00356	
4	.70	.020	.382	.01400	.005350	.00204	.000572
5	.198	.020	.722	.00396	.002860	.00206	.001190
6	.218	.032	.748	.00697	.005220	.00390	
Σ			.382	.06026	.023016	.01156	.001762

$I_{cg} = .01156 + .001762 - .382^2 \times .06026 = .013322 - .0088 = .004522$ ins^4

$Z_c = Z_T = \dfrac{.004522}{.35} = .0129$ ins^3

$f_b = \dfrac{372}{.0129} = 28800$ lbs/ins^2

Direct Comp Stress $= \dfrac{269}{.06026} = 4480$ lbs/ins^2

Net Comp Stress $= 28800 + 4480 = 33280$ lbs/ins^2

Fcc By Needhams Formulae For Gusset At Tip

$$= .342 \dfrac{\left(50300 \times 9.55 \times 10^6\right)^{1/2}}{\left(\dfrac{.322 + .25}{2 \times .032}\right)} = \dfrac{238000}{5.17} = 46200 \text{ lbs/ins}^2$$

$\text{RF}_{cy} = \dfrac{46200}{33280}$ 1.365

AVRO

AVRO AIRCRAFT LIMITED
MALTON - ONTARIO
TECHNICAL DEPARTMENT

REPORT No 70558-38	SHEET No 158a	
AIRCRAFT: C105	Lower Shroud	
PREPARED BY: A. Ferens	DATE: May/56	
CHECKED BY	DATE	

Check Stringer Gusset Bending

At one point in the Gusset 1 Leg will be cut off

$\therefore A = .06026 - .00697 = .05329$

$Ax = .023016 - .000111 = .022905$

$Ax^2 = .01156$

$I_{cg} = .001762$

$\bar{y} = .022905/.05329 = .43$

$I = .01156 + .001762 - .43^2 \times .05329 = .013322 - .009875 = .003447$

$Z_c = \frac{.003447}{.43} = .00801$

$f_b = \frac{37.2}{.00801} = 4680$

Direct Comp Stress $= \frac{.269}{.05329} = 5050$

Fcc for .020 Crown. Both edges supported at Temp

$= \frac{.366 \times (50800 \times 9.5 \times 10^6)}{\left(\frac{.351 \cdot 20}{24 \times 20}\right)^{.75}} = \frac{259000}{4.36} = 57200$

F_{cy} To Be used

$RF = \frac{50800}{4680 + 5050}$ .992

Check Gusset

$f_{b_{max}} = \frac{372 \times .399}{.003447} = 43200$ lbs/ins² Comp

Total Comp Stress $= 43200 + 5050 = 48200$ lbs/ins²

Assuming Room Temp Conditions $E = 10.5 \times 10^6$

$\frac{t}{b} = \frac{.032}{.135} = .0315$

From Bruhn For Sides Point One Free $K = .8$

Pitch approx

$a/b = \frac{.65}{.35} = 1.855$

$F_{scr} = K E \, t/b^2 \quad .8 (10.5 \times 10^6) \times .0315^2 = 70500$ lbs/ins²

From ANC-5 Page 84 At $S = 35000$, $E = 9.5 \times 10^6$

Trying $E = 8.0 \times 10^6$ & $S = 33200$

$F_{scr} = .8 \times 8.0 \times 10^6 \times .0315^2 = 33600$ Check OK

Correction for Temp $= \frac{9.81}{10.5} \times 33600 = 43500$

$RF_{cy} = \frac{48500}{48250}$ 1.0

FORM 1319A

AVRO AIRCRAFT LIMITED
MALTON - ONTARIO
TECHNICAL DEPARTMENT

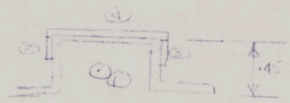
AIRCRAFT:		REPORT No 7-0558-28
C105	Lower Shroud	SHEET No. 159

PREPARED BY	DATE
Ferenc	May/56
CHECKED BY	DATE

Check Stiffness Of Gusset & Stringer

Gusset + Stringer should replace stiffness of cut stringer i.e. I of gusset + stringer ≥ I stringer alone.

Refer To Cut 4



.45 A stringer = .05359 Z = .308 $\bar{I}_{cg}$ = .001886

Item	b	t	A	y	Ay	Ay²	Icg
1			.05359	.308	.01642	.00507	.001886
2		Cut 1.38	* .00697	$\bar{.036}$	$.\bar{0}00251$	Neg	Neg
3		"	* .00697	-.036	$\bar{.000251}$	Neg	Neg
4		"	* .01400	.161	$.\bar{0}0225$	.000363	Neg
Σ			.08153	.1675	.013668	.005433	.001886

I_{cg} = .005433 + .001886 − .1675² × .08153 = .007319 − .001370 = .005049 ins⁴

$$K_F = \frac{.005049}{.003772} = 1.335$$

The Remainder Of Gusset [illegible] [illegible] Inspection.

* One Leg Has Been Removed

A = .08153 − .00697 = .07456

Ay = .013668 + .000251 = .013919

$$\bar{y} = \frac{.013919}{.07456} = .1865$$

Ay² = .005433

I_{cg} = .001886

I_{NA} = .005433 + .001886 − .1865² × .07456 = .007319 − .00269 = .004629

$$R_F = \frac{.004629}{.003772}$$ 1.225

FORM 1319A