

QCX
Avro
CF105
R-7-0510-9
ANALYZED



TECHNICAL REPORT



ANALYZED

A. V. ROE CANADA LIMITED
MALTON, ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C 105

REPORT NO. 7/0510/9 Summary

FILE NO:

NO. OF SHEETS: 37 + 2 Drawings

TITLE: Summary of Centre Fuselage Analysis

Classification cancelled / Changed to UNCLASSBy authority of PNRSDate 30 Sept 66Signature [Signature]Unit / Rank / Appointment PNRS S~~CONFIDENTIAL~~

This summary presents geometry, general data, and stress and deflection matrices by extraction of sections 3, 15, and 16 and figures 1 and 2 from the complete report.

NRC - CISTI
AERO / M.E.
LIBRARY

87- 12 10

BIBLIOTHÈQUE
AÉRO / G.M.
CNRC - ICISTPREPARED BY [Signature]

DATE Dec '55

CHECKED BY [Signature]

DATE Dec '55

SUPERVISED BY [Signature]

DATE Dec '55

APPROVED BY

DATE

ISSUE NO.	REVISION NO.	REVISED BY	APPROVED BY	DATE	REMARKS

15865741

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0510/9

SHEET No.

AIRCRAFT:

C 105

Centre Fuselage

PREPARED BY

DATE

G. E. Burrell

Dec 1944

CHECKED BY

DATE

CONFIDENTIAL

SUMMARY

Stresses and deflections due to each of twenty three unit loads are predicted from a strain energy analysis of the C 105 Centre Fuselage. The seventy one redundant analysis gives results for use in detail analysis of the component and in a strain energy analysis of the complete aeroplane.

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0510/9 Summary

SHEET No. 1-1

AIRCRAFT:

C 105

Centre Fuselage

PREPARED BY

DATE

C. E. Burrell

Dec '55

CHECKED BY

DATE

INDEX

EXTRACTS FROM C 105 CENTRE FUSELAGE ANALYSIS

§3. Introduction

§15. Stresses due to Unit Loads

§16. Deflections due to Unit Loads

Figs 1 and 2. Geometry Data

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/ 0510/9

SHEET NO. 3-1

AIRCRAFT:

C 105

Centre Fuselage

PREPARED BY

DATE

C. E. Burrell

Dec '55

CHECKED BY

DATE

INTRODUCTION

This report contains an overall analysis of the C 105 Centre Fuselage for symmetric loads. The structure considered extends from stations 255.00in. to 485.00in..

Stresses and deflections due to unit loads are predicted from a strain energy analysis based on a method developed in Avro Stress Report Gen/1090/336 which permits the handling of large scale problems.

SPECIAL FEATURES

In all, twenty three loads are considered as acting symmetrically on the component. Of these, twenty represent inertia and air loads, three join the wing to the fuselage, and one acts as a dummy load to permit computation of the fore and aft deflection of the tank pin at station 485.00in..

In this solution the redundants total seventy one. Since it is economically impossible, even using the present method, to solve a problem of this magnitude by desk calculator, as much use as possible has been made of the company's C. P. C. type digital computers. Even then, the capacities of these machines has made a further simplification desirable. This was achieved by partitioning the structure and solving four sub-problems. The results from these sub-problems are then combined to obtain the final solution.

This combination is effected by adding the four deflections from the sub-problems and operating on the combined matrix to produce a

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0510/9

SHEET No. 3-2

AIRCRAFT:

C 105

Centre Fuselage

PREPARED BY

DATE

C. E. Burrell

Dec '55

CHECKED BY

DATE

relationship between the applied loads and the joining loads. Using this, the final stresses and deflections can be calculated. For details, see section 5 of this report.

In order to prevent waste of expensive machine time, all calculations were checked before the necessary data was released to the Computing Group. The Computing Group checks all of their calculations. In addition to these checks, all computer calculations were spot checked by members of the Stress Office. All checks indicate that the final results are mathematically correct and with better than slide rule accuracy.

REPRESENTATION OF STRUCTURE

In order to reduce the number of redundants to seventy one, it has been necessary to lump the frames by twos, with the lumped frame located between the two. The frame at 469.50^{in.} is considered separately, as

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0510/9

SHEET No. 33

AIRCRAFT:

C 105

Centre Fuselage

PREPARED BY

DATE

C. E. Burrell

Dec '55

CHECKED BY

DATE

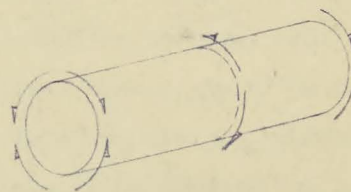
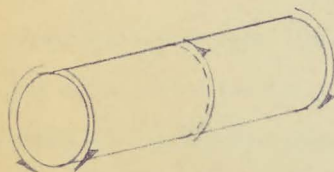
are, of course, the end bulkheads. Then each lumped frame is represented by a shear panel of equivalent thickness to provide the same strain energy. The load (in kips) transferred to each outer wall from the tank is

$$0.131\ 238\ \tau_m \quad \text{for one frame}$$

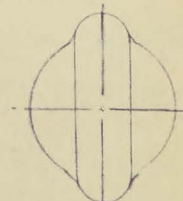
$$0.262\ 475\ \tau_m \quad \text{for two frames}$$

The side wall and the fuel tank are both represented by two booms and a shear web, following the usual lumping method.

In the duct, several stresses are associated with a significant stress quoted at one stress point. Both the bending and warping axial stresses have an associated shear stress. These stresses are derived from redundant loads acting on the duct as shown.



In sub-problem 1, the stress pattern is complicated by the shape of the duct. For detail stressing, see section nine of this report.



In sub-problems 2 and 3, the duct stresses are given by the formulae:

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT:

C 105

Centre Fuselage

REPORT NO. 7/0510/9

SHEET NO. 3-5

PREPARED BY

DATE

C. E. Burrell

Dec 1955

CHECKED BY

DATE

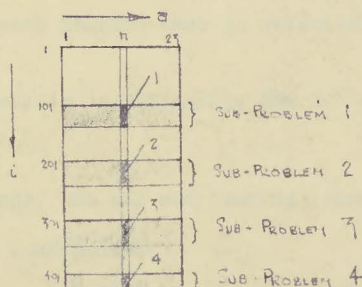
DATA OBTAINED

Stresses are quoted for 163 stress points in kips per square inch for a load of one kip applied to each side of the structure. Of these 163 points, twenty one are duplicates due to the partitioning. This provides 142 different points, which may be used, if necessary, as a basis for interpolation. The load at any stress point may be found by multiplying the stress by the area shown on figure 1.

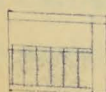
The set of pages of section 15 give all stresses due to all twenty three loads. These stresses are arranged for computer work as a matrix (here designated S_{ia}) with the columns of stresses positioned as shown.



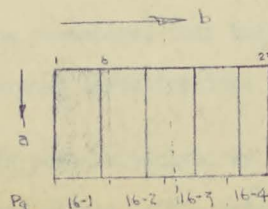
PAGE 15-17



Deflections are also given for each of the twenty three load points due to all twenty three loads. All deflections are given in inches per kip per side. The set of pages of section 16 are arranged for computer work (here designated Z_{ab} or E_{ab}^w interchangeably) with the columns of deflections positioned as shown.



PAGE 16-1



Z_{ab}

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/9

SHEET NO. 3-6

AIRCRAFT:

C 105

Centre Fuselage

PREPARED BY

DATE

C. E. Burrell

Dec '55

CHECKED BY

DATE

USING THE RESULTS

Initially, these results are intended for use in an overall analysis of the complete aeroplane and for detail analysis and weight saving of the component.

In the overall analysis of the complete aeroplane, the various component strain energy analyses (Centre Fuselage etc) must be added together. This method is described in section 5 of this report.

For detail stressing, stresses can be quickly established due to actual loading cases. For this, a Q^{am} matrix expressing the quantity of each unit load required for each loading case is established.

Until the complete aeroplane analysis is available, the following procedure is recommended.

1. For each loading case, the air and inertia loads should be distributed to the load points.
2. For the forward wing pick-up load, use the loads of the Tank 3 Analysis (Ref 7/0910/8) distributed to the unit load points 15 and 16
3. To cater for wing deflections, the deflections given by the Wing Analysis (Ref 7/0510/2) should be applied to bulkhead 485.00.

In using these results, it should be remembered that bulkhead 255.00 in has been assumed rigid, pending further investigation.

Then the S^{ia} matrix (See pg 3-5) is post multiplied by the Q^{am} matrix to obtain the total stresses at each stress point. Details of this

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/9

SHEET NO. 3-7

AIRCRAFT:

C 105

Centre Fuselage

PREPARED BY

DATE

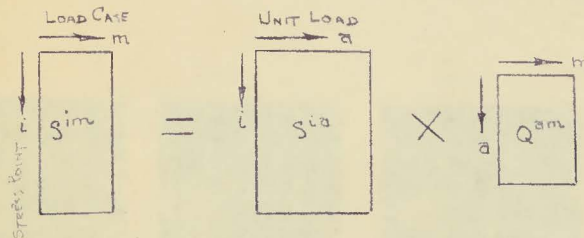
C. E. Burrell

Dec '55

CHECKED BY

DATE

procedure have been established and are known to the Computing Group.



To speed the aeroplane analysis and the detail calculations, all results are already available on punch cards.

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7 (510-0)

SHEET NO. 15-1

AIRCRAFT

C 105

CENTRE FUSELAGE

PREPARED BY

DATE

Loc. 55

CHECKED BY

DATE

LOAD No. 1

STRESSES DUE TO 1 KP PER SQ IN AT LOCATION NO. 1

STRESS
POINT

PART 1

PART 2

PART 3

PART 4

1	9826200	1830800	16248700	19979000
2	5955600*	594400*	4668500*	2652500*
3	2039287	266756	2161949	2946369
4	5122	25932*	98589	529481*
5	259000*	9900*	10700*	83400
6	4963300	1102000	10102200	5272600
7	184439	8376	159518	290974
8	967515*	243697*	4103610*	7264471*
9	11326080	1453894	13700990	15813340
10	17881620*	1652807*	16958250*	20027360*
11	3300308	471569	6674442	10430890
12		29063	795315	1824298
13	14884006	1484377	17399369	21254119
14	6097128*	524543*	4195241*	1820636*
15	2811345	250877	2178513	635437
16	16114	6262*	47879*	504169*
17	64130	1452	14591*	10796*
18	8054741	1078032	9447081	1791869
19	76933	572	175221	123075*
20	1787843*	291477*	4918507*	7849307*
21	12658413	1436591	13966107	16963018
22	17825571*	1637170*	17616095*	21567751*
23	4406033	499360	7447492	5756939
24	344047	45174	1096019	3630040
25	18374965	1499243	18637493	21088852
26	6067716*	507955*	3559323*	
27	3445443	226291	2443166	
28	235447*	10981	276782*	18216528
29	119706	10697*	9023*	23341166*
30	10069106	1031938	8034488	
31	107838*	17838	210261	
32	2327634*	345014*	6002911*	
33	13814008	1394217	14699495	1433638*
34	17272495*	1657368*	18568152*	123075*
35	3577649	615267	8680619	
36	490460	46169	1426324	
37	290818	79592	1823786	
38	18307800	1624900	19979000	
39	5943500*	466800*	2652500*	
40	98900*	1100*	83400	
41	11019700	1010200	5272600	
42	14539640	1370159	15813440	
43	16528270*	1695785*	20027360*	

AIR V. ROE CANADA LIMITED
 MADE IN CANADA
 TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7 510 3

SHEET NO. 15-2

AIRCRAFT

C 105

CENTRE FUSELAGE

PREPARED BY

DATE

Dec. 55

CHECKED BY

DATE

LOAD No. 1

STRESSES DUE TO 1 LB. PER SQ. IN. AT LOCATION No. 2

STRESS
POINT

PART 1

PART 2

PART 3

PART 4

1	1491700	600200	7952800	11841600
2	204000	114100*	1915000*	1407500*
3	2207569	259044	2395795	2679744
4	245523*	46615*	208348*	489703*
5	188900*	12500*	18000*	70200
6	4111600*	295000	4489800	2926100
7	130671	7973	171027	353440
8	792938*	192687*	3155457*	5141674*
9	504800	501604	6975300	9488430
10	2767600*	606461*	8712200*	12034200*
11	2811890	390057	5607776	8274286
12		12559	656926	1811908
13	3049078	580375	9206487	12961953
14	314253*	151380*	1903708*	1018454*
15	2835499	255742	2382807	1191580
16	392016*	35311*	258185*	452969*
17	31391	286*	23930*	3819*
18	924529*	362593	4610859	1092987
19	90222	2107	194904	126330*
20	1318893*	230770*	3689831*	5505733*
21	2007158	594092	7596400	10469741
22	3983119*	695872*	9647925*	13287686*
23	3570324	417527	6190620	5031476
24	244585	27780	978972	3748886
25	4904550	665950	10503874	13665835
26	778239*	181853*	1752601*	
27	3280249	244331	2485353	
28	519073*	23678*	364069*	11607156
29	148592	10896*	22858*	14746365*
30	1480796	39967	4200942	
31	153994*	17791	246197	
32	1773456*	270339*	4373307*	
33	3524975	653013	8437056	2038721*
34	5093273*	786253*	10708127*	126330*
35	2665975*	517411	7060186	
36	413475	32092	1346870	
37	126096	65668	1811483	
38	6002100	795200	11841600	
39	1140800*	191500*	1407500*	
40	124900*	1800*	70200	
41	2949600	449000	2926100	
42	5015940	697560	9488430	
43	6065410*	871260*	12034200*	

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7 051079

SHEETS 15-3

AIRCRAFT

C 105

CENTRE FUSELAGE

PREPARED BY

DATE

CHECKED BY

Dec. 55

DATE

LOAD No. 3

STRESS - 100 TO 1 KIP PER SQ IN AT LOCATION NO. 1

STRESS CUT	PART 1	PART 2	PART 3	PART 4
1	1234900	403400	6687800	10572800
2	119600	34700*	1505000*	1215900*
3	2496855*	277336	2461737	2635739
4	131959	38513*	191484*	461981*
5	1812800*	22900*	49400*	61900
6	2128500*	152000	3562300	2543200
7	1459523	10735*	116467	348963
8	569472	199022*	3061795*	4820836*
9	769100*	347836	5805000	8412400
10	1379800*	439941*	7344270*	10720010*
11	2676195	351617	5383552	7917371
12		111786	897348	1855982
13	826041	440779	7949245	11662577
14	865462	93928*	1560246*	892393*
15	3469488	267580	2426908	1285894
16	433317*	29846*	243158*	433116*
17	1092878	6989*	45858*	5904*
18	2782234*	240331	3816327	978910
19	346993*	10186*	162443	128455*
20	1458208*	232464*	3533312*	5146705*
21	949771	449958	6495736	9370096
22	2114524*	541273*	8328872*	11929449*
23	2657610	389661	5960757	4907444
24	2689231	91648	1116640	3812113
25	2616909	538857	9245253	12376419
26	165956	133194*	1478588*	
27	3686882	253061	2495375	
28	474232*	20694*	344339*	10489374
29	240149	14110*	38201*	13338332*
30	110464*	294286	3573821	
31	500911*	9140	229392	
32	1889775*	266560*	4139714*	
33	2056109	523867	7363920	2147366*
34	3238996*	641703*	9410907*	128455*
35	2074004	492341	6784048	
36	2060243	72584	1422794	
37	1119602	89609	1855477	
38	4033400	668800	10572800	
39	347100*	150600*	1215900*	
40	229300*	4900*	61900	
41	1519600	356200	2543200	
42	3479260	580450	8412400	
43	4398910*	734517*	10720010*	

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7-0510-4

SHEET NO. 15-4

AIRCRAFT

C 105

CENTRE FUSELAGE

PREPARED BY

DATE

Dec. 55

CHECKED BY

DATE

LOAD No. 4

STRESS IS DUE TO 1 KIP P. R. SIDE AT LOCATION No. 1

STRESS
IN

	PART 1	PART 2	PART 3	PART 4
1	973900	194900	5410700	9308400
2	145900	56000	1079200*	1022000*
3	1753978*	298828	2546677	2596433
4	29042	35812*	175817*	431862*
5	1552100*	13900*	86300*	52100
6	1005500*	23100*	2581700	2153400
7	987670	32643*	42308	337585
8	420565	201058*	2971137*	4503699*
9	1259100*	224068	4662200	7335160
10	654000*	291520*	5998240*	9407820*
11	1751989	295077	5106231	7545160
12		238513	1247669	1928556
13	762186	296183	6688803	10368599
14	656939	30777*	1208983*	764731*
15	1831205*	282918	2483809	1387726
16	81002*	26183*	226330*	412366*
17	68599*	10093*	73287*	9589*
18	1371144*	100769	2992293	863035
19	1146772	25578*	115786	131779*
20	181468	232458*	3381391*	4791782*
21	125579*	319824	5406474	6267150
22	1140641*	395973*	7021020*	10571602*
23	2370398	349994	5697096	4775816
24	1866496	177216	1327308	3907240
25	165376	409064	7988733	11095301
26	1274353	81136*	1200875*	
27	4059384	264692	2512800	
28	530941*	18509*	322307*	9366302
29	1359115	18024*	56945*	11929897*
30	2221112*	179705	2930902	
31	873900*	2211*	202789	
32	1889732*	262581*	3910618*	
33	1120028	401121	6294086	2268513*
34	1736036*	501653*	8118989*	131779*
35	1228874	458372	6466007	
36	4017322	129076	1545619	
37	2386394	124749	1927974	
38	1949300	541100	9308400	
39	559500	108000*	1022000*	
40	139400*	8600*	52100	
41	231100*	258200	2153400	
42	2240580	466140	7335160	
43	2915100*	599774*	9407820*	

TECHNICAL DEPARTMENT (Aircraft)

SHEET NO. 15-5

AIRCRAFT

C 105

CENTRE FUSELAGE

PREPARED BY

DATE

Dec. 55

CHECKED BY

DATE

LOAD No. 5

STRESSES DUE TO 1 KIP PER SIDE AT LOCATION No. 5

STRESS POINT	PART 1	PART 2	PART 3	PART 4
1	663000	340000*	4141100	8073900
2	282100	165000	645300*	832000*
3	1067963*	334520	2680618	2564746*
4	6657*	34710*	137152*	390850*
5	1001500*	105200	133700*	37800
6	595300*	210000*	1583000	1760700
7	590432	62650*	79052*	311120
8	232854	204794*	2916275*	4206490*
9	1002600*	124700	3517700	6238620
10	494200*	157000*	4660100*	8093630*
11	1047272	220437	4758208	7156224
12		434140	1845191	2057028
13	575756	143688	5450461	9104732
14	598604	40375	859422*	636495*
15	1209003*	310556	2568312	1500745
16	118889*	22118*	189202*	386022*
17	223993*	6804	113917*	16370*
18	810324*	45293*	2157161	746472
19	714752	49171*	37026	137404*
20	65912	236952*	3258670*	4456432*
21	324920*	200190	4303408	7144698
22	782457*	256774*	5713968*	9214745*
23	1466561	295728	5390633	4633194
24	1126646	316085	1695574	4062292
25	226996	277371	6759912	9846790
26	1026170	25878*	927461*	
27	1524879*	284523	2546225	
28	65784*	14225*	284776*	8221917
29	28553	19037*	85690*	10520161*
30	1314876*	62024	2281881	
31	981021	20462*	154182	
32	207937*	263002*	3705222*	
33	432482	280974	5205952	2408253*
34	1155392*	364003*	6825261*	137404*
35	1918597	413102	6162368	
36	2499739	223269	1765543	
37	4342569	184490	2056868	
38	340300*	414100	8073900	
39	1649900	64500*	832000*	
40	1051600	13400*	37800	
41	2099800*	158300	1760700	
42	1246400	351730	6238820	
43	1569000*	466030*	8093630*	

TECHNICAL DEPARTMENT (Aircraft)

SHEET 15-6

AIR PART

C 105

CENTRE FUSELAGE

PREPARED BY

DATE

Dec. 55

CHECKED BY

DATE

LOAD No. 6

STRESSES DUE TO EXTERNAL SIDE LOADS (NO. 6)

STRESS
POINT

PART 1

PART 2

PART 3

PART 4

1	343200	3300	2699800	6781300
2	163700	104400	27600*	602100*
3	708600*	104818*	2873960	2562024
4	31600*	13753*	175285*	364318*
5	690000*	18300	138500*	27300
6	351500*	125700*	459100	1348200
7	397000	67169	191010*	280231
8	163900	41497*	2754314*	3870024*
9	927400*	47600	2533100	5181490
10	270100*	95100*	3442270*	6799840*
11	681200	187077	4302884	6719335
12		308598	2470214	2223903
13	543200	41640*	4090719	7795846
14	369800	154390	392059*	492209*
15	815200*	340775	2701613	1646291
16	165900*	36090*	198473*	367262*
17	191600*	84787	140144*	25057*
18	483800*	227736*	1246629	624382
19	500100	64179*	41634*	144937*
20	79400	213330*	3065548*	4008031*
21	487600*	127539	3298243	6050246
22	457500*	151990*	4477915*	7662718*
23	942800	227281	5004072	4469053
24	759900	435348	2109042	4271493
25	376500	121730	5446991	8566067
26	641800	57711	581149*	
27	1073500*	310759	2617152	
28	200100*	22209*	273544*	7098540
29	108300*	5305*	108735*	9111727*
30	787900*	76114*	1589557	
31	721100	35751*	101677	
32	59100*	246321*	3451228*	
33	21600*	187621	4179618	2581502*
34	691000*	246467*	5572144*	144927*
35	1254100	354194	5777128	
36	1725800	314272	2030767	
37	3087100	246906	2223363	
38	32800	270000	6781300	
39	1043800	2800*	602100*	
40	182800	13900*	27300	
41	1256500*	45900	1348200	
42	475600	253320	5181490	
43	950700*	344287*	6799840*	

AVROE CANADA LIMITED
 TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7 0510 9
 SHEET NO. 15-7

AIRCRAFT		PREPARED BY	DATE
C 105	CENTRE FUSELAGE	CHECKED BY	Dec. 55

LOAD No. 7

STRESS IN TONS PER SQ. IN. AT 100 LBS. PER SQ. IN. No. 7

STRESS POINT	PART 1	PART 2	PART 3	PART 4
1	429300	22000	1155500	3471500
2	106900	66500	710100	349500*
3	463200*	74490*	3128202	2582020
4	39700*	26138*	252121*	339898*
5	463800*	400*	13400*	15600
6	211800*	75900*	864800*	907300
7	263200	51803	338370*	230853
8	111300	20702*	2541854*	3523187*
9	787700*	6200	1690000	4140250
10	154100*	57800*	2335230*	5520750*
11	437500	121948	3677661	6221722
12		212718	3349430	2475578
13	470600	9333*	2667578	6478067
14	239300	99966	149503	333248*
15	536100*	100928*	2883816	1825224
16	170100*	15268*	227346*	350010*
17	143100*	4129	146174*	37842*
18	293000*	137491*	220795	495307
19	343100	66194	150694*	156051*
20	67100	48238*	2845928*	3715306*
21	505700*	55791	2367782	4961800
22	274100*	94052*	3303163*	6518874*
23	596900	192388	4500110	4276124
24	504900	310447	2710409	4582821
25	410200	48006*	4098268	7288150
26	411000	160019	191036*	
27	717500*	341038	2723375	
28	244000*	37138*	270513*	5973259
29	108300*	78576	132180*	7703393*
30	476900*	249096*	834137	
31	507700	53333*	25472	
32	5100*	219515*	3182334*	
33	226800*	120636	3191083	2799047*
34	418700*	148379*	4351124*	156051*
35	798300	272381	5310490	
36	1168200	435313	2426491	
37	2128200	334917	2474859	
38	220000	115600	5471500	
39	664300	71000	349500*	
40	3500*	1300*	15600	
41	758900*	86500*	907300	
42	62500	169010	4140250	
43	578000*	233443*	5520750*	

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

REPORT NO. 7 0510 9

SHEET NO. 15-8

AIRCRAFT

C 105

CENTRE FUSELAGE

PREPARED BY

DATE

Dec. 55

CHECKED BY

DATE

LOAD No. 8

STRESSES DUE TO 1 KP PER SIDE AT LOCATION NO. 8

STRESS POINT	PART 1	PART 2	PART 3	PART 4
1	332400	29900	530400*	4125600
2	75300	42100	1614300	60100*
3	295500*	49319*	3426641	2634414
4	39900*	29958*	392656*	321476*
5	305300*	4100*	788200	7300
6	131200*	46500*	2532500*	418900
7	171200	37677	513129*	155676
8	72500	10210*	2233095*	3155350*
9	638400*	14300*	1067100	3131810
10	92200*	35000*	1400200*	4266270*
11	275600	77311	2824639	5628711
12		143371	4539160	2850351
13	391000	9530	1142136	5136592
14	158700	63542	803264	153988*
15	342300*	71303*	3127317	2054958
16	155800*	27644*	290119*	336750*
17	101800*	9568*	21102*	56826*
18	180900*	83976*	1000838*	354431
19	230500	51314	297854*	172675*
20	49200	25100*	2576706*	3331581*
21	461700*	14879	1561916	3888955
22	167000*	57478*	2226610*	5185138*
23	370600	125458	3820047	4039693
24	328800	214648	3570477	5042947
25	391200	13250*	2686047	6005032
26	264900	102880	267376	
27	460800*	101046*	2878601	
28	242500*	15958*	282682*	4851777
29	85400*	3080*	134923*	6294959*
30	292600*	152372*	28383*	
31	348600	75650	84634*	
32	11700	51731*	2865039*	
33	301100*	55152	2269850	3083993*
34	254600*	92026*	3182806*	172678*
35	497500	229320	4717349	
36	775100	310969	3008417	
37	1435100	453910	2849554	
38	299600	53100*	4125600	
39	420300	161400	60100*	
40	40500*	78800	7300	
41	464600*	253300*	418900	
42	142700*	106700	3131810	
43	350800*	140000*	4266270*	

TECHNICAL DEPARTMENT (Aircraft)

SHEET NO. 15-9

AIRCRAFT

C 105

CENTRE FUSELAGE

PREPARED BY

DATE

Dec. 55

CHECKED BY

DATE

LOAD No. 9

STRESS DUE TO 1 LB PER SQ IN AT LOCATION NO. 9

STRESS POINT	PART 1	PART 2	PART 3	PART 4
1	253000	31500	146100 *	2702800 *
2	50500	25400	1005100	287800
3	183200 *	31000 *	1002362 *	2729509
4	35900 *	29000 *	176994 *	320554 *
5	196400 *	4400 *	12500 *	24300
6	80800 *	27900 *	1519700 *	152200 *
7	108800	27000	777788	46899
8	46200	4100 *	505256 *	2743312 *
9	501600 *	23000 *	475900	2194170
10	52900 *	20400 *	846100 *	3053180 *
11	168300	47300	2347236	4885497
12		94900	3259604	3396823
13	315000	18800	540357 *	3736613
14	101500	38200	1615527	58675
15	211100 *	46900 *	3412938	2360291
16	133700 *	31500 *	421409 *	334507 *
17	71000 *	10000 *	780149	62910 *
18	110200 *	50400 *	2569937 *	193356
19	152200	37800	472054 *	197199 *
20	35600	11500 *	2202591 *	2921355 *
21	394600 *	7100 *	967082	2855608
22	98200 *	33900 *	1307772 *	3868990 *
23	222400	78800	2900044	3736062
24	209200	145700	4725562	5717875
25	346400	7600	1157339	4691715
26	164300	62700	831470	
27	284500 *	71200 *	3093043	
28	218900 *	28300 *	332437 *	3749796
29	63600 *	14600 *	3282 *	4687923 *
30	176100 *	91800 *	1078056 *	
31	235300	57700	233348 *	
32	18800	25100 *	2527211 *	
33	308300 *	16400	1473051	3468839 *
34	149000 *	54900 *	2101973 *	197199 *
35	299900	148700	3930981	
36	503600	216800	3835424	
37	949200	325900	3396123	
38	314200	14600 *	2702800	
39	254600	100600	287800	
40	44200 *	1300 *	24300	
41	278900 *	152000 *	152200 *	
42	230700 *	47500	2191170	
43	204300 *	84700 *	3053180 *	

AIR CANADA LIMITED
 TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 71510-9

SHEET NO. 15-10

PREPARED BY

DATE

C 105

CENTRE FUSELAGE

Dec. 55

CHECKED BY

DATE

LOAD No. 10

SERIES OF POINTS OF APPLICATION NO. 10

STRESS POINT	PART 1	PART 2	PART 3	PART 4
1	187100	29000	79800	1145200
2	23900	13000	552700	727100
3	107500*	18300*	710104*	2871905
4	30400*	25500*	292269*	352634*
5	121900*	4000*	137400*	173000
6	46400*	15600*	853200*	863300*
7	66300	19000	598756	100178*
8	29600	400	182801*	2242975*
9	379100*	24900*	124200	1369540
10	21700*	10300*	462100*	1907990*
11	97200	27300	1485311	3916585
12		60900	2277697	4171398
13	244700	21700	140582*	2225738
14	52700	19900	928110	324237
15	121600*	28800*	1027074*	2776724
16	108100*	30100*	192011*	352855*
17	48500*	8200*	16789*	113695*
18	62700*	28300*	1443774*	1321*
19	97500	27200	816919	233222*
20	28600	2100*	398819*	2455031*
21	318000*	17100*	408420	1897263
22	47500*	17400*	734570*	2581955*
23	125000*	46600	2341636	3332432
24	127700	96500	3406918	6700001
25	288800	17900	546285*	3309797
26	85000	33200	1546115	
27	103400*	46200*	3349462	
28	183800*	31800*	445573*	2692816
29	45900*	13300*	799157	3483590*
30	98500*	51700*	2449294*	
31	154700	42100	412375*	
32	27000	7400*	2049025*	
33	278000*	4700*	877597	4000082*
34	74300*	29200*	1159076*	233222*
35	169400	90700	2880767	
36	316700	147900	4947383	
37	609900	227800	4170809	
38	290600	8000	1145200	
39	129700	55300	727100	
40	40100*	13700*	173000	
41	155800*	85500*	863300*	
42	249000*	12400	1369540	
43	102500*	46200*	1907990*	

AIR CANADA LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7 0510 2

15-11

SHEET NO.

PREPARED BY

DATE

Dec. 55

CHECKED BY

AIRCRAFT

C 105

CENTRE FUSELAGE

LOAD No. 11

STRESSES DUE TO CARBIDE S.D. AT 100 LBS. AC. FT.

STRESS
POINT

PART 1

PART 2

PART 3

PART 4

1	130300	24300	183000	614700*
2	10400*	2100	197600	1292600
3	55300*	9300*	463730*	3018200
4	23000*	20300*	309089*	436611*
5	70200*	3400*	131700*	962500
6	19600*	6600*	379100*	1814400*
7	37300	13000	440057	267155*
8	19900	5000	60492	1569738*
9	266900*	22500*	67700*	737400
10	9900	2200*	177200*	867700*
11	49200	13600	840860*	2647973
12		36900	1543787	5204672
13	178200	20700	80717	525761
14	2900	4600	383893	671698
15	59600*	16000*	744942*	3348158
16	80900*	25200*	284582*	405201*
17	31600*	6400*	144397*	125279*
18	26700*	12400*	639014*	252097*
19	59000	19200	632154	287448*
20	29300	6100	16590	1880805*
21	236200*	19900*	72090	1067807
22	3100*	5000*	307636*	1341008*
23	59300	24500	1377036	2784402
24	72000	61300	2373258	8121329
25	223100	20500	161168*	1798880
26	11900	9800	723721	
27	78800*	27900*	1123253*	
28	141200*	29200*	186429*	1720534
29	32500*	10300*	3682*	2084355*
30	41700*	22800*	1078840*	
31	97200	30000	868179	
32	41600	6800	112882*	
33	224900*	14500*	319849	4735528*
34	13400*	10300*	522254*	287448*
35	81000	49500	2157304	*
36	186400	97600	3565546	
37	370500	154500	5204005	
38	242400	18400	614700*	
39	21400	19800	1292600	
40	34400*	13200*	962500	
41	65900*	37900*	1814400*	
42	224100*	6700*	737400	
43	22300*	17700*	867700*	

AIR FORCE CANADA LIMITED
 TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7-0519-0
 SHEET NO. 15-12

AIRCRAFT

C 105

CENTRE FUSELAGE

PREPARED BY

Doc. 55

CHECKED BY

DATE

LOAD No. 12

STRESSES DUE TO LIFT AND SIDE AT LOCATION NO. 12

STRESS P.L.B.	PART 1	PART 2	PART 3	PART 4
1	92500	19800	202800	381300*
2	45400*	6200*	37900*	530300
3	27400*	4400*	327400*	1671174*
4	17700*	15700*	275800*	99462*
5	42800*	3100*	114700*	103000
6	1800*	1000*	87300*	372200*
7	21800	9600	348100	1099676
8	17100	8900	233600	361059
9	188300*	18900*	141500*	279900
10	38500	3300	2600	277800*
11	23600	6100	471300	2114911
12		24200	1137200	4159888
13	131300	18000	151100	929911*
14	41300*	6300*	31100	1015120
15	25000*	9000*	577400*	3944833
16	60000*	20200*	281500*	480951*
17	22700*	5100*	151700*	20429
18	3700*	2400*	145000*	495149*
19	38300	14800	514300	307964*
20	36000	12800	293600	1355668*
21	173900*	18800*	78200*	591417
22	31800	3200	41600*	473219*
23	24300	12000	801400	2218837
24	42200	42400	1793500	9218651
25	172500	19600	14500*	550087
26	47600*	6300*	194800	
27	31500*	17500*	972500*	
28	106900*	24400*	233900*	1089387
29	24800*	8200*	151500*	1056096*
30	5800*	5600*	236700*	
31	65700	23200	740200	
32	60300	17500	338300	
33	176500*	17200*	47800	5509715*
34	31100	2200	126500*	307964*
35	32800	26300	1300000	
36	116500	70000	2763300	
37	240900	113800	4160400	
38	197400	20300	381300*	
39	61700*	3800*	530300	
40	31000*	11500*	103000	
41	9600*	8700*	372200*	
42	188300*	14100*	279900	
43	33700	300	277800*	

AIR VEE CANADA LIMITED
 TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7 0510 9
 SHEET NO. 15-13

AIRPLANE

C 105

CENTRE FUSELAGE

PREPARED BY

DATE

CHECKED BY

Dec. 55
 DATE

LOAD No. 13

STRESSES DUE TO 1 KIP PER SIDE AT LOCATION NO. 13

POINT	PART 1	PART 2	PART 3	PART 4
1	1491700	600200	7952800	11841600
2	204000	114100*	1915000*	1407500*
3	2207569	259044	2395795	2679744
4	245523*	46615*	208348*	489703*
5	188900*	12500*	18000*	70200
6	4111600*	295000	4489800	2926100
7	130671	7973	171027	353440
8	792938*	192687*	3155467*	5141674*
9	504800	501604	6975300	9488430
10	2767600*	606461*	8712200*	12034200*
11	2811890	390057	5607776	8274286
12		12559	656926	1811908
13	3049078	580375	9206487	12961953
14	314253*	151380*	1903708*	1018454*
15	2835499	255742	2382807	1191560
16	392016*	35311*	256185*	452969*
17	31391	286*	23930*	3819*
18	924529*	362593	4610859	1092987
19	902221	2107	194904	126330*
20	1318853*	230770*	3669831*	5505733*
21	2007158	594092	7596400	10469741
22	3983119*	695872*	9647925*	13287686*
23	3570324	417527	6190620	5031476
24	244588	27780	978972	3746886
25	4904550	665950	10503874	13665635
26	778239*	181853*	1752601*	
27	3280249	244331	2485353	
28	519073*	23678*	364069*	11607156
29	148592	10096*	22858*	14746365*
30	1480796	399667	4200942	
31	153994*	17791	246197	
32	1773456*	270339*	4375307*	
33	3524975	653013	8437056	2038721*
34	5093273*	766253*	10708127*	126330*
35	2666975	517411	7060186	
36	413475	32092	1346870	
37	126096	65668	1811483	
38	6002100	795200	11841600	
39	1140800*	191500*	1407500*	
40	124200*	1800*	70200	
41	2949600	449000	2926100	
42	5015940	697560	9488430	
43	6065410*	871260*	12034200*	

TECHNICAL DEPARTMENT (Aircraft)

SHEET NO.

15-14

AIRCRAFT

PREPARED BY

DATE

C 105

CENTRE FUSELAGE

CHECKED BY

DATE

LOAD No. 14

STRESSES DUE TO LIFT FOR SECTION NO. 14

STRESS
POINT

PART 1

PART 2

PART 3

PART 4

1	125900	250900	5165700	9089300
2	951900	16500	943200 *	970600 *
3	1307356	194928	2369977	2584233
4	107019 *	57412 *	369717 *	506162 *
5	1057200	13000 *	36900	77600
6	1356700 *	11100 *	2572100	2146500
7	661566 *	43257	291408	407685
8	536292 *	148858 *	2693537 *	4374499 *
9	681700	167468	4856000	7479960
10	1497000 *	266520 *	6070640 *	9435720 *
11	895371 *	402477	5357331	7578160
12		196687 *	38969	1693456
13	697252	296483	6433203	10163099
14	650840 *	31877 *	1088983 *	742131 *
15	1832079	221918	2400009	1363626
16	252588 *	53283 *	391030 *	451066 *
17	89434	6907	13413	5711
18	1456366 *	104469	2981493	861935
19	818258 *	27622	277686	122979 *
20	942747 *	184658 *	3173891 *	4672182 *
21	45040	317724	5589174	8401650
22	1194577 *	394273 *	7082720 *	10587502 *
23	740727 *	415494	5831596	4808716
24	1252025 *	115084 *	588308	3668340
25	1696203	391564	7748033	10888601
26	360107	69836 *	1111975 *	
27	1892684	230592	2476900	
28	483938 *	44109 *	437207 *	9501902
29	885245 *	3124 *	1545 *	11938197 *
30	2008650 *	178205	2920202	
31	441049	35489	307689	
32	1143616 *	225681 *	3752118 *	
33	494473 *	417421	6456986	2208113 *
34	1082506 *	507753 *	8163989 *	122979 *
35	2973103	502772	6555207	
36	2790103 *	63024 *	1111919	
37	1966966 *	4149	1692774	
38	2509100	516600	9089300	
39	164400	94300 *	970600 *	
40	130300 *	3700	77600	
41	111400 *	257200	2146500	
42	1674780	485640	7479960	
43	2665800 *	606974 *	9435720 *	

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7 0510 9

SHEET NO. 15-15

AIRCRAFT

C 105

CENTRE FUSELAGE

PREPARED BY

DATE

CHECKED BY

Dec 55

DATE

LOAD No. 15

STRESSES DUE TO 1 KIP PER SIDE AT LOCATION NO. 15

STRESS
DUE TO

	PART 1	PART 2	PART 3	PART 4
1	122000*	60900	2717000	6411100
2	777800	70200	70500*	526600*
3	457400	120268	2134660	2444724
4	10900*	19469*	439785*	495518*
5	463000	9200*	67000	88200
6	502500*	119800*	478700	1351900
7	258900*	60333*	484490	495731
8	257900*	112425*	2147214*	3539624*
9	483700	8400*	2519300	5423490
10	834200*	74700*	3410070*	6840040*
11	326500*	23566*	5246484	6928035
12		219595*	1057586*	1426003
13	7200*	107905	3864819	7416746
14	693900	38433	306159*	461009*
15	611300	138069	2291113	1499191
16	1000	35043*	446173*	431768*
17	156900	79290*	57856	23543
18	541100*	216226*	1263929	629382
19	362000*	59078	417966	113327*
20	540500*	134652*	2577548*	3771431*
21	255100	26288*	3455243	6297846
22	720100*	85053*	4618013*	7887118*
23	280900*	428096	5573072	4620253
24	497600*	333389*	143558*	3413593
25	244400	172830	5115291	8147467
26	689900	18344	473649*	
27	933300	183097	2393753	
28	61500*	42635*	466244*	7373040
29	93900	14564*	29665	9128527*
30	743700*	77015*	1598257	
31	529400*	61613	413977	
32	850900*	173276*	3057428*	
33	79500	143702	4399816	2314302*
34	701200*	226305*	5623344*	113327*
35	264500*	505852	6118928	
36	1196400*	221933*	644767	
37	2196200*	105647*	1425363	
38	608900	271800	6411100	
39	702300	7000*	526600*	
40	92400*	6700	88200*	
41	1197600*	47900	1351900	
42	83800*	251980	5423490	
43	747900*	341787*	6840040*	

AIR CANADA LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 70510-9
SHEET NO. 15-16

AIRCRAFT

C 105

CENTRE FUSELAGE

PREPARED BY

Dec. 55

CHECKED BY

DATE

LOAD No. 16

STRESS VALUES FOR LOAD No. 16

STRESS
VALUES

	PART 1	PART 2	PART 3	PART 4
1	89300*	3100	1036700	3833600
2	464600	60000	400500	101500*
3	132900	34200	1300941	2150114
4	4600	2600	282656*	455176*
5	164100	6900	911600*	102300
6	191200*	46300*	2559400*	458100
7	88000*	28900*	831671	663076
8	112900*	65100*	1424095*	2559850*
9	222500	9300	57500*	3330310
10	425200*	44300*	817300*	4255970*
11	96200*	6300	5233239	6331811
12		83500*	3283240*	795251
13	83000*	17500	1732836	4711092
14	451700	56200	290064	182686*
15	156900	64700	1797117	1525758
16	28700	3000*	391119*	393158*
17	69700	8700	71502*	60574
18	209500*	80500*	953138*	379431
19	135600*	42300*	699346	68275*
20	269000*	90900*	1806106*	2736781*
21	155100	5400	1181416	4172355
22	388500*	52500*	1985110*	5187838*
23	67900*	8200*	5442847	4506193
24	170100	138200*	1715123*	2736747
25	46200*	45800	2710947	5405232
26	504600	54400	91676	
27	237400	116700	2072001	
28	36000	15300*	439582*	5245177
29	64900	4300*	49177	6318859*
30	290800*	142800*	26617	
31	212500*	58600*	631466	
32	449600*	122000*	2203039*	
33	119100	200	2268250	2269093*
34	399600*	64400*	3105806*	68275*
35	58100*	6100*	5782340	
36	433200*	217600*	396283*	
37	835700*	328400*	794654	
38	30900	103700	3833600	
39	600800	40000	101500*	
40	68800	91200*	102300	
41	462700*	255900*	458100	
42	93300	5700*	3330310	
43	442900*	81800*	4255970*	

AVIATION CANADA LIMITED
 TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7 0510
 SHEET NO. 15-17

AIRCRAFT

C 105

CENTRE FUSELAGE

PREPARED BY

DATE

Dec. 55

CHECKED BY

DATE

LOAD No. 17

STRESSES ARE IN LBS PER SQ IN AT LOCATION NO. 17

STRESS
POINT

PART 1

PART 2

PART 3

PART 4

1	32100 *	3600 *	178500	1583600
2	200200	29800	404600	153900
3	25800	6000	577338	1355705
4	2600	2600	51004 *	334834 *
5	42800	3300	61500	75300 *
6	62100 *	15400 *	788900 *	818900 *
7	22100 *	9700 *	329205 *	1005822
8	39200 *	27300 *	744203 *	1344375 *
9	62600	5200	44200	1156640
10	168300 *	19300 *	376800 *	1667090 *
11	18200 *	400	275	5768585
12		23600 *	1127805 *	830502 *
13	38400 *	800 *	458527	2143738
14	203200	31700	390759	35437
15	21100	14000	1038648	1226424
16	12200	1900	159022 *	309555 *
17	22100	5200	87183 *	129105
18	69400 *	25900 *	1398436 *	57579
19	38000 *	15100 *	445855 *	26922 *
20	103000 *	39200 *	988314 *	1498831 *
21	49900	5600	26939	1985663
22	159600 *	23600 *	484878 *	2479355 *
23	4700 *		95037	4558332
24	42900 *	41900 *	1748256 *	1079001
25	42100 *	4600	993975	2677997
26	237800	37200	259894	
27	30500	29600	1067796	
28	19900	100	266197 *	3107316
29	23300	6700	960534 *	3508790 *
30	97900 *	44900 *	2494614 *	
31	63000 *	22700 *	1038384	
32	179100 *	54500 *	1108759 *	
33	48700	5400	10426	1852382 *
34	170100 *	29700 *	639573 *	26922 *
35	2200	800 *	5479071	
36	116500 *	70100 *	2588725 *	
37	236200 *	112700 *	630832 *	
38	36500 *	17900	1503600	
39	298700	40400	153900	
40	32900	6200	76300 *	
41	153700 *	78900 *	818900 *	
42	52900	4400	1156640	
43	192700 *	37700 *	1687090 *	

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7 0510 9

SHEET NO. 15-18

AIRCRAFT

C 105

CENTRE FUSELAGE

PREPARED BY

DATE

Dec. 55

CHECKED BY

DATE

LOAD No. 18

STRESS DET. TO 1 KIP PER SIDE AT LOCATION NO. 18

STRESS LBS.	PART 1	PART 2	PART 3	PART 4
1	7300 *	1200 *	8100	218500
2	36400	5500	89300	83300
3	4000	400	61100	382894
4	400	500	8100 *	106028 *
5	7800	600	20700	19300 *
6	6000 *	1300 *	48400 *	864900 *
7	3900 *	1900 *	62600 *	197065 *
8	6700 *	4600 *	124100 *	276024 *
9	9600	500	9700 *	87100 *
10	27100 *	3000 *	50200 *	78300 *
11	4300 *	500 *	53500 *	222543 *
12		4300 *	203700 *	747970 *
13	9300 *	800 *	37800	378104
14	36900	6100	102500	23368
15	1900	1300	118500	260587
16	2100	500	22100 *	122670 *
17	3800	1000	27400	3692 *
18	6300 *	1900 *	83500 *	413050 *
19	6900 *	2700 *	92800 *	55400
20	17700 *	6500 *	165600 *	257916 *
21	6700	200	23700 *	136714 *
22	25500 *	3400 *	59400 *	82346 *
23	3400 *	1400 *	80700 *	487744 *
24	7900 *	7400 *	332000 *	165568 *
25	11100 *	300 *	100100	505500
26	43300	7500	106500	
27	1300	2800	215400	
28	3500	200	54100 *	1118621
29	4200	1500	27100	105775 *
30	8200 *	2900 *	147500 *	
31	11200 *	4000 *	133200 *	
32	30400 *	9100 *	216400 *	
33	5500	200 *	47500 *	806937 *
34	26500 *	4100 *	68600 *	55400
35	3700 *	3000 *	136200 *	
36	21200 *	12500 *	496500 *	
37	42500 *	20300 *	748000 *	
38	11500 *	800	218500	
39	55100	9000	83300	
40	5800	2100	19300 *	
41	12600 *	4800 *	264900 *	
42	4700	1000	87100 *	
43	29500 *	5000 *	78300 *	

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO.

7 051 7

SHEET NO.

15-13

AIRCRAFT

PREPARED BY

DATE

C 105

CENTRE FUSELAGE

CHECKED BY

DATE

LOAD No. 19

STRESSES DUE TO 1 KIP PER SIDE AT LOCATION No. 19

STRESS POINT	PART 1	PART 2	PART 3	PART 4
1	15300	9400	142800	19400
2	64300*	11000*	216900*	346500*
3	2100*	900	79000*	653733*
4	6400*	6100*	129300*	107538*
5	12800*	1800*	61800*	38800*
6	5300	1300	18000	29900
7	5900	4300	173600	603292
8	11200	9200	273000	665596
9	61400*	7700*	85400*	40900
10	47200	5800	103100	123900
11	2600	700	147800	848343
12		8500	506600	2083647
13	52800	9600	152700	176594*
14	67100*	13100*	276900*	297598*
15	6100	100	175800*	1876635*
16	22000*	6400*	145400*	20845
17	9300*	2600*	85100*	16160
18	6100	1600	20700	47010
19	13300	6900	265900	157386*
20	31600	13500	376300	957628
21	60000*	8500*	71900*	157100
22	45900	7200	118100	98986
23	500*	2400	279200	1228434
24	11600	16700	833900	4725804
25	75100	11600	124400	730990*
26	82500*	16900*	323200*	
27	12700	2600*	358300*	
28	40600*	10600*	144400*	479542
29	11300*	4300*	90800*	39126*
30	9100	1600	23400	10000000
31	24700	11200	396600	
32	57000	19500	506300	
33	66400*	8700*	35900*	2164157
34	50100	8500	126900	157386*
35	400*	7100	485400	
36	37500	29600	1335500	
37	84400	50600	2083900	
38	93200	14300	19400	
39	110100*	21700*	346500*	
40	17700*	6200*	38800	
41	13100	1800	29900	
42	77100*	8500*	40900	
43	57900	10300	123900	

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

REPORT NO. 7-0510.9
SHEET NO. 15-20

AIRCRAFT

C 105

CENTRE FUSELAGE

PREPARED BY

DATE

Dec. 55

CHECKED BY

DATE

LOAD No. 20

STRESSES DUE TO 1 LB. PER SIDE AT LOCATION NO. 20

STRESS POINT	PART 1	PART 2	PART 3	PART 4
1	15900*	4200*	64500*	8800*
2	28900	5000	97700	156100
3	1000	400*	35600	294471
4	2800	2800	58100	48657
5	5800	800	27900	17300*
6	2400*	600*	8200*	13600*
7	2500*	2000*	78400*	271900*
8	5000*	4200*	123100*	300055*
9	27800	3500	38700	18300*
10	21200*	2600*	46500*	59700*
11	1200*	300*	66500*	382326*
12		3700*	228500*	939015*
13	23900*	4300*	68600*	79493
14	30300	5700	122000	134043
15	2800*		79100	710452
16	9700	3800	65600	9334*
17	4200	1400	38300	7280*
18	2800*	700*	9200*	21330*
19	5900*	3100*	119800*	70843
20	14400*	6200*	169500*	431676*
21	27000	3900	32700	70695*
22	20900*	3200*	53300*	44416*
23	300	1000*	125800*	553553*
24	5300*	7400*	375700*	2129720*
25	33700*	5400*	56100*	329170
26	37100	7700	145600	
27	5800*	1100	152500	
28	18200	4800	65100	215976*
29	5100	1900	40900	13152
30	4200*	700*	10600*	
31	11300*	5000*	178700*	10000000
32	25800*	8700*	228100*	
33	30100	3900	16200	975200*
34	22600*	3900*	57200*	70843
35	200	3100*	218800*	*
36	16700*	13300*	601700*	
37	38000*	22800*	939100*	
38	41900*	6500*	8800*	
39	49700	9800	156100	
40	6000	2800	17300*	
41	6000*	800*	13600*	
42	34600	3900	18300*	
43	25200*	4700*	55700*	

A. V. ROE CANADA LIMITED
 (INCORPORATED IN CANADA)
 TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/9
 SHEET NO. 15-21

AIRCRAFT

C 105

CENTRE FUSELAGE

PREPARED BY

DATE

Dec. 55

CHECKED BY

LOAD No. 21

STRESS IS DUE TO 1 KIP PER SIDE AT LOCATION No. 21

STRESS POINT	PART 1	PART 2	PART 3	PART 4
1	122700*	32500*	498900*	67800*
2	224400	38400	757300	1210400
3	6500	3500*	275100	2283446
4	28400	21500	451100	375712
5	44800	6200	215800	135900*
6	18600*	4600*	63200*	104500*
7	20500*	15000*	606100*	2106996*
8	38900*	32000*	953500*	2323918*
9	214300	26900	298400	1428000*
10	164900*	20300*	360100*	432600*
11	9600*	2300*	516400*	2962962*
12		29600*	1769400*	7277116*
13	164300*	33100*	833700*	616593
14	234100	45200	946100	1039173
15	21400*	300*	613300	5506050
16	76500	29400	507500	72746*
17	32400	9400	297100	56240*
18	21600*	5300*	72300*	164000*
19	46300*	24100*	928600*	549797
20	110100*	47300*	1313900*	3344609*
21	209700	29700	251100	548460*
22	160100*	24800*	412600*	345656*
23	2300	8500*	975200*	4290091*
24	40700*	58100*	2918200*	16503933*
25	261800*	40700*	434400*	2552350
26	283100	59800	1128600	
27	44600*	9000	1181100	
28	142100	37600	504500	1674171*
29	39700	14700	317000	101573
30	31700*	5100*	82100*	
31	66500*	38900*	1385200*	
32	199200*	67900*	1763000*	10000000
33	231900	30700	125300	7557772*
34	175200*	30100*	443500*	549797
35	1000	24500*	1695100*	
36	130900*	103500*	4663200*	
37	294800*	176700*	7277500*	
38	325200*	49900*	67800*	
39	384400	75700	1210400	
40	61800	21600	135900*	
41	45600*	6300*	104500*	
42	269200	29800	142800*	
43	202500*	36100*	432600*	

AIR VITE CANADA LIMITED
 (INCORPORATED IN CANADA)
 TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 70519

SHEET NO.

15-32

PREPARED BY

DATE
Dec. 55

CHECKED BY

C 105

CENTRE FUSELAGE

LOAD No. 22

STRESS IN TENSILE STRESS IN COMPRESSIVE STRESS IN SHEAR

STRESS IN T	PART 1	PART 2	PART 3	PART 4
1	149500*	25100*	53500*	935800*
2	467300*	50900*	1881600*	5791500*
3	74900*	25900*	794800*	2269445*
4	39600*	32700*	581900*	623001*
5	47900*	800*	13700*	105100*
6	51400*	11500*	234900*	235500*
7	25800*	4200*	47600*	228690*
8	42700*	46100*	1434700*	3999211*
9	439700*	42600*	408600*	339100*
10	298500*	38300*	846400*	2036600*
11	43500*	11600*	443700*	899045*
12		22900*	508600*	454775*
13	206300*	18000*	98400*	1558500*
14	504900*	97400*	2702700*	7695261*
15	138800*	37900*	1143500*	2408469*
16	129800*	41800*	638700*	584726*
17	15900*	400*	42200*	33800*
18	60200*	14500*	278500*	67915*
19	36800*	4700*	8200*	12247*
20	158300*	70700*	2042900*	5396730*
21	412500*	42600*	400600*	286436*
22	303600*	48900*	1127500*	2558285*
23	59200*	18100*	600500*	499167*
24	51700*	30800*	598400*	276883*
25	248400*	13900*	403500*	2262252*
26	618700*	135200*	3960700*	10000000*
27	222400*	55300*	1645200*	
28	225100*	50500*	658700*	244763*
29	11100*	500*	84000*	3252930*
30	61400*	19400*	298600*	
31	55700*	5800*	84400*	
32	292500*	101000*	2925200*	
33	410900*	41400*	381300*	966609*
34	333700*	63900*	1508600*	12247*
35	79600*	31100*	764300*	
36	122900*	39900*	614100*	
37	228300*	50900*	455400*	
38	250700*	6300*	935800*	
39	809100*	188200*	5791500*	
40	7600*	1400*	105100*	
41	114800*	23500*	835500*	
42	426200*	40900*	339100*	
43	332900*	84600*	2036600*	

AIR ROE CANADA LIMITED
 (INCORPORATED IN CANADA)
 TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7 0510 9

SHEET NO. 15-43

AIRCRAFT

C 105

CENTRE FUSELAGE

PREPARED BY

DATE

Dec. 55

CHECKED BY

DATE

LOAD No. 23

STRESSES DUE TO LOAD SET AT LOCATION NO. 23

STRESS POINT	PART 1	PART 2	PART 3	PART 4
1	51100 *	13500 *	207300 *	23000 *
2	93100 *	15900 *	314800 *	502900 *
3	2700 *	1500 *	114600 *	948750 *
4	9200 *	9100 *	187600 *	156237 *
5	10600 *	2600 *	89600 *	56500 *
6	7800 *	1900 *	26100 *	43600 *
7	8500 *	6200 *	251900 *	875536 *
8	16200 *	13400 *	396200 *	965561 *
9	89500 *	11100 *	124000 *	59400 *
10	68600 *	8500 *	149600 *	172800 *
11	4100 *	900 *	214700 *	1231270 *
12		12300 *	735400 *	3023788 *
13	76900 *	13600 *	221700 *	256293 *
14	97200 *	18900 *	393100 *	431706 *
15	8900 *	100 *	254900 *	228800 *
16	31400 *	12100 *	210800 *	30200 *
17	13400 *	4000 *	123600 *	23510 *
18	8900 *	2200 *	30200 *	68160 *
19	19200 *	9900 *	386800 *	228436 *
20	45800 *	19600 *	545800 *	1389842 *
21	87500 *	12500 *	104200 *	227900 *
22	66500 *	10300 *	171300 *	143627 *
23	900 *	3700 *	405200 *	1782812 *
24	17000 *	24100 *	1210100 *	6857997 *
25	108800 *	16800 *	180400 *	1060850 *
26	119700 *	24800 *	469000 *	
27	18600 *	3700 *	490900 *	
28	58900 *	15500 *	209700 *	695743 *
29	16500 *	6200 *	131700 *	42286 *
30	13100 *	2300 *	34000 *	
31	38900 *	16300 *	575700 *	
32	82800 *	28100 *	734600 *	
33	96300 *	12700 *	52000 *	685939 *
34	72800 *	12600 *	184200 *	228436 *
35	800 *	10200 *	704300 *	
36	54300 *	43100 *	1937600 *	
37	122500 *	73500 *	3024000 *	
38	135100 *	20800 *	28000 *	
39	159700 *	31500 *	502900 *	
40	25700 *	9000 *	56500 *	
41	19000 *	2600 *	43600 *	
42	111800 *	12400 *	59400 *	
43	84100 *	15000 *	179800 *	

TECHNICAL DEPARTMENT (Aircraft)

C 105

CENTRE FUSELAGE

REPORT NO.

7 0510 9

SHEET NO.

10-1

PREPARED BY

CHECKED BY

DATE

Dec. 55

DEFLECTIONS

DUE TO LIGHT LOADS

INCHES PER KIP PER SIDE

(EIGHT PLACES OF DECIMALS)

AT LOAD - 10"	LOAD 1	LOAD 2	LOAD 3	LOAD 4	LOAD 5	LOAD 6
1	46129128	24339153	20743273	17353923	14290150	11508992
2	24339322	14359819	12587447	10814352	9131240	7532518
3	20743367	12587361	11429716	9868911	8352357	6932886
4	17354466	10814442	9869094	8929216	7602005	6342159
5	14290956	9131718	8362528	7602283	6888984	5788796
6	11509706	7532574	6933150	6342028	5788805	5196540
7	9001790	6031683	5379213	5134864	4717382	4280412
8	6766835	4645851	4320996	4001514	3701726	3391976
9	4803512	3388413	3170796	2957771	2737507	2553619
10	3113299	2272798	2143746	2017957	1901128	1783728
11	1685981	1305474	1247549	1192765	1143069	1095446
12	789840	687800	673540	661660	653820	649040
13	24339322	14359819	12587447	10814352	9131240	7532518
14	17289894	10760302	9598511	8402218	7274142	6115339
15	11450699	7496032	6812583	6109635	5373988	4624920
16	6662346	4559553	4203208	3838137	3459559	3068031
17	2925972	2096064	1957065	1815027	1669352	1516904
18	497790	385330	366770	347120	327070	306020
19	116670	166580	175340	185290	196650	210690
20	52640*	74740*	79150*	83360*	88660*	95060*
21	408310*	580670*	611920*	646080*	686040*	735480*
22	1225660*	828170*	766310*	704640*	645600*	580010*
23	169470*	241360*	254130*	268390*	285150*	305700*

DEFLECTIONS

TECHNICAL DEPARTMENT (Aircraft)

C 105

CENTRE FUSELAGE

REPORT NO. 7-0510/9

SHEET NO. 10-2

PREPARED BY DATE
CHECKED BY DATE
Dec. 55

DEFLECTIONS DUE TO UNIT LOADS INCHES PER KIP PER SIDE

AT LOAD POINT	LOAD 7	LOAD 8	LOAD 9	LOAD 10	LOAD 11	LOAD 12
1	9001230	6766207	4803214	3112983	1685806	790080
2	6031124	4645699	3388373	2272642	1304908	687560
3	5579068	4320549	3170750	2143677	1247428	673700
4	5134504	4001795	2957718	2017790	1192419	661740
5	4717228	3701581	2757624	1900853	1142729	653580
6	4280317	3391271	2553640	1783557	1095329	648990
7	3846930	3090278	2358016	1673792	1054429	650450
8	3090172	2793401	2171603	1573825	1023019	661380
9	2358192	2171649	1989444	1484085	1003793	685140
10	1673839	1573880	1484289	1401660	999367	727550
11	1084751	1022937	1003836	999360	1008621	793480
12	650390	661310	685160	727580	793320	870380
13	6031124	4645699	3388373	2272642	1304908	687560
14	4981204	3900095	2891518	1976390	1167819	646440
15	3872601	3110125	2362640	1656657	1014829	593590
16	2650857	2197807	1743903	1274825	820319	511980
17	1355366	1178758	980780	752141	531324	364750
18	283380	258830	230980	198730	159480	123150
19	228350	251800	283050	326400	386460	449490
20	102790*	113280*	127480*	147300*	174100*	202530*
21	797540*	878490*	988540*	1139800*	1349450*	1569830*
22	512570*	441000*	362040*	270270*	156420*	49680*
23	331360*	365220*	410770*	473940*	560690*	652400*

DEFLECTIONS

TECHNICAL DEPARTMENT (Aircraft)

AIR FORCE CANADA LIMITED

REPORT NO.

Z 0510/9

SHEET NO.

16-3

AIRCRAFT

C 105

CENTRE FUSELAGE

CHECKED BY

DATE

DEFLECTIONS

DUE TO UNIT LOADS

INCHES PER KIP PER SIDE

UNIT: 1 TON = 22.24 KIP

AT LOAD POINT	LOAD 1	LOAD 14	LOAD 15	LOAD 16	LOAD 17	LOAD 18
1	24339153	17289455	11449939	6661893	2925732	497680
2	14359819	10760011	7495932	4559527	2095898	385420
3	18587361	9598431	6812392	4202978	1956986	366570
4	10814442	8402016	6109737	3838128	1815152	347180
5	9131718	7274325	5373885	3459416	1669168	326980
6	7532574	6115628	4625057	3067997	1517125	305910
7	6031683	4981364	3873063	2650757	1355313	283580
8	4645851	3899714	3110345	2197707	1178899	258920
9	3388413	2891571	2362419	1743949	980597	230890
10	2272798	1976257	1557028	1274880	752519	198620
11	1305484	1168165	1014946	820537	531306	159370
12	687800	646360	593540	512110	364680	123250
13	14359819	10760011	7495932	4559527	2095898	385420
14	10760302	8768453	6266637	3894928	1830352	349820
15	7496032	6266835	5062675	3249197	1571925	315810
16	4559553	3894937	3249131	2628707	1326799	285620
17	2096064	1830527	1572104	1326958	1092168	263720
18	385330	349720	315920	285630	263630	257380
19	166580	180390	188690	185300	151200	65880
20	74740*	81060*	85160*	83480*	68000*	29760*
21	580670*	628980*	659380*	646390*	527800*	229910*
22	828170*	688440*	539610*	374100*	189970*	31880*
23	2411360*	261290*	274200*	268720*	219440*	95560*

DEFLECTIONS

DEFLECTION

DUE TO UNIT LOADS

HUNDS PER KIP PER SIDE

(RIGHT PLACES OF DECIMAL)

AT LOAD POINT	LOAD 17	LOAD 18	LOAD 21	LOAD 22	LOAD 23
1	116890	52740*	408540*	1225560*	169770*
2	166400	75040*	580900*	828010*	241480*
3	175250	78950*	611950*	766140*	254340*
4	185100	83460*	646400*	704580*	268590*
5	196550	88560*	686250*	645620*	285150*
6	210700	94970*	735610*	579940*	305710*
7	228450	102880*	797660*	512290*	331360*
8	251610	113380*	878710*	440710*	365120*
9	283060	127590*	988560*	362250*	410680*
10	326310	147100*	1140010*	284470*	473530*
11	386460	174100*	1342460*	21110*	560790*
12	449530	202530*	1570040*	1010*	652540*
13	166400	75040*	580900*	1010*	241480*
14	180200	81160*	629200*	560*	261390*
15	188800	85070*	659510*	640*	274110*
16	185210	83480*	646510*	4010*	268720*
17	151110	68100*	527910*	9770*	219430*
18	65830	29560*	230020*	31780*	25460*
19	868110	307900	1279820*	78860	559750*
20	307910	360270	675200	35620*	252340
21	1279820*	675210	11008210	275450*	1954820
22	78670	35430*	275460*	1121510	114550*
23	559740*	252340	1954810	114350*	812200

DEFLECTIONS

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

REPORT NO. 7/0510/9

SHEET NO. 5-1

AIRCRAFT:

Centre Fuselage
Analysis.

PREPARED BY

DATE

A. Grze.

April 1955

CHECKED BY

DATE

COMPUTATION PROGRAMME.

Since the Centre Fuselage Analysis presents a large scale problem all computations have been organized according to principles of partitioning. Thus the problem has been reduced to a solution of

4 small partial problems with matrix order for inversion 14, 14, 14, 6 and of

1 general problem with matrix order for inversion 19.

Because while partitioning three unknowns had to be added the expenditure of punch cards was brought down from

$$(14 + 14 + 14 + 6 + 19 - 3)^3 = 262144$$

to

$$14^3 + 14^3 + 14^3 + 6^3 + 19^3 = 15307$$

A) Indices and Matrices given:

1) Indices

Stress points	i, k	101 - 143	1 st prob.
		201 - 243	2 nd prob.
		301 - 343	3 rd prob.
		401 - 426	4 th prob.
Redundancies	p, q	101 - 114	1 st prob.
		201 - 214	2 nd prob.
		301 - 314	3 rd prob.
		401 - 406	4 th prob.

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

REPORT NO. 7/0510/9

SHEET NO. 5-2

AIRCRAFT:

Centre Fuselage
Analysis.

PREPARED BY

DATE

A. Grze. *Rev. C. B.*

CHECKED BY

DATE

Loads $a, b = 1, 2, \dots, 23$ (always)
Reactions $a, b = 101 - 110$ in 1st prob.
 $106 - 115$ in 2nd prob.
 $111 - 120$ in 3rd prob.
 $116 - 120$ in 4th prob.

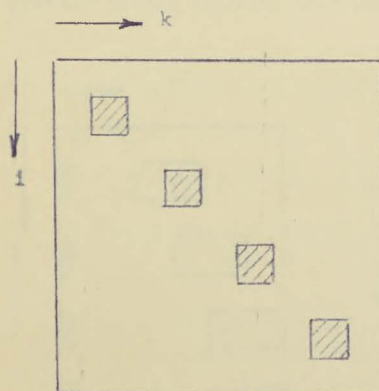
In the general problem

Loads $a, b = 1, 2, \dots, 23$
Reactions $u, v = 1, 2, \dots, 20.$

2) Matrices given:

Overall Elastic Influence Coefficients

C_{ik}



426 x 426

Only the elements contained in the index range indicated by shaded areas are different from zero.

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO 7/0510/9

SHEET NO 5-3

AIRCRAFT:

Centre Fuselage
Analysis.

PREPARED BY

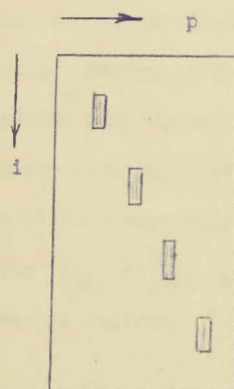
DATE

A. Grze.

CHECKED BY

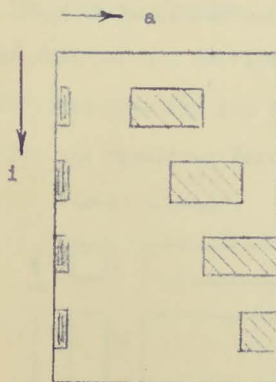
DATE

Overall Stress to Redundant, Load Relationship



426 x 406

Overall Stress to Unit Load Relationship (partial problems)



426 x 120

Only the elements contained in the index range indicated by shaded areas are different from zero.

TECHNICAL DEPARTMENT (Aircraft)

REPORT No 7/0510/9

SHEET No 5-4

AIRCRAFT:

Centre Fuselage
Analysis.

PREPARED BY

DATE

A. Grze.

Rev. 6.3

CHECKED BY

DATE

B) Solution of a partial Problem.

By suppressing the "hundred" digits in indices i, k and p, q the following matrices can be obtained separately:

4 Matrices Stress to Unit Load S_{ia}

4 Matrices Displacement to Unit Load Z_{ab}

For method of computation see Stress Rep. 7/0510/8 Tank 3 Analysis, in part. sheet 102 D) Computation required: 1).....

Denoting $H_{ip} = C_{ik} K_{kp}$ and observing rules of transposing matrices compute as follows:

$$S_{ia} = (b_{ik} - K_{ip} D_{pq}^{-1} H_{qk}) T_{ka}$$

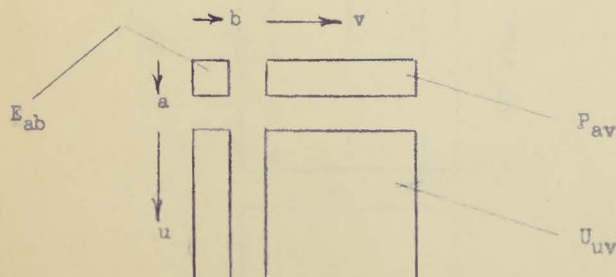
(print results)

$$Z_{ab} = T_{ai} C_{ik} S_{kb}$$

C) Solution of the General Problem. Displacement Matrix.

1) Add four partial Matrices Z_{ab} as shown on the next page.

The Matrices have the non zero elements as indicated by shaded areas. Then split the resulting Matrix as indicated on this page.



In doing so the range of $a, b = 101, 102, \dots, 120$ has to be replaced by the range of $u, v = 1, 2, 3, \dots, 20$. The range of $a, b = 1, 2, \dots, 23$ remains the same, however.

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/9

SHEET NO. 5-6

AIRCRAFT:

Centre Fuselage
Analysis.

PREPARED BY

DATE

A. Grze.

CHECKED BY

DATE

2) Consider the new Matrices:

Combined Influence Coefficients

Gross Product

Initial Influence Coefficients

U_{uv}

P_{ua}

E_{ab}

Compute the following:

a) Invert

$$U_{uv} \text{ to } U_{vu}^{-1}$$

b) Multiply

$$R_{ua} = - U_{uv}^{-1} P_{va}$$

c) Multiply

$$E_{ab}^* = P_{au} R_{ub} \quad (= P_{au} U_{uv}^{-1} P_{vb})$$

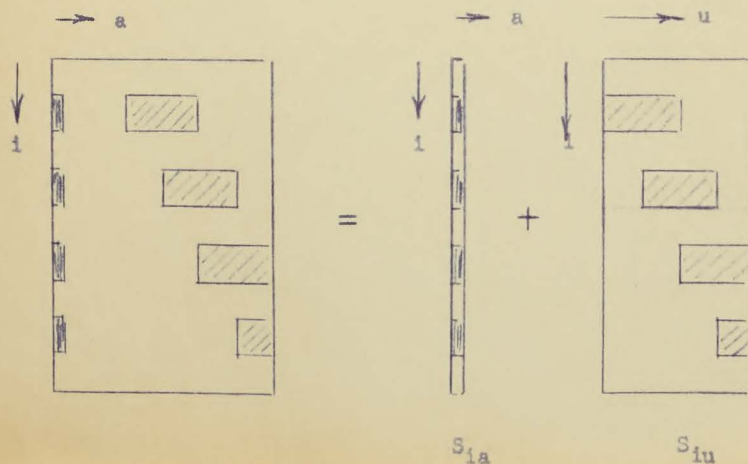
d) Add

$$E_{ab} + E_{ab}^* \quad (\text{print})$$

This quantity enters the general wing analysis.

D) Solution of the General Problem. Stress Matrix.

1) The four partial stress matrices when combined into one overall matrix appear as shown in figure below. After the range a, b is split, see paragraph C), two matrices are obtained S_{ia} and S_{iu} .



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

REPORT NO. 7/0510/9

SHEET NO. 5-10

AIRCRAFT:

C105

CENTOS FIBREGLASS

PREPARED BY

DATE

C.B.

Oct 55

CHECKED BY

DATE

CONTAINING FORMS

4. CHECK SUMS

ADD ELEMENTS FOR EACH OF THE ABOVE MATRICES TO GIVE CHECK SUMS & PLACE AS FOLLOWS

SUM OF ELEMENTS 1 TO 23 INC = ELEMENT 24

SUM OF ELEMENTS 51 TO 70 INC = ELEMENT 71

ADD = ELEMENT 72*

ADD BOTH BY ROWS & BY COLUMNS.

*NOTE $24, 24 + 71, 71 + 24, 71 + 71, 24 = 72, 72$

5. INVERSE U_{00} TO U_{00}^{-1} (20 by 20)

NOTE: ROW & COLUMN NO 55 ALL ZERO ELEMENTS

6. MULTIPLY $R_{00} = I - U_{00}^{-1} R_0$

$$\begin{vmatrix} | & | & | \\ | & \times & | \\ | & | & | \end{vmatrix} \quad (20 \times 20) \times (20 \times 23) = 20 \times 23$$

Note change of sign.

7. MULTIPLY $E_{00}^* = P_{00} \cdot R_{00} = -P_{00} \cdot U_{00}^{-1} \cdot R_0$ (Symmetric)

$$(23 \times 20) \times (20 \times 23) = (23 \times 23)$$

8. ADD. $E_{00}^{**} = E_{00} + E_{00}^*$ (Symmetric - Point)

9. P.E.-INDEX LOAD NUMBERS IN SIG MATRICES

As in Step 1

10. SPLIT

By SECOND INDEX ONLY

INTO SECOND INDEX 1 TO 23 INCLUSIVE
SECOND INDEX 51 TO 70 INCLUSIVE

SIG

SIG



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/9

SHEET NO. 6-2

AIRCRAFT:

C105

CENTRE FUSELAGE

PREPARED BY

DATE

CB

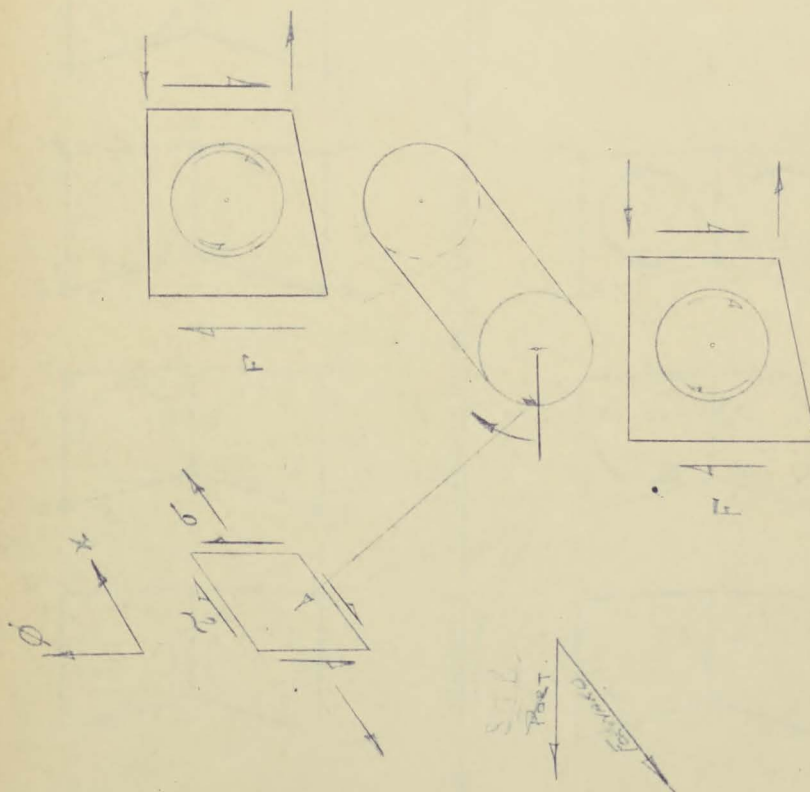
May 3 55

CHECKED BY

DATE

SIGN CONVENTION - DIST.

Positive as shown.



NOTE: STRESSES ON THE FRAMES PRODUCE EQUIVALENT ENERGY IN THE FRAME TO THE "F" LOADS.



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/9

SHEET NO. 7-1

AIRCRAFT

9105

CENTRE FUSELAGE

PREPARED BY

C.B.

DATE

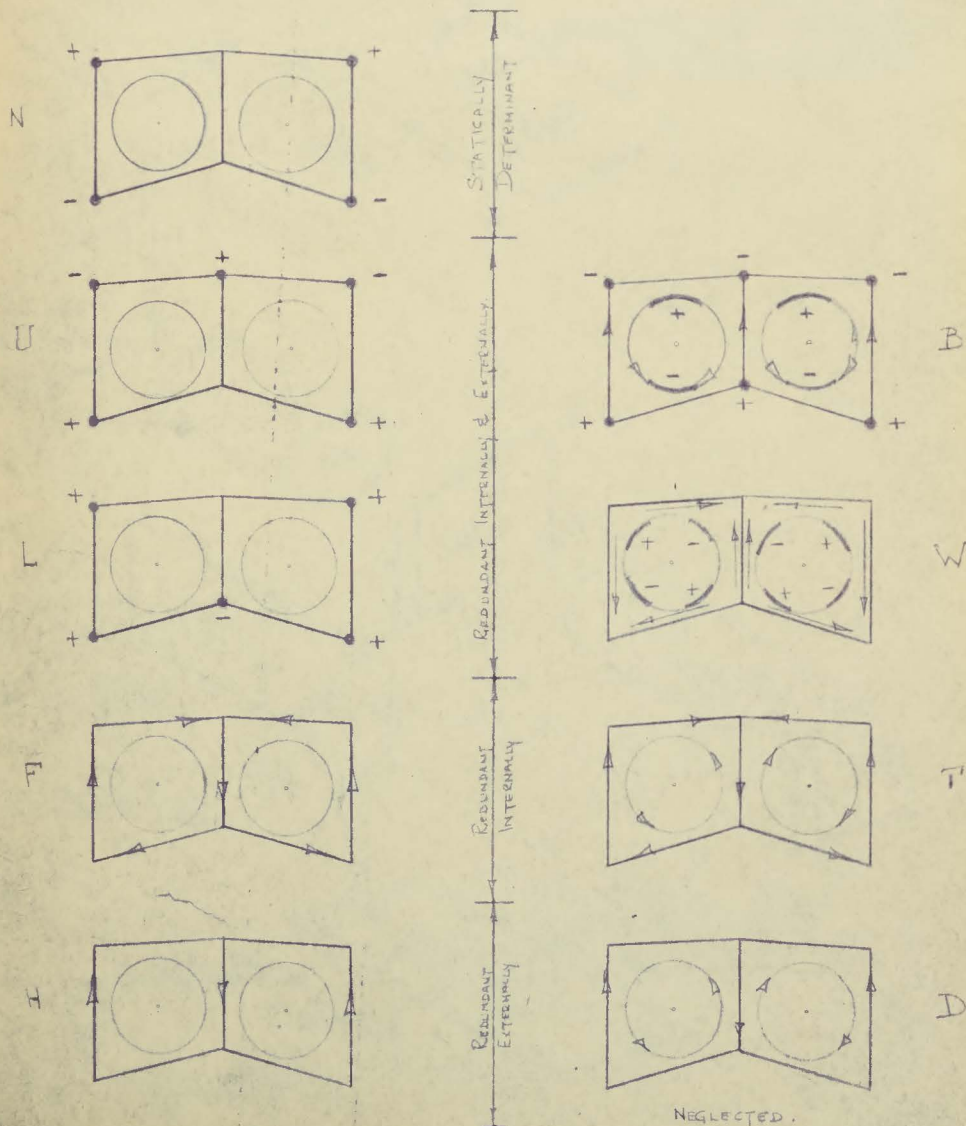
May 4/55.

CHECKED BY

DATE

REDUNDANT GROUPS.

SEVEN TYPES OF REDUNDANT GROUPS WERE CONSIDERED TO
ACT ON THE STRUCTURE.





AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO 710510/9

SHEET NO 7-3

AIRCRAFT

3105

CENTRE FUSelage

PREPARED BY

CB.

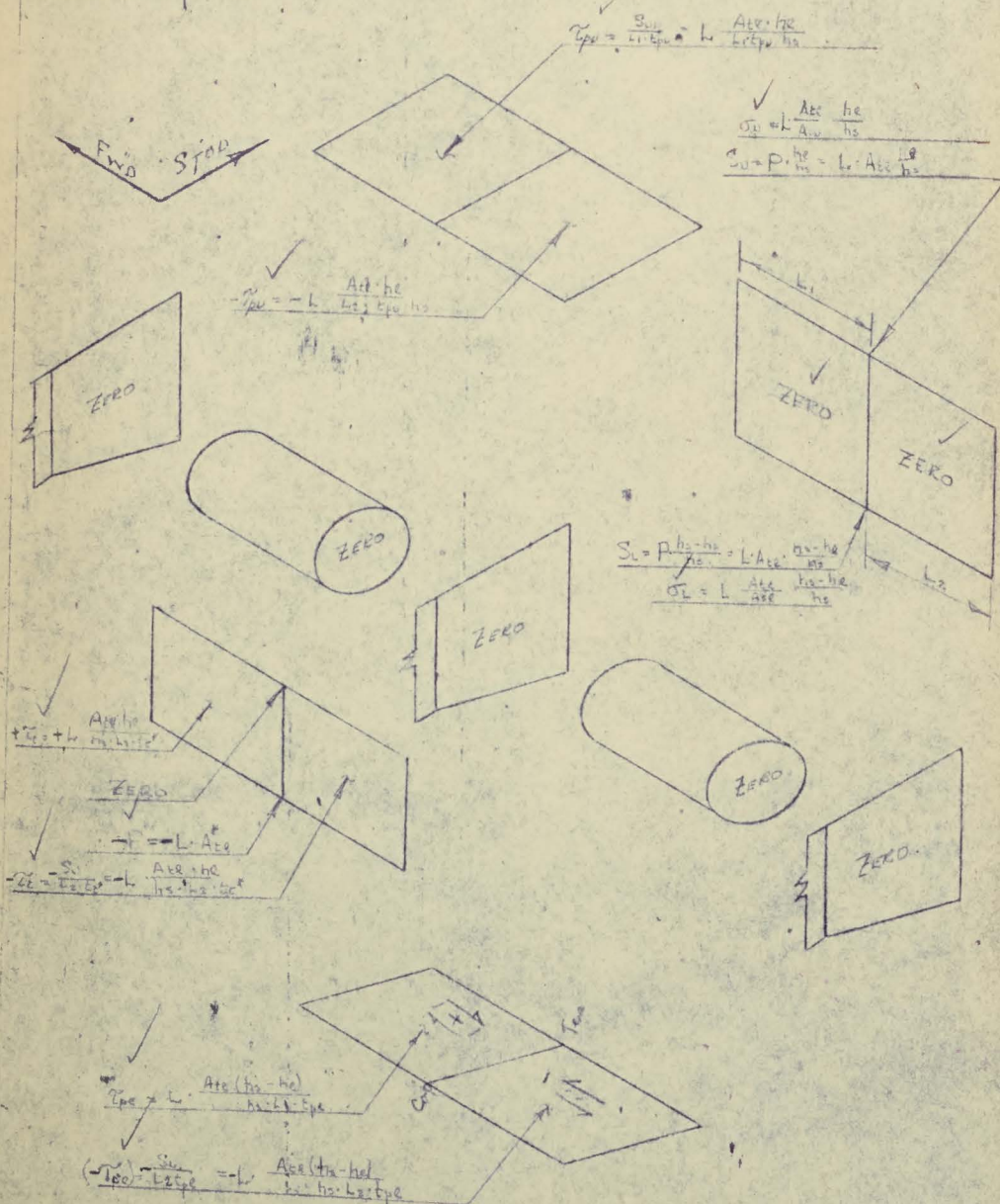
DATE

May 3.55

CHECKED BY

DATE

L GROUP



$$T_{pu} \cdot t_{pu} + T_{pe} \cdot t_{pe} = \pm (S_{pu} + S_{pe})/L = P/4$$



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/9

SHEET NO. 3-2

AIRCRAFT

C-105

PREPARED BY

DATE

B. AUGUSTINE

18/MAY/55

CHECKED BY

DATE

B

JUNE/55

1. ACTUAL STRUCTURE

THE TABLE ON THE RIGHT-HAND SIDE OF THE DRAWING GIVES CROSS-SECTIONAL AREAS OF LONGITUDINAL MEMBERS. ALL AREAS QUOTED ARE EQUIVALENT ALUMINUM AREAS.

THE AREAS OF MEMBERS —

- UPPER INNER LONGERON (U.I.L.)
- TOP SURFACE STRINGERS (Nos. 2 → 8)
- Side " " (Nos. 9 → 16)
- BOTTOM OUTER LONGERON (B.O.L.)
- STRINGER No. 17.
- ARE THE AVERAGE VALUES FOR TWO CASES, ONE TENSION ON TOP & ONE COMPRESSION ON TOP, FOUND FROM A PRELIMINARY ANALYSIS (FRONT CENTRE SECTION GROUP-REQ. No. 0554/5) OF THE STRUCTURE CONCERNED. THE VALUES AVERAGED WERE FROM CASE 2.1(3) (ROOM TEMP. WT = 4700#, TENSION ON TOP) & CASE 3/4 + V.E.B.M. (ROOM TEMP. WT = 4700#, COMPRESSION ON TOP).
- THE PRELIMINARY ANALYSIS, MENTIONED ABOVE, WAS DONE ON A STATICALLY DETERMINATE BASIS. THAT IS, A DETERMINATE LOAD PATH WAS CHOSEN & THIS PATH WAS SUCH THAT LOADS TRAVELLED OUTBOARD TO THE FUSE. SIDE & THENCE FORWARD & AFT. HENCE AREAS ON TANK SIDE, BELOW U.I.L. & ON ARMAMENT



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/9

SHEET NO. 8-3

AIRCRAFT:

C-105

PREPARED BY

DATE

E. AUGUSTINE

18/MAY/55

CHECKED BY

DATE

3.

JUNE 55

RAY FLOOR, OTHER THAN STRINGER "M", WERE
CONSIDERED HIGH-EFFECTIVE.

THE ABOVE HIGH-EFFECTIVE AREAS ARE
CONSIDERED IN THIS ANALYSIS & COMPUTATIONS
APPEAR REASONABLE. IN THESE COMPUTATIONS, AS IN
THOSE INVOLVING THE PRELIMINARY ANALYSIS,
EFFECTIVE AREAS ARE TAKEN AS BEING AN
AVERAGE OF TWO—

— LONGITUDINAL MEMBER AREA } SIMULATING COMPRESSION
+ 30" WIDTH OF SKIN

— LONGITUDINAL MEMBER AREA } SIMULATING TENSION
+ > 30" WIDTH OF SKIN

(FACTOR .6 TO CONVERT MAG. AREAS TO EQUIV. AN. AREAS)

1. INTERCOSTALS ON ARMAMENT RAY FLOOR. —

(DESIGNATED AS 18, 19, 20, 21, 22, REF. DWG. 7/0510/9)

FIG 1

THESE MEMBERS ARE TEST 7 SECTIONS WITH
FRANGES BRACED AT FORWARD STATIONS, BUT
HAVING CONTINUOUS SHEAR CONNECTIONS.

BASIC AREA IS CONSIDERED TO BE THE
FLOOR CONNECTING FRANGE OF 4. SECTION.

$$\text{BASIC AREA} = .55 \times .032 = .0176$$

$$30" \text{ WIDTH SKIN (MAG.) (COMPRESSION)} = 30 \times .040 = 1.2 \text{ IN.}$$

$$\text{WIDTH SKIN (TENSION)} = 7.0 \text{ IN.}$$

$$\text{AREA SKIN (COMPRESSION)} = 1.2 \times .04 \times .6 = .0288$$

$$\text{" " (TENSION)} = 7.0 \times .04 \times .6 = .1680$$

$$\text{EFFECTIVE AREA} = .0176 + \left(\frac{.0288 + .1680}{2} \right) = .116 \text{ IN.}^2$$



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT:

C-105

REPORT NO. 7/0510/9

SHEET NO. 84

PREPARED BY

DATE

E. AUGUSTINE

18/MAY/55

CHECKED BY

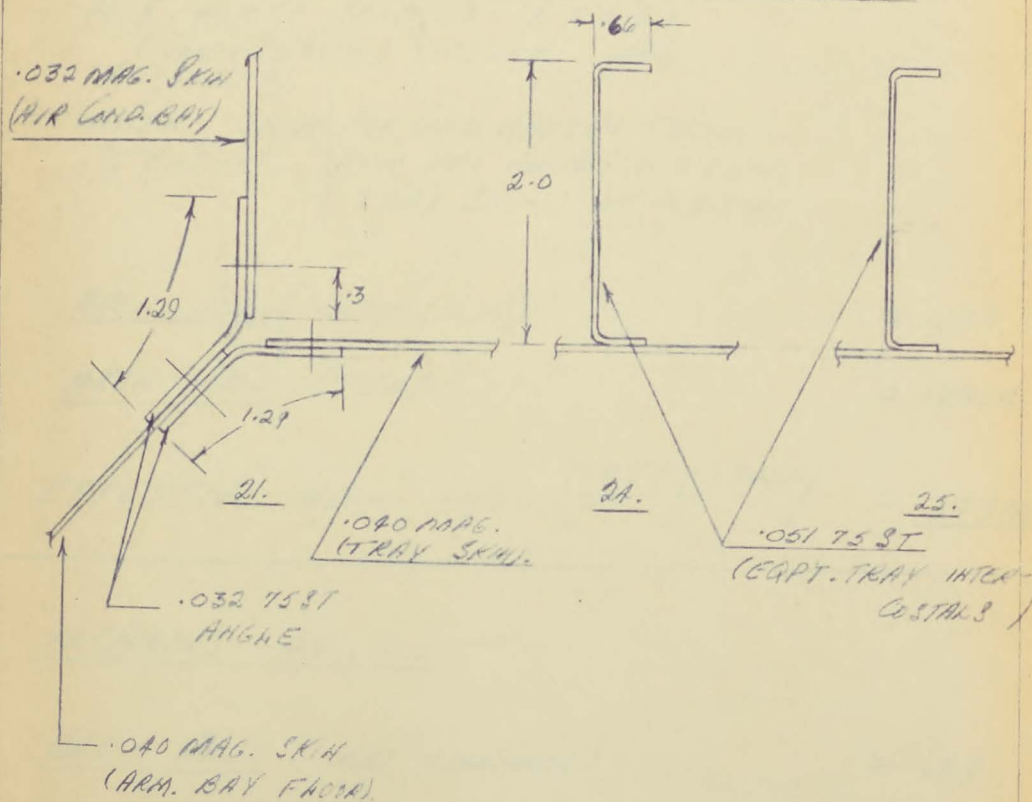
DATE

B

JUNE '55

MEMBERS 19 & 20 ARE IN FACT BROKEN AT STATION 315 (REF. DWG. 7/0510/9 FIG. 1), BUT ARE HERE ASSUMED CONTINUOUS.

2. MEMBERS 21, 24, 25 FORWARD OF STA. 315.





AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0510/9

SHEET No. 8-5

AIRCRAFT.

C-105

PREPARED BY

DATE

E. AUGUSTINE

12/MAY/55

CHECKED BY

DATE

B

JUNE '55

MEMBER 21 -

$$\text{BASIC AREA (ANGLES + } 3 \times 3 \text{ IN SKIN)} = .103$$

$$\begin{array}{l|l} 15 \text{ T WIDTH SKIN} & .04 \times 15 = 0.6 \text{ IN} \\ \text{(COMPRESSION)} & .032 \times 15 = .48 \text{ IN} \\ \text{(REF. SHEET 3).} & \end{array}$$

$$\begin{array}{l|l} \text{WIDTH SKIN} & \text{AIR GND. BAY: } 10 = 2.0 \text{ IN.} \\ \text{(TENSION)} & \text{ARM. BAY FLOOR: } 10 = 5.0 \text{ IN.} \\ & \text{TRAY SKIN: } 10 = 3.5 \text{ IN.} \end{array}$$

$$\text{AREA SKIN (COMPRESSION)} = .038$$

$$\text{AREA SKIN (TENSION)} = .2424$$

$$\text{EFFECTIVE AREA} = .103 + \left(\frac{.038 + .2424}{2} \right) = .243 \text{ IN}^2$$

MEMBERS 24, 25 -

$$\text{BASIC AREA (.051 CHAMFER)} = .169$$

$$30 \text{ T WIDTH SKIN (COMPRESSION)} = 33 \times .00 = 1.2 \text{ IN}$$

$$\text{WIDTH SKIN (TENSION)} = 7.0 \text{ IN.}$$

$$\text{AREA SKIN (COMPRESSION)} = .0288$$

$$\text{AREA SKIN (TENSION)} = .168$$

$$\text{EFFECTIVE AREA} = .169 + \left(\frac{.0288 + .168}{2} \right) = .267 \text{ IN}^2$$



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/2

SHEET NO. 8-6

AIRCRAFT:

C-105

PREPARED BY

DATE

E. AUGUSTINE

19/MAY/55

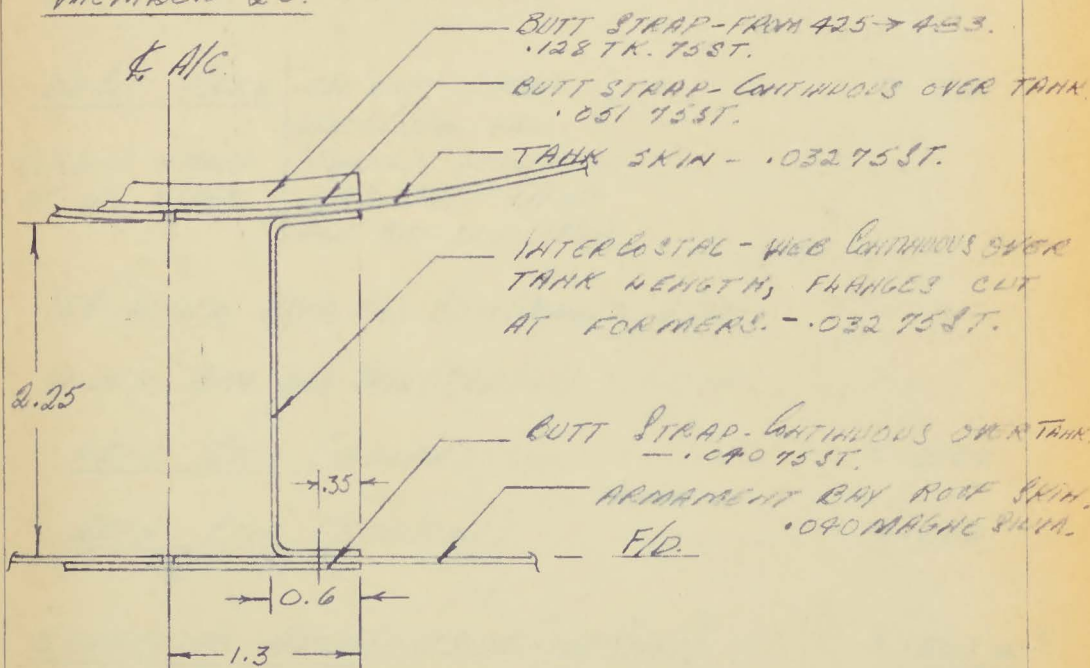
CHECKED BY

DATE

B

JUNE '55

MEMBER 23.



TANK SKINS ARE NOT INCLUDED IN AREA CALCULATIONS AT THIS STAGE. FROM AN INVESTIGATION OF TANK SKIN EFFECTIVENESS UNDER INTERNAL PRESSURIZATION, IT WAS FOUND THAT TANK SKINS WILL BE FULLY EFFECTIVE. (REF. NACA ARR #45810).

THE REDUCTION OF THIS EFFECTIVE AREA TO EQUIVALENT LUMPED AREAS WILL BE DEALT WITH IN THE SECTION ON IDEALISING THE STRUCTURE (TANK SIDES ARE THERE TREATED AS BEAMS).



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 76510/3

SHEET NO. 8-7

AIRCRAFT

C-105

PREPARED BY

DATE

E. AUGUSTINE

12/MAY/55

CHECKED BY

DATE

 β

JUNE 55

MEMBER 23. - CONT'D.BASIC AREA

.051 BUTT STRAP

= .214

INTERCOSTAL WEB

(OUTSIDE RANGE
OF .128 STRAP)

(FANGLES NEGLECTED)

.040 BUTT STRAP

ARM. BAY SKIN (.95 IN.)

15% WIDTH ARM. BAY SKIN (COMP) = .60 IN.

WIDTH ARM. BAY SKIN (TENSION) = 3.5 IN.

AREA SKIN (COMPRESSION)

= .0144

AREA SKIN (TENSION)

= .084

EFFECTIVE AREA (OUTSIDE .128 STRAP)= .263 IN²EFFECTIVE AREA (INCLUDING .128 STRAP)= .430 IN²



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 71650/9

SHEET NO.

8-8

AIRCRAFT:

C-105

PREPARED BY

DATE

E. AUGUSTINE

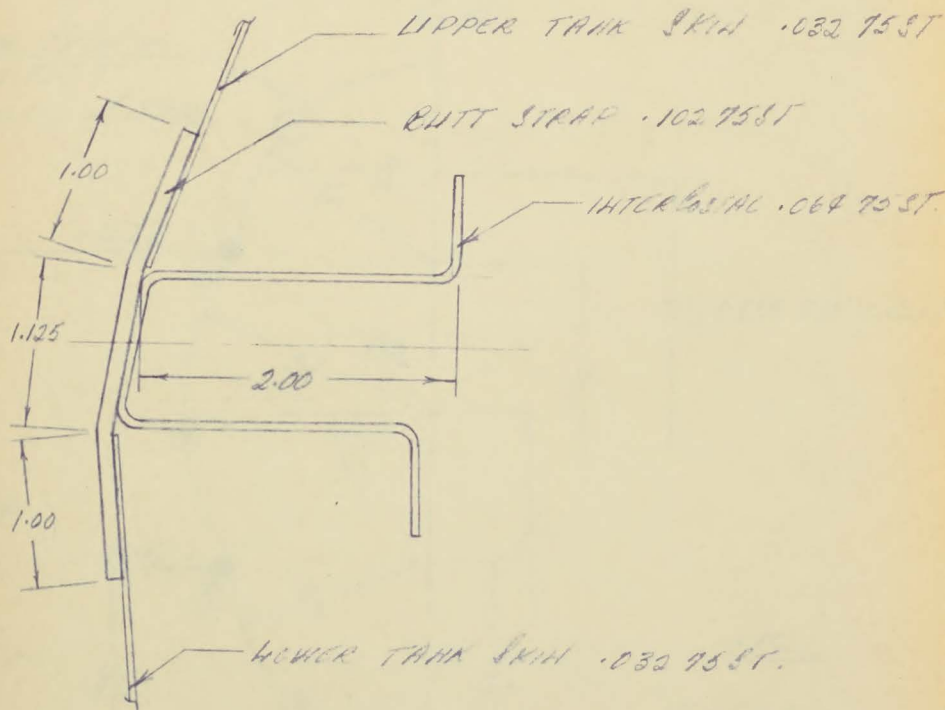
10/MAY/55

CHECKED BY

DATE

B

JUNE '55

TANK MIDDLE LONGERON. -

TANK SKINS NOT INCLUDED FOR SAME REASON AS FOR MEMBER 23.

RUTT STRAP (.102) CONTINUOUS OVER TANK LENGTH.

INTERNAL WEBS (2 AT 2.00 IN DIA) ARE EFFECTIVELY CONTINUOUS OVER TANK LENGTH. THE FLANGES ARE NOT & ARE NEGLECTED IN AREA CALCULATIONS.

$$\text{EFFECTIVE AREA} = .575 \text{ in}^2$$



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT

C-105

REPORT NO. 7/25/55

SHEET NO. 8-9

PREPARED BY

DATE

E. AUGUSTINE

JUNE 1955

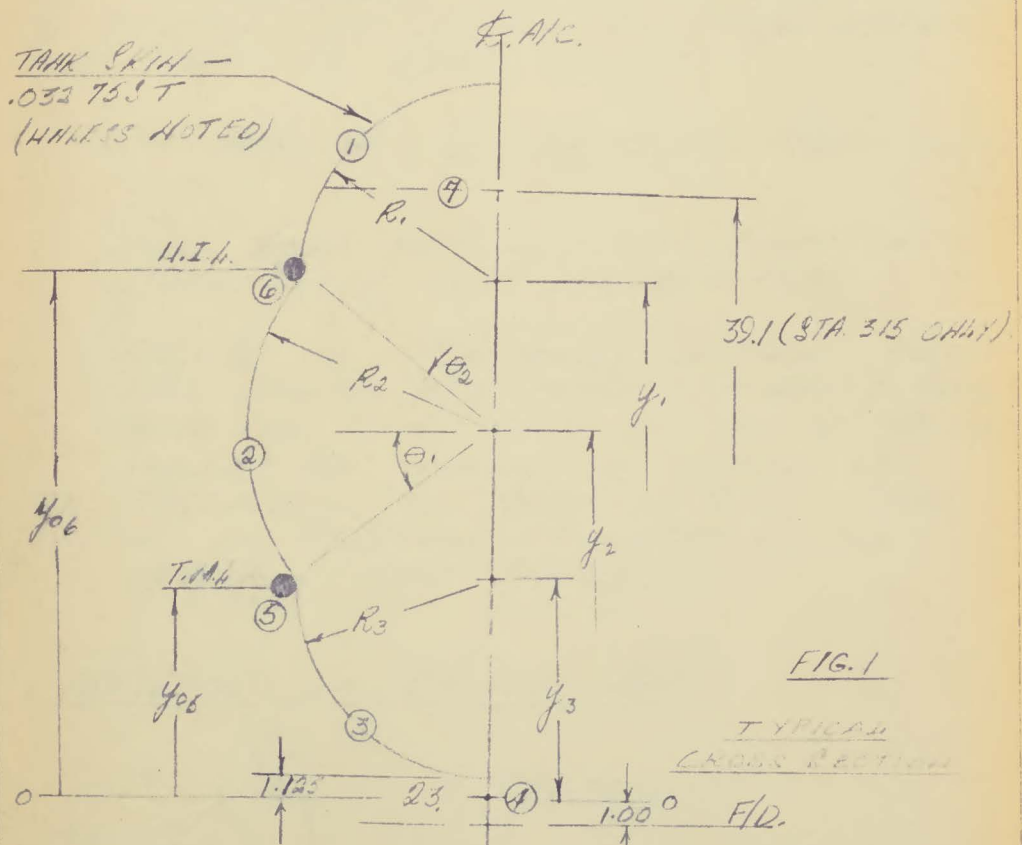
CHECKED BY

DATE

B.

JUNE '55

MOMENT OF INERTIA OF FUEL TANK VIBRA.



$$I = \sum_{i=1}^7 (y_o^2 \cdot \Delta A + \bar{I}_{o_i}) - \bar{y}^2 \cdot A$$

WHERE THE SUMMATION IS TAKEN OVER THE 7 ELEMENTS DESIGNATED AS (1) - (7).

y_o - DIST. FROM REF. AXIS TO CENTROID OF ELEMENT

$\bar{I}_{o_i}$ - MOM. OF INERTIA, ABOUT C.G. OF EACH ELEMENT.

$\bar{y}$ - DIST. FROM REF. AXIS TO TANK C.G.



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/6510/9

SHEET NO. 8-10

AIRCRAFT

C-105

PREPARED BY

DATE

E. AUGUSTINE

20/11/55

CHECKED BY

DATE

B

JUNE '55

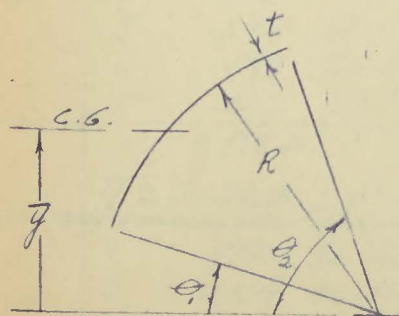
MOMENTS OF INERTIA ARE CALCULATED AT 3 STATIONS. 315, 392, & 478. ELEMENT ⑦ APPEARS AT STATION 315 ONLY WHERE THE TANK IS FLAT ON TOP.

FOR AREAS ④ & ⑤, REFERENCE SHEETS 7E'8.

TANK SKINS ARE .03275 ST EXCEPT AT STATION 315 (REF. DWG. 7/6510/9 FIG.1).

AREA ⑥ IS PRACTICALLY CONSTANT OVER TANK LENGTH (REF. TABLE OF AREAS-DWG. 7/6510/9 FIG.1); AT EACH OF THE 3 STATIONS WHERE THE MOMENT OF INERTIA IS CALCULATED, THE AREA OF ⑥ IS TAKEN AS THE AVERAGE OF AREAS IN THE REGION OF THAT STATION.

PROPERTIES OF CIRCULAR ARCS.



$$A = Rt(\theta_2 - \theta_1)$$

$$\bar{y} = -\frac{R}{(\theta_2 - \theta_1)} (\cos \theta_2 - \cos \theta_1)$$

$$I_o = (\frac{R^2}{2} - \bar{y}^2)A - \frac{R^3}{2} (\sin \theta_2 \cos \theta_2 - \sin \theta_1 \cos \theta_1)$$

I_o - MOMENT OF INERTIA ABOUT C.G.



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/9

SHEET NO. 8-11

AIRCRAFT:

C-105

PREPARED BY

DATE

E. AUGUSTINE

20 MAR 55

CHECKED BY

DATE

B

June '55

CALCULATION OF $\bar{y}$ & $\bar{y}^2$ OF THE CIRCULAR ELEMENTS
(SEE TABLE SHEET 8-13).

STA.	R ₁	y ₁	θ_1	θ_2	R ₂	y ₂	θ_1	θ_2	R ₃	y ₃	θ_1	θ_2	y ₃	y ₃ ²
315	15.03	34.6	0	175	18.25	24.65	45.5	34	13.25	14.50	90	14	12.0	35.0
392	11.52	35.22	0	90	16.0	24.55	35	41.5	13.25	14.50	90	0	16.0	36.0
472	7.95	35.6	0	00	13.56	24.65	24.5	55	13.25	14.50	90	20	14.5	36.0

CALCULATION OF MOMENT OF INERTIA OF TANK.

(REF. DWG. 7/0510/9 FIG. 1 & SHEET 13).

STA.	ELEM.	ΔA	y ₀	y ₀ ΔA	y ₀ ²	y ₀ ² ΔA	I ₀
315	1	.1836	36.8773		1360.0090		.3093
	2	.8103	22.9251		527.9128		38.1452
	3	.5625	4.8076		29.1130		5.2771
	4	.263	0		0		0
	5	.575	12.0		144.0		0
	6	.549	35.0		1225.0		0
	7	.560	37.1		1528.8100		0
		8.5034		76.0948		2301.5061	44.5316

$$\bar{y} = 21.7203; \bar{y}^2 A = 1652.8040; I = 693.2337$$

392	1	.5790	42.5538		1810.8360		7.2784
	2	.6836	25.3912		644.7130		23.6462
	3	.6660	6.0648		36.7818		11.0747
	4	.263	0		0		0
	5	.575	16.0		256.0		0
	6	.568	36.0		1296.0		0
		3.3316		75.6732		2397.0187	42.0493

$$\bar{y} = 22.6963; \bar{y}^2 A = 1717.7259; I = 721.3421$$



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/9

SHEET NO. 8-12

AIRCRAFT:

C-105

PREPARED BY

DATE

L. AUGUSTINE

29/09/55

CHECKED BY

DATE

B

JUNE '55

TANK MOMENT OF INERTIA (CONT'D.)

SIP.	F.A.C.A.	ΔA	y_0	y_0^2	$y_0^2 \Delta A$	I_0
	1	.3987	41.6479	1652.2129	2.3747	
	2	.6021	27.1373	710.5339	15.5477	
478	3	.8190	8.6147	64.2359	5.5255	
	4	.430	0	0	0	
	5	.575	17.5	310.2500	0	
	6	.575	26.0	1296.0000	0	
		3.3948	71.4692	2144.2417	42.6780	

$$\bar{y} = 21.0511; \bar{y}^2 A = 1504.4016; I = 623.1181.$$

TABLE FOR CALCULATING CONTRIBUTION TO LOSS OF TENSILE
OF TANK FROM CIRCULAR HOLE(S)
RADIUS (R) & AREAS (A) TAKEN FROM ENDS OF 3
PRESSURE BULKHEADS.

7-0154-0008 SUP 2 14.2, 7-0154-0008 SUP 3 14.5, SUP 4 14.7

		1	2	3	4	5	6	7	8	9	10
RTA.	$\frac{1}{2}$ $\frac{3}{4}$ $\frac{1}{4}$	T	RT	(R-6) (RADIUS) @ x 3	A	R (R-6) = 4/D	LOSS	LOSS	Q-Q	F	R ² 2
		.040	.6012	.30594	.1836	.49 .3077	.1537	.1	.0463	.2	.107 .752
315		.032	.5840	.38754	.8103	.13 .1528	.8290	.7000	.1231	T .6849	.16 .692
		.032	.4240	.32645	.5625	.9 .9891	.5703	0	.9703	.9	.6 .148
		.032	.3686	.57080	.5790	.4 .3338	0	.1	.0	.7 .3338	.12 .570
392		.032	.5120	.32518	.6836	.11 .9834	.7410	.8892	.0722	.8412	.127 .292
		.032	.4240	.57080	.6660	.8 .4352	.1	0	.1	.8 .4352	.16 .622
		.032	.3538	.57080	.3987	.5 .0384	0	.1	.0	.5 .0489	.5 .932
478		.032	.4339	.38754	.6021	.9 .7127	.5736	.9100	.3364	.3 .2875	.81 .12
		.032	.4240	.91986	.8140	.6 .9015	.9377	0	.9377	.6 .4853	.45 .70

MALTON, ONTARIO
TECHNICAL DEPT. (AIRFRAME)

REPORT NO.

SHEET

DATE _____

PREPARED BY -

SHATING (CONTINUING TO WEST OF TOWN)
OF TANK FROM CINCINNATI BRICKS.
LOS (6) TANK FROM END OF 3
KNOWLEDGE.

0154-0008 SUT3 125, SUT4 127,

AIRCRAFT

WEIGHT

C. G. POSITION

[illegible]

TECHNICAL DEPT. (AIRFRAME)

C. G. POSITION

PREPARED BY -

8-12

June 25

E. A.

[illegible]

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT:

C-105

REPORT NO 7/05/0/5

SHEET NO.

8-14

PREPARED BY

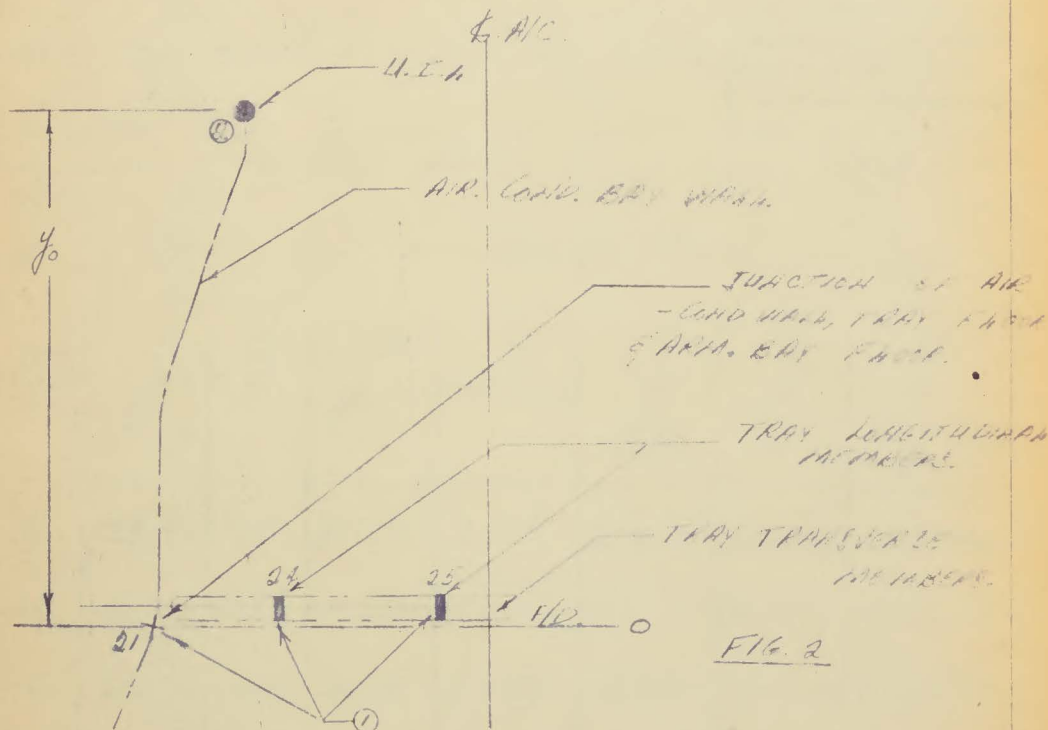
0 A 7

CHECKED BY

CASE

JUNE '55

MOMENT OF INERTIA OF AIR-COND. RAY



Typical cases from

AIR CONDITIONING BY HAND IS NEGLECTED IN THESE CALCULATIONS.

AREAS 24, 25, TRAY LONGITUDINAL MEMBERS, ARE ASSUMED TO ADD TO AREA 21, ON FUSELAGE DATUM, TO FORM THE BOTTOM CAP IN THIS REGION.

$$A_1 = .243 + 2 \times .267 = .777 \text{ in.}^2 \text{ (REF. SHEETS 8-5)}$$



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/6510/9

SHEET NO. 8-10

AIRCRAFT:

C-105

PREPARED BY

DATE

E. A. HARTING

25/JUN/55

CHECKED BY

DATE

P

JUNE '55

MOMENT OF INERTIA OF AIR-RAILWAY WAGON UNIT.

REF. DWG. 7/6510/9 FIG. 1 FOR A_2 (U.I.A.) A_2 TAKEN AS
 AVERAGE OF AREAS AT
 STAS. 24 & 25. STANDING STA.
 WHERE I CALCULATED

MOMENT OF INERTIA OF TRAY LONGITUDINAL
 MEMBERS (24, 25) ABOUT THEIR CENT AXIS IS
 NEGLECTED.

y_0 - SCALD FROM DWG. 7/6510/9 FIG. 1.

STA.	① y_0	② A_2	③ $y_0 A_2$	④ $(A_1 + A_2)$	⑤ $\bar{y}$ ③/④	⑥ $\bar{y}^2$ ⑤ ²	⑦ I ① x ③ - ⑥ x ④
255	33.5	1.310	45.56	2.157	21.390	454.5033	554.9394
268	34.0	.896	30.464	1.673	18.292	331.5750	481.0510
292	35.0	.605	21.175	1.382	15.8220	224.3237	416.4816
315	36.0	.549	19.764	1.326	14.9550	222.1590	416.9212



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/19

SHEET NO. 8-17

AIRCRAFT:

C-105

PREPARED BY

DATE

E. AUGUSTINE

25/MAY/55

CHECKED BY

DATE

E

JUNE '55

STA. 292. AREAS - AVERAGE OF STAS 268, 292, 315

AREA.	ΔA	y_0	$y_0 \Delta A$	y_0^2	$y_0^2 \Delta A$
	1.193				
	.1443				
	.1473				
	.1056				
	.0817				
	.0818				
	.0917				
	.1437				
	.140				
	.263	44.5		1980.2500	

2.4001

31.8374

1095.2589

$$\bar{y} = 13.2650; \bar{y}^2 A = 422.3221; I = 672.9268$$

STA 337 AREAS AVERAGE OF STAS 315, 337, 359.

AREA.	ΔA	y_0	$y_0 \Delta A$	y_0^2	$y_0^2 \Delta A$
	1.6757				
	.1373				
	.142				
	.148				
	.140				
	.1627				
	.1537				
	.1517				
	.148				
	.5753	45.0		2025.0	

3.4384

51.8056

1883.3934

$$\bar{y} = 15.0668; \bar{y}^2 A = 780.8460; I = 1102.8474$$



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/65019

SHEET NO. 8-20

AIRCRAFT

C-105

PREPARED BY

DATE

E. AUGUSTINE

19/04/55

CHECKED BY

DATE

B

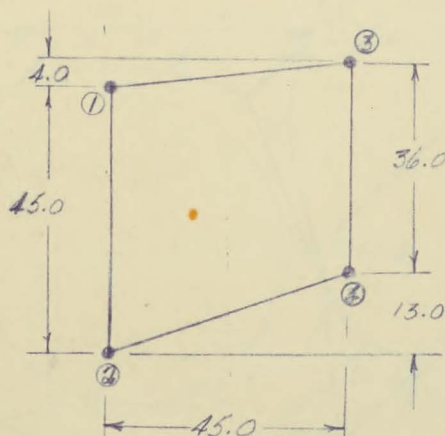
JUNE 55

2.

IDEALIZED STRUCTURE

THIS SECTION IS CONCERNED WITH REDUCING THE FORGING LONGITUDINAL MEMBER AREAS OF THE ACTUAL STRUCTURE TO LUMPED AREAS OF THE IDEALIZED STRUCTURE.

THE LONGITUDINAL MEMBER AREAS IN THE IDEALIZED STRUCTURE ARE CONCENTRATED IN FOUR LOCATIONS ON THE CROSS SECTION (FIG. 1). THIS CROSS SECTION REMAINS CONSTANT THROUGHOUT THE STRUCTURE LENGTH.

FIG. A.

①-② SIMULATES
FUSELAGE SIDE.



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/9

SHEET NO. 8-21

AIRCRAFT:

C-105

PREPARED BY

DATE

E. AUGUSTINE

20 MAR 55

CHECKED BY

DATE

B

JUNE 55

THE DIRECT STRESS DISTRIBUTION ON THE OUTER SHELL LONGITUDINAL MEMBERS IS ANTICIPATED TO BE OF THE FORM INDICATED IN FIG. 2.

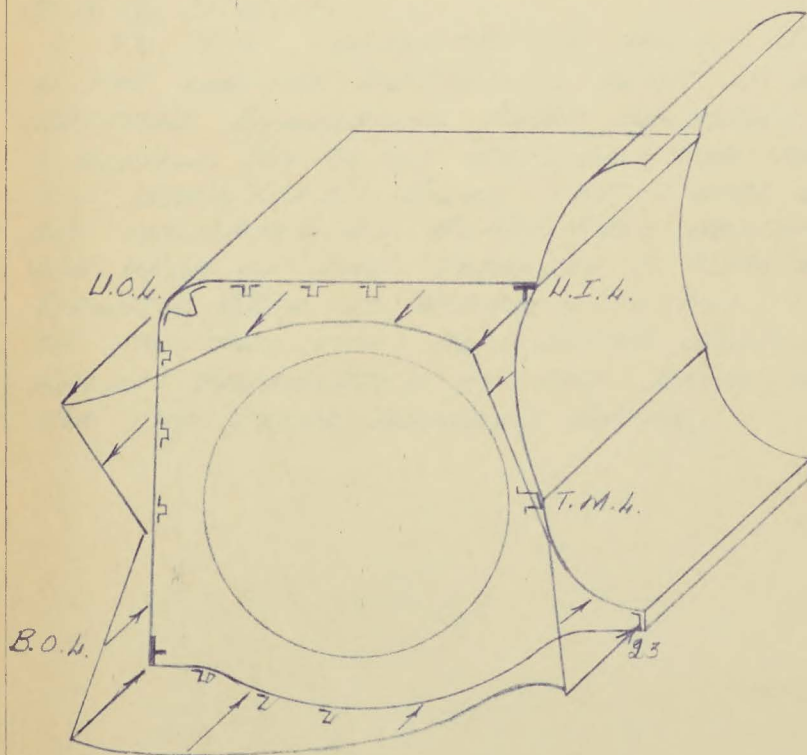


FIG. 5.

CONSEQUENTLY A RATIONAL PROCEDURE IS INDICATED FOR THE SUMMING OF AREAS. CONSIDERING THE OUTER FINISH SIDE, FIG. 2, A STRESS DISTRIBUTION OF A TYPE ARISING FROM BENDING IS SHOWN. THEREFORE THE STRUCTURE FORMED BY THE U.O.L., B.O.L. & MATERIAL BETWEEN IS CONSIDERED AS FORMING A BEAM WHICH IS REPLACED BY A BEAM OF EQUIVALENT MOMENT



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/2510/9

SHEET NO. E-22

AIRCRAFT

C-105

PREPARED BY

DATE

H. AUGUSTINE 20 MAR 55

CHECKED BY

DATE

J. JUNE 55

OF INERTIA BUT HAVING CONCENTRATED CAPS IN THE LOCATIONS ① & ②. AS SHOWN IN FIG. 1. THE TANK WING*, HAVING THE SAME DISTRIBUTION IS ASSUMED IN A SIMILAR MANNER. THE CALCULATED CAPS BEING NOW PAIRED AT LOCATIONS ③ & ④ (FIG. 1).

TO THE CONCENTRATED AREAS CALCULATED IN THE MANNER DESCRIBED ABOVE IS NOW ADDED MATERIAL SIMULATING THAT CONTAINED ON TOP & BOTTOM SURFACES BETWEEN THE TWO WINGS (I.E. STRINGER). SINCE THE STRESS DISTRIBUTIONS ARE INDICATED AS REASONABLY UNIFORM (FIG. 2), ONE HALF OF THE TOTAL TOP & TOTAL BOTTOM STRINGER AREA IS ASSUMED TO ADD TO EACH OF THE TWO TOP & TWO BOTTOM CONCENTRATED AREAS CALCULATED BY THE BEAM THEORY, FOR THE TOTAL ASSUMED AREAS.

* & AIR CONDITIONING RAY WING.



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/25019

SHEET NO. 8-23

AIRCRAFT:

C-105

PREPARED BY

DATE

E. MEDLINE

2/11/55

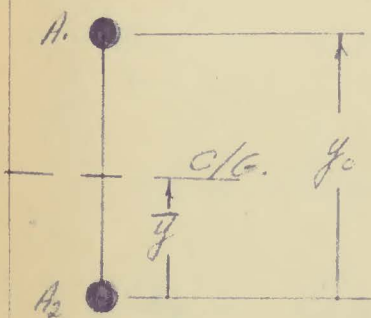
CHECKED BY

DATE

3

JUNE '55

FORMULAE FOR CALCULATING ACTUAL MOMENTS OF INERTIA OF OUTER FUSELAGE SIDE WALL & TANK & AIR-COOL. RAY WING TO EQUIVALENT MOMENT OF INERTIA REPRESENTED BY CONCENTRATED CAPS AT LOCATIONS ①-② & ③-④ RESPECTIVELY ON IDEALIZED STRUCTURE.



A_1, A_2 - Concentrated Cap Areas

y_0 - 36.0 IN. OR 45.0 IN. (REF. FIG. 4 SHEET 20)

$\bar{y}$ - DIST. FROM BOTTOM CAP TO C/G. POSN. THIS DIST. MUST BE THE SAME RELATIONSHIP TO y_0 AS IN THE ACTUAL STRUCTURE.

FOR THE DETERMINATION OF $A_1, A_2, \bar{y}$ THERE ARE 3 CONDITIONS -

$$① \quad y_0^2 A_1 - \bar{y}^2 (A_1 + A_2) = I \text{ (CALCULATED FOR ACTUAL STRUCTURE)}$$

$$② \quad \bar{y} = -k y_0 = \frac{y_0 A_1}{(A_1 + A_2)} \quad (k = \bar{y}/y_0 \text{ RIGID BODY SIMILARITY})$$

REFERENCE CALCULATIONS - ACTUAL STRUCTURE - FOR I & k . (SHEETS 8-11-12, 15-16 & 359).



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/9

SHEET NO. 8-24

AIRCRAFT:

C-105

PREPARED BY

DATE

F. A. HALLS

8/10/55

CHECKED BY

DATE

J

JUNE '55

REDUCE TO 2 EQUATIONS IN A_1', A_2' —

$$y_0^2 A_1' - k^2 y_0^2 (A_1' + A_2') = I$$

$$-k(A_1' + A_2') + A_1' = 0$$

OR —

$$(1 - k^2) A_1' - k^2 A_2' = I/y_0^2$$

$$(1 - k) A_1' - k A_2' = 0$$

8

$$A_1' = \frac{I}{(1 - k) y_0^2} =$$

CATERPILLAR SIDE

$$\frac{I}{9025(1 - k)}$$

TANK SIDE

$$\frac{I}{1296(1 - k)}$$

$$A_2' = \frac{I}{k y_0^2} =$$

$$\frac{I}{9025k}$$

$$\frac{I}{1296k}$$

THE TOTAL DAMPED AREAS, A_1, A_2, A_3, A_4 , FOR THE IDEALIZED STRUCTURE ARE THEN —

$$A_1 = A_1' + \frac{1}{2} A_T ; \quad A_3 = A_2' + \frac{1}{2} A_T$$

$$A_2 = A_2' + \frac{1}{2} A_B ; \quad A_4 = A_2' + \frac{1}{2} A_B$$

WHERE —

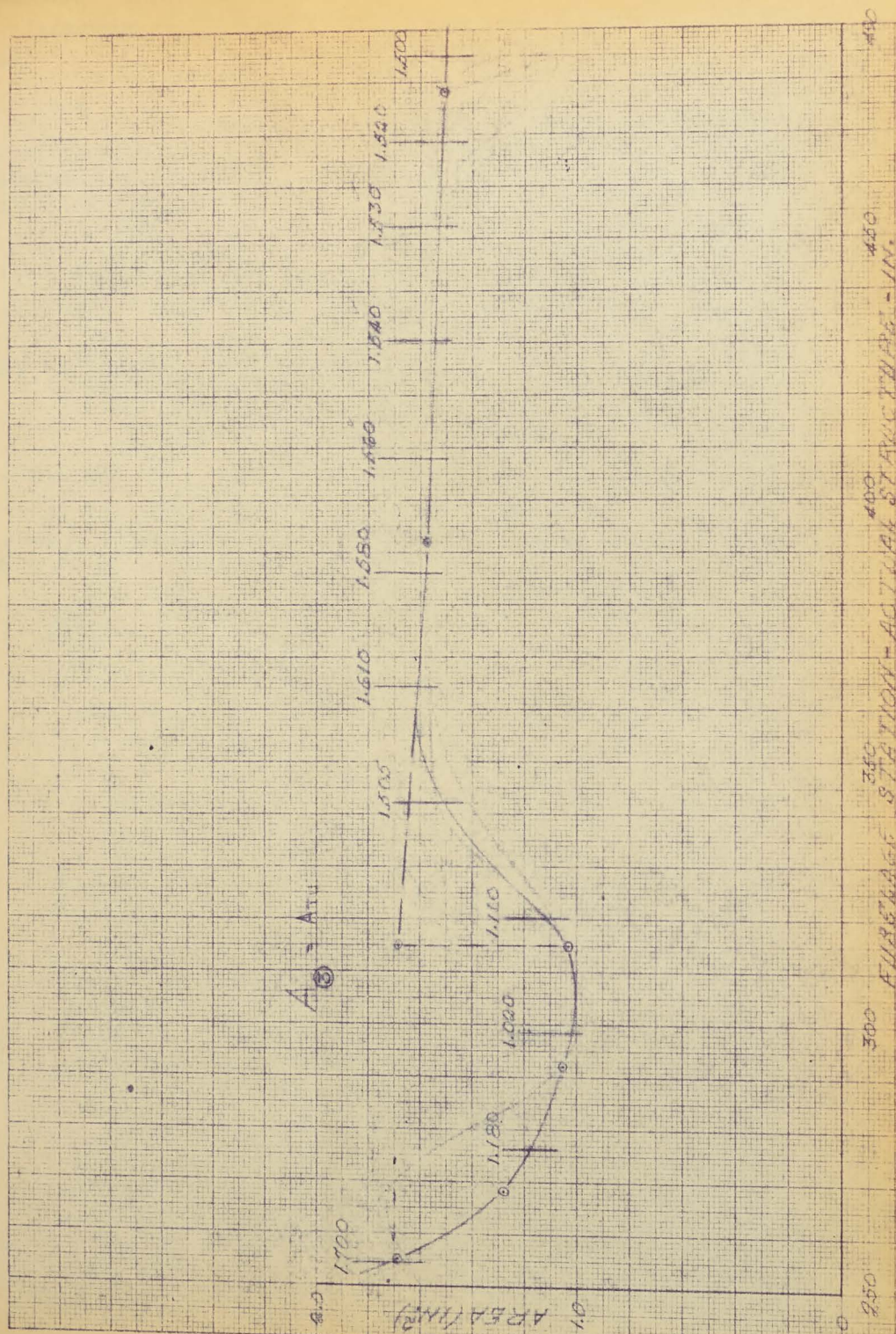
REF. DWG.

7/0510/9

FIG. 1

A_T = SUM OF AREAS OF MEMBERS 2, 7 & 8 ON TOP SPAN.

A_B = SUM OF AREAS OF MEMBERS —
17 → 20 : STA 255 TO STA 315
17 → 22 : STA 315 TO STA 485





AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/9

SHEET NO. 9-0

AIRCRAFT

C105

Engine Section

PREPARED BY

DATE

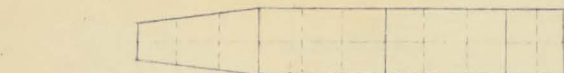
CB

May 30, 1949

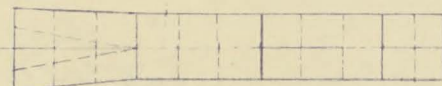
CHECKED BY

DATE

DUCT



Matl. .032 24-7.



755

330

485

STN	LINEAL VARIATION	BEFORE	AFTER	BEFORE	AFTER	BEFORE	AFTER	BEFORE	AFTER
254.500	12.24 8.88	106	107	.007 841	.002 796				
276.500		118	117	.015 462	.005 707				
298.500		120	129	.015 027	.005 799				
320.500	18.000	141	140	.007 376	.002 913				
	0	206	205	.007 914	.002 913				
342.500				.006 036	.001 040				
	0	215	217	.015 828	.005 836				
364.500	18.000	230	229	.015 828	.005 836				
	0			.006 036	.001 040				
386.500	18.000	241	240	.007 914	.002 913				
	0	306	305	.007 914	.002 913				
408.500	18.000	318	317	.015 828	.005 836				
	0			.006 036	.001 040				
430.500	18.000	330	329	.015 828	.005 836				
	0			.006 036	.001 040				
452.500	18.000	341	340	.007 914	.002 913				
	0	406	405		.002 913				
468.750	18.000				.001 796				
	0								
485.000	18.000								
	0								

Ref Page

9-16 9-49 9-43 9-20 9-19 9-49 9-46



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/9

SHEET NO. 9-11

AIRCRAFT:

C105

CENTRE Fuselage

PREPARED BY

DATE

C.B.

May 20, 21

CHECKED BY

DATE

H. Gardner

June 5, 21

DUCT BENDING - MIDWAY

$$\text{AXIAL ENERGY} = \frac{.032 \times 22,000 \times 4 \times 5,242.325093}{3 \times 19,56 \times 19,56^2} \left[\sigma_{b1}^2 + \sigma_{b1}\sigma_{b2} + \sigma_{b2}^2 \right]$$

$$= .00121337 \left[\sigma_{b1}^2 + \sigma_{b1}\sigma_{b2} + \sigma_{b2}^2 \right]$$

$$\text{SHEAR ENERGY} = \frac{.032 \times 4 \times 1,658.027937358}{4,000 \times 19,56^2 \times 22,000} \left[\sigma_{b1}^2 - 2\sigma_{b1}\sigma_{b2} + \sigma_{b2}^2 \right]$$

$$= .00641756 \left[\sigma_{b1}^2 - 2\sigma_{b1}\sigma_{b2} + \sigma_{b2}^2 \right]$$

$$\text{TOTAL: } 2V = .00763092 \left[\sigma_{b1}^2 + \sigma_{b2}^2 \right] - .00581088 \times 2 \sigma_{b1}\sigma_{b2}$$

FRAME LOAD:

$$P = \frac{\sigma_b \times .032 \times 4 \times 5,242.325058}{19,56 \times 22,000}$$

$$= 1.55934651 \sigma_b$$

TOTAL - Down on Duct
For End of Air Frame

$$3.11869302 \sigma_b$$

TOTAL - Up on Duct
For Centre Frame

TECHNICAL DEPT. (AIRFRAME)

C. G. POSITION

PREPARED BY -

	7	8	9	10	11	12	13	14	15	16	17	18	19
(S)	+	=	view	6	7	8						To	T
000000	0	104 976	1	- 324 000 000	0	104 976 000 000							7.212
7624 21 703 877		97 743 976 622	5	- 1 511 481 120	108 519 385	493 719 882 110							
993772 21 000 000		78 731 920 580	1	- 242 999 773	81 000 000	72 731 926 580						.540	7.09
000000 162 000 100		52 48 512 48.	6	- 972 000 000	972 000 000	514 905 194 910							
000000 242 999 713		26 244 000 000	1	- 81 000 000	242 999 773	26 244 000 000						.342	7.409
9 877 202 296 224		70;2 026 153	5	- 108 519 385	1 511 481 120	20 160 290 775							
324 000 000	0	0	1	0	324 000 000	0						1 000	0
3 7(16 85)		28.274 337 20		- 3 240 000 278	3 240 000 278	1 043 760 295 377							
				- 45 800 442 987	4 580 442 987	1 494 069 803 898						(S _i +v)L = 396	



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/9

SHEET NO. 9-15-1

AIRCRAFT

C105

CENTRE FUSELAGE

PREPARED BY

DATE

C.B.

May 24, 55

CHECKED BY

DATE

J. Gardner

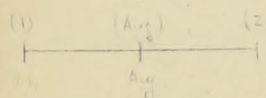
June 55

DUCT

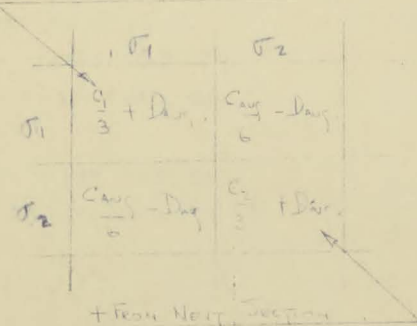
STATION	$\sigma_1 + \sigma_2$	$2\epsilon_{avg}$	CENTRE FUSELAGE LOAD				
250	.007 261 68	-.005 519 47	3.225 421				
			* 3.159 477				
MIDWAY	.007 630 92	-.005 810 04	3.118 693				
			* 3.072 035				
320 Aft @ 22"	.007 914 36	-.006 036 51	2.961 094				
320 Aft @ 62"	.007 944 51	-.005 537 52					

However, the Duct is NOT UNIFORM ALONG ITS LENGTH.

$$\text{Average strain energy} = 2V = \frac{C}{3} \sigma_1^2 + \frac{C_{avg}}{3} \sigma_1 \sigma_2 + \frac{C_2}{3} \sigma_2^2 + D_{avg} (\sigma_1 - \sigma_2)^2$$



+ From Next Section



where $\frac{C}{3}$ is the coefficient of
strain energy

& D is the coefficient of
shear energy.

* COMPUTE INTERMEDIATE VALUES by $y = ax^2 + bx + c = 0$.



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO 7/0510/9

SHEET NO 3 17

AIRCRAFT:

C10-5

CENTRE FUSelage

PREPARED BY

CE.

DATE

May 17-33

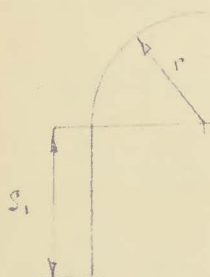
CHECKED BY

V. Hardaway

DATE

June 25

DUCT - TORSION



$$\frac{\partial \psi}{\partial x} + \frac{\partial \tilde{\psi}}{\partial s} = 0$$

$$\frac{\partial \tilde{\psi}}{\partial x} + \frac{\partial \psi}{\partial s} = 0 \quad \frac{\partial \tilde{\psi}}{\partial x} = 0$$

$$\psi = 0$$

$$\tilde{\psi} = f(s)$$

$$\frac{\partial \psi}{\partial x} = -f'(s); \quad \psi_x = -f'(s)x + g(s)$$

BOUNDARY CONDITIONS: $x=0 \quad \psi_x=0 \quad \therefore g(s)=0$
 $x=l \quad \psi_x=0 \quad \therefore f'(s)=0$

$$\tilde{\psi} = \text{constant}$$

DUCT ENERGY - STATION 253.14

$$2V = \frac{1}{G} \int \int \tau_{xz}^2 dx ds = \frac{t}{G} \int_0^{2\pi} \int_0^l \tau_{xz}^2 dx ds = \frac{t}{G} \tau_{xz}^2 S \cdot l$$

$$= \frac{.032 \times 28,106.55 \times 22,000 \times 4 \tau_{xz}^2}{4,000}$$

$$= .01978700 \tau_{xz}^2$$



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/9

SHEET NO. 9-19

AIRCRAFT:

C105

CENTRE FUSELAGE

PREPARED BY

DATE

C.B.

May 19 '55

CHECKED BY

DATE

J. H. Anderson

June 55

Duct - Torsion - Station Midway Btwn Stn 255 & Spout or Circular Duct

 S_1 & r VARYING LINEARLY

$$S_1 = \frac{8,880.00 + 0.00}{2} = 4,440.00 \text{ in}^2$$

$$r = \frac{12,240.00 + 18,000.00}{2} = 15,120.00 \text{ in}^2$$

$$S = 4,440.00 + \frac{3,141,593 \times 15,120.00}{2} = 28,190,443$$

$$2V = \frac{.032 \times 28,190,443 \times 4 \times 22,000 \text{ in}^2}{4,000} = \frac{.019846 \text{ in}^2}{1}$$

RELATIVE τ FOR CONSTANT TORQUE $\tau_{255} \times A_{255} = \tau_{220} \times A_{220}$

$$A_{255} = 4S_1r + \pi r^2 = 4 \times 8,880 \times 12,240 + 3,141,593 \times 12,240^2$$

$$= 434,764,800 + 470,665,923 = 905,430,723 \text{ in}^2$$

$$A_{mid} = 4 \times 4,440 \times 15,120 + 3,141,593 \times 15,120^2$$

$$= 268,551,200 + 718,213,899 = 986,765,099 \text{ in}^2$$

$$A_{220} = 3,141,593 \times 18,000^2 = 101,376,132 \text{ in}^2$$

	AF	2AC	Square
$\times .032$	$= 28,973,793$	$57,947,566$	$839,490,101$
	$31,575,827$	$60,089,114$	$977,032,851$
	$32,572,036$	$63,181,687$	$1,060,937,529$
		$64,876,792$	
		$68,144,072$	



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO 7/0510/9

SHEET NO 9-20

AIRCRAFT

C 105

Centre Fuselage

PREPARED BY

DATE

W. G. G. R. R.

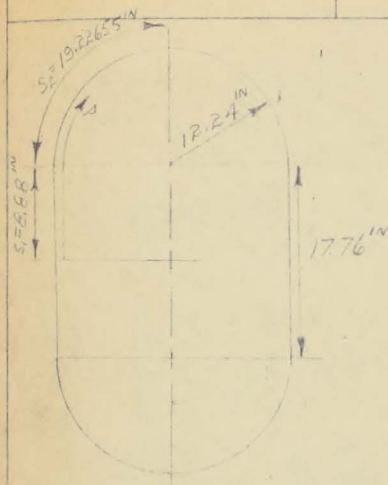
MAY 17, 1955

CHECKED BY

DATE

J. H. G. R. R.

JUNE 5



WARPING IN Duct

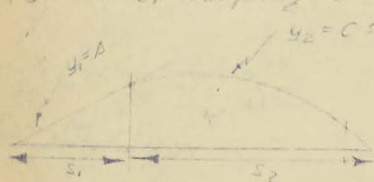
ST 2 255 Ref. S. Young

$$S_1 = 9.88000$$

$$S_2 = \frac{12.24 \pi}{2} = 19.2265$$

$$S = S_1 + S_2 = 28.10655$$

Pattern of Warping Stresses



$$y_2 = C \sin \frac{K\pi}{S} (S-s)$$

$$\text{at } s = S_1 \quad y_1 = y_2 \quad y_1' = y_2'$$

$$\therefore S_1 = C \sin \frac{K\pi}{S} S_2$$

$$\text{and} \quad 1 = -C \frac{K\pi}{S} \cos \frac{K\pi}{S} S_2$$

$$\therefore C = \frac{S_1}{\sin \frac{K\pi}{S} S_2}$$

$$1 = \frac{-S_1 \cos \frac{K\pi}{S} S_2}{\sin \frac{K\pi}{S} S_2} \cdot \frac{K\pi}{S}$$

$$\text{or } \tan \frac{K\pi}{S} S_2 = -\frac{K\pi S_1}{S}$$

Thus K may be uniquely determined graphically see sheet

$$K = 106000$$

$$\text{Now } \sigma = \frac{E_w}{y_{max}} \frac{x}{l} [y_1, y_2] \quad (\text{ie } y_1 \text{ or } y_2)$$

$$= \frac{E_w}{y_{max}} \frac{x}{l} \left[\Delta, S_1 \frac{\sin \frac{K\pi}{S} (S-s)}{\sin \frac{K\pi}{S} S_2} \right]$$



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/6510/3

SHEET NO. 9-21

AIRCRAFT

C 105

Centre Fuselage

PREPARED BY

DATE

W. C. P. V. R.

May 13, 1955

CHECKED BY

DATE

J. F. Gardner

June 24

and

$$\frac{\partial \sigma}{\partial x} + \frac{\partial T}{\partial s} = 0$$

$$\frac{\partial T}{\partial x} = 0$$

$$\text{Therefore } T = -\frac{\sigma_w}{y_{max} L} \left[\frac{s^2}{2} + C_1 + \frac{s_1 s}{K\pi} \left(\frac{\cos \frac{K\pi}{s} (s-s_1)}{\sin \frac{K\pi}{s} s_2} + C_2 \right) \right]$$

$$\text{at } s = s_1$$

$$\frac{s_1 s}{K\pi} \left(\cot \frac{K\pi}{s} s_2 + C_2 \right) = 0$$

$$\therefore T = -\frac{\sigma_w}{y_{max} L} \left[C_1 + \frac{s^2}{2} + \frac{s_1 s}{K\pi} \left\{ \frac{\cos \frac{K\pi}{s} (s-s_1)}{\sin \frac{K\pi}{s} s_2} + \frac{s}{K\pi s_1} \right\} \right]$$

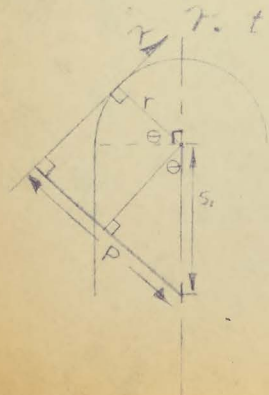
Up to s_1 the torque may be evaluated by simple integration. $-\frac{\sigma_w}{y_{max} L}$ has been neglected throughout & will be reinserted later

$\therefore$ first find

$$r.t. \left[\frac{s_1^2}{2} + C_1 s_1 \right] = 45.71082 + 3.4712 C_1$$

where $t = .032$ (skin thickness)

To find the torque distribution about the circular part of the frame we have



T . t. moment arm

P is the moment arm & can be seen to equal

$$r + s_1 \sin \theta = r + s_1 \sin \left(\frac{s-s_1}{r} \right)$$

from above neglecting $-\frac{\sigma_w}{y_{max} L}$

$$T = \left[\frac{s_1^2}{2} + C_1 + \frac{s_1 s}{K\pi} \left(\frac{\cos \frac{K\pi}{s} (s-s_1)}{\sin \frac{K\pi}{s} s_2} + \frac{s}{K\pi s_1} \right) \right]$$



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/510/9

SHEET NO. 9-22

AIRCRAFT

C 105

Centre Fuselage

PREPARED BY

DATE

W. Gierer

June 15/55

CHECKED BY

DATE

B.

June 22/55

Duct Warping:

$$\therefore \text{Torque} = \int \left[\frac{s^2}{2} + C_1 + \frac{s_1 s}{\pi r} \left(\frac{\cos \pi r / s (s - s_1)}{\sin \pi r / s} \right) + \frac{s}{\pi r s_1} \right] t (r + s_1 \sin(\frac{s - s_1}{r})) ds$$

The above expression is far too cumbersome for ordinary integration & hence numerical integration must be used. The "Weddle" & "Six Strip" formulae were used and the answers agree to 8 or 9 significant figures. These computations are shown on pages 9-22 & 9-23

$$\text{Torque} = 1629.33368 + 11.00880 C_1$$

Thus the total torque is:

$$1675.04450 + 14.46692 C_1$$

Equating this to zero we find

$$C_1 = -115.62461$$

The shear and axial stresses may now be calculated & hence the strain energy found.

Shear Stresses (neglecting $-\frac{C_1}{s_{\max}}$) $0 \leq s \leq s_1$

$$\tau = \frac{s^2}{2} + C_1$$

a) $s=0$

$$\tau = -115.62461$$

b) $s = \frac{s_1}{3}$

$$\tau = -111.24381$$

c) $s = \frac{2s_1}{3}$

$$\tau = -98.10141$$

d) $s = s_1$

$$\tau = -76.19741$$

τ for circular portion shown on page 9-24

Computing $\frac{s_1^2}{2} + \frac{s_2^2}{K\pi} \left(\frac{K\pi}{3} (S_{20}) + \frac{S}{K\pi S_1} \right) \sin \frac{K\pi}{S} S_2$ 6 positive

where $S_1 = 88000$ $R = 1.05920$ $S_2 = 19.226547$ $S = 28.1065$

	1	2	3	4	5	6	7	8	9	10	
n	$S_1 + \frac{R}{6} S_2$	$\frac{S_1^2}{2}$	$\frac{K\pi}{S}$	$\frac{1}{3}$	$(4) S_1$	$(4) \frac{S}{S_1}$	$(S-1)$	$(7) \times (6)$	$(3) \times S_2$	$(6) S$	$\sin$
1	12.08772	39.42120	12.67337	0.33333	352000	0.00284	93270	16.02212	1.93770	2.52120	35.732
2	15.28885						1281770	1.54759			0.3204
3	18.42327						2.61327	14.6069			3.32707
4	21.69770						6.40885	7.7380			7.1526
5	24.90212						3.26442	3.8690			9.2612
6	28.10655						0	0			1
7	8.88000	1	1	1	1	1	19.22655	2.32141	1	1	68.075

MALTON, ONTARIO
ENGINEERING DIVISION

7/05/013

9-27

May 16. 1882

C 105

.....

H. Grove

is W on following sheet

↑ referred to as S2 on following sheet.

	1	2	3	4	5	6	7	8	9	10	11	
Station	$\Psi + C_1$	S_2	$(1) \times (2)$	Δ_1	Δ_2	Δ_3	Δ_4	Δ_5	Δ_6	Wedge Coeff's	$(10) \times (3)$	
S_1	39.42720	30.1685	15.44245								15.44245	
	+ C_1		+ 531686	+ 1216033							+ 301685	
$S_1 + \frac{1}{6} S_2$	72.24767	46.523	53.61173	+ 613556	7.12378						5	168.05890
	+ C_1		+ 465230	25.29311	- 605620	3.54191					+ 2326156	
$S_1 + \frac{1}{3} S_2$	110.35833	53.376	58.20489	+ 66530	3.58187						1	58.90489
	+ C_1		+ 533766	28.87409	- 600660	5.46325	+ 600656	+ 1.50845			+ 533766	
$S_1 + \frac{1}{2} S_2$	148.12419	59.261	87.77987	+ 658856	1.88138	- 604016	+ 41299	+ 60628	+ 27113		6	586.67922
	+ C_1		+ 592610	26.85360	- 613690	5.87614	+ 600936	+ 171958	+ 600070		3.555660	
$S_1 + \frac{2}{3} S_2$	179.96659	63.777	114.77347	+ 645166	7.75752	- 607026	+ 1.36669	+ 60021			1	114.77347
	+ C_1		+ 637770	12.25102	- 616776	4.50945	+ 601146				+ 637770	
$S_1 + \frac{3}{4} S_2$	201.16721	66.616	134.00233	+ 628306	12.7669	- 601946					5	670.04775
	+ C_1		+ 666160	6.96911	- 618710						+ 3.33096	
S	206.59709	67.594	140.91266	+ 608656							1	140.91266
	+ C_1		+ 675940								+ 675940	
Σ											1694.85341	

4	5	6	7	8	9	10	11	12	13	14	15	16	17
Δ_1	Δ_2	Δ_3	Δ_4	Δ_5	Δ_6	Wedge Coeff's	(10) x (3)		Six Strip Coeff's	(13) x (3)			
						1	15.44745		41	633.14045			
							+			+			
							391684			16.05888 C ₁			Wed
						5	168.05890		216	7260.14415			
							+			+			
							2326154			100.42068 C ₁			
						1	58.90489		27	1590.47203			
							+			+			
							533764			14.41152 C ₁			
						6	586.67922		272	23876.12464			Six
							+			+			
							3535664			161.16992 C ₁			
						1	114.77347		27	3098.88369			
							+						
							637774			17.21979			
						5	670.0775		216	229.46.06220			
							+			+			
							333084			143.89056 C ₁			
						1	140.9786		41	5780.13506			
							+			+			
							675894			27.76244 C ₁			
$\Sigma 1694.88574 + 11.45164$										$\Sigma 71164.91315 + 480.96979 C_1$			

MALTON, ONTARIO
ENGINEERING DIVISION

C. G. POSITION

PREPARED BY

7/0510/9

Q-1

May 17, 1958

W. B. G. V. 100

[illegible]



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 1/0210/9

SHEET No. 9-28

AIRCRAFT

C 105

Centre Fuselage

PREPARED BY

DATE

W. MEYER

12/02/51

CHECKED BY

DATE

A

12/02/51

Duct Warplings:Strain Energy due to Shears:The shear stress at any point is $\left(\frac{\sigma_{w1}}{y_{maxL}} - \frac{\sigma_{w2}}{y_{maxL}}\right) K_{T5}$ where K_{T5} is the coefficient previously determined.The strain energy $V = \iint_G \frac{\tau^2}{G} dA dx$

$$\therefore 2V = \frac{(\sigma_{w1} - \sigma_{w2})^2}{LG y_{max}^2} + \int K_{T5}^2 ds$$

Thus the problem is reduced to the evaluation of the above integral & this can be done using the "Weddle" & "3/8" Rules.

Station	K_{T5}	K_{T5}^2	"3/8" Coeff's	Product
3	-115.62461	13369.05044	1	13369.05044
S ₁ /3	-111.24381	12375.18526	3	37125.55578
2S ₁ /3	-98.10141	9623.88664	3	28871.65992
S ₁	-76.19741	5806.04522	1	5806.04522

$$\Sigma = 85172.31143$$

$$I = 1/11 (85172.31143) = 94541.26569$$

Station	K_{T5}	K_{T5}^2	Weddle Coeff's	Product
S ₁	-76.19741	5806.04522	1	5806.04522
S ₁ + 1/6 S ₂	-73.37694	5384.05022	5	9407.79460
S ₁ + 1/3 S ₂	-5.26622	27.73307	1	27.73307
S ₁ + 1/2 S ₂	32.49957	1056.22205	6	6337.33230
S ₁ + 2/3 S ₂	64.37592	4133.11832	1	4133.11832
S ₁ + 5/6 S ₂	85.54260	7317.53641	5	36587.68205
S ₂	92.97308	8643.09360	1	8643.09360

$$\Sigma = 70949.69923$$

$$I = 1/6612275 (70949.69923) = 68205.89600$$

$$\therefore \text{Total } I = 162,747.16268$$

$$\therefore 2V = \frac{0.32 \times 162747.16268}{22 \times 4,000 \times (12.1436)^2} [\sigma_{w1} - \sigma_{w2}]^2 = 0.000401351 [\sigma_{w1} - \sigma_{w2}]^2$$



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/3

SHEET NO. 9-29

AIRCRAFT

C105

Centre Fuselage

PREPARED BY

DATE

W. Grover

June 16/55

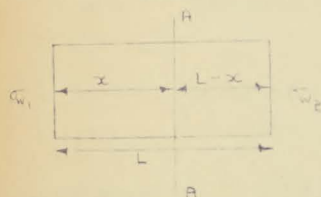
CHECKED BY

DATE

July 55

Duct Warping:

Strain Energy due to Axial Stresses



The formula for axial strain energy is

$$2V = \iint \frac{\sigma^2}{E} dA dx$$

$$\text{but } \sigma = \frac{1}{y_{\max}} \left[\sigma_{w1} \frac{x}{L} + \sigma_{w2} \frac{L-x}{L} \right] K \sigma_s$$

$$\therefore 2V = \frac{L}{E y_{\max}^2} \iint (\sigma_{w1} \frac{x}{L} + \sigma_{w2} (1 - \frac{x}{L}))^2 K^2 \sigma_s^2 t dA dx$$

$$= \frac{tL}{3E y_{\max}^2} \int (\sigma_{w1}^2 + \sigma_{w1} \sigma_{w2} + \sigma_{w2}^2) K^2 \sigma_s^2 dA$$

The evaluation of this integral is analogous to the shear integral on the preceding page.

Station	$K \sigma_s$	$K^2 \sigma_s^2$	$\frac{1}{6}$ coeff's	Product
σ	0	0	1	0
$s/3$	2.96000	8.76160	3	26.28480
$2s/3$	5.92000	35.04640	3	105.13920
s_1	8.88000	78.85440	1	78.85440

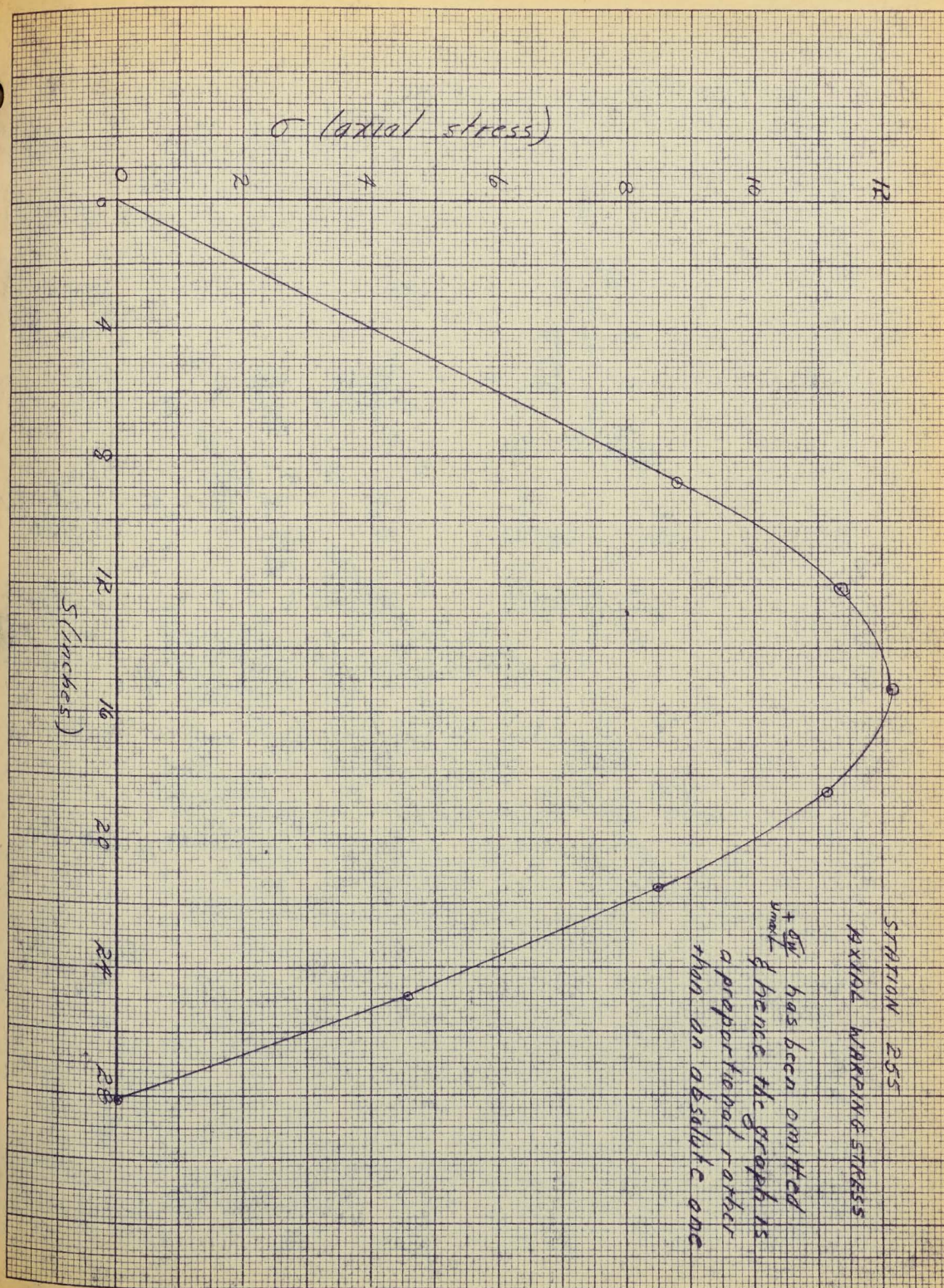
$$\Sigma = 210.27840$$

$$I = 1/11(210.27840) = 233.40902$$

Station	$K \sigma_s$	$K^2 \sigma_s^2$	$\frac{1}{6}$ coeff's	Product
s_1	8.88000	78.85440	1	78.85440
$s_1 + s_2/6$	11.34878	128.79461	5	643.97405
$s_1 + s_2/3$	12.12978	147.37426	1	147.37426
$s_1 + s_2/2$	11.13616	124.01406	6	744.08436
$s_1 + 2s_2/3$	8.48630	72.01729	1	72.01729
$s_1 + 5s_2/6$	4.58182	20.99307	5	104.96535
s	0	0	1	0

$$\Sigma = 1791.26971$$

$$I = 1/11(1791.26971) = 1721.99683$$





AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO

7/0510/0

SHEET NO

7.36

AIRCRAFT

C.105

Centre Fuselage

PREPARED BY

W. Grover

DATE

June 16/55

CHECKED BY

DATE

July '55

Duct Warping: Manual.Calculation of σ (neglecting $\frac{\sigma_w}{Y_{max}} L$) see graph page

a) $\Delta = 0$

$\sigma = 0$

b) $\Delta = S_1$

$\sigma = S_1 = 4.44$

c) $\Delta = S_1 + \frac{1}{2} S_2 = 8.39541$

$\sigma = S_1 \frac{\sin \frac{\pi}{2} (S_2/\Delta)}{\sin \frac{\pi}{2} S_2} = 9.919571 \sin 2.31274$

$\sigma = 7.83348$

d) $\Delta = 12.35661$

$\sigma = 9.919571 \sin 1.785004$
 $= 9.69286$

e) $\Delta = 16.31522$

$\sigma = 9.919571 \sin 1.338753$
 $= 9.65371$

f) $\Delta = 20.27363$

$\sigma = 9.919571 \sin .892502$
 $= 7.72381$

g) $\Delta = 24.23203$

$\sigma = 9.919571 \sin .446251$
 $= 4.28115$

Calculation of Frame Force:

(a) Straight Portion:

$rt(S_1 \frac{3}{4} + C_1 S_1) = 7.058228 + 2.14825 C_1 = 193.29754$

Station	σ	$\cos^2 \theta$	$rt \cos^2 \theta$	T	Y_{max}	$\frac{W_{max}}{Y_{max}}$	Product
S_1	1.000000	1.000000	.48334	27.40187	46.35606	1	40.35466
$S_1 + \frac{1}{2} S_2$	.96593	.97302	.45113	58.70480	26.50111	5	132.50555
$S_1 + \frac{1}{2} S_2$	.86603	.75000	.36228	23.42830	8.50131	1	3.50131
$S_1 + \frac{1}{2} S_2$	.70711	.50000	.24192	15.51036	3.75227	6	22.51362
$S_1 + \frac{2}{3} S_2$	.50000	.25000	.12006	50.42645	6.10684	1	6.10684
$S_1 + \frac{5}{6} S_2$	.25882	.06699	.03241	74.64878	2.41931	5	12.09685
S	0	0	0	83.26569	0	1	0

$\Sigma = 140.64621$

$I = \frac{23.75049}{20} (140.64621) = 167.02047$

$$e^{-\left(s_1 \sin\left(\frac{\theta - s_1}{r}\right) + r\right)t}$$

[illegible]

MALTON, ONTARIO
ENGINEERING DIVISION

SHEET . . . 9-38

DATE . . May 26/55

PREPARED BY - W. Grover

AIRCRAFT . . . C 105

WEIGHT

C. G. POSITION - _____

[illegible]



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/9

SHEET NO. 12-1

AIRCRAFT:

C105

CENTRE FUSELAGE

PREPARED BY

C.B.

DATE

JULY '51

CHECKED BY

E.A.

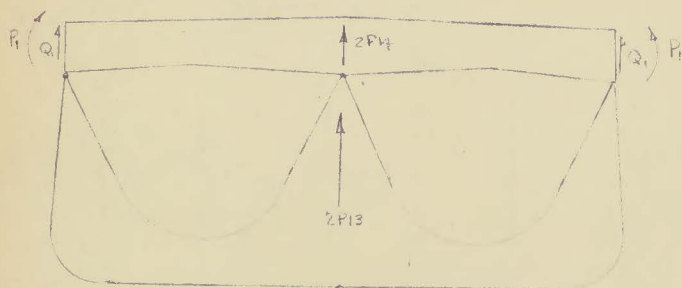
DATE

BULK HEAD 485

FROM RPT 7/0556/27 & 112 REVISED

$$2 \cdot E \cdot U = 109.6956 P_0^2 + 62.5775 Q_1^2 + 1382 P_1^2 + 47.9414 P_0 Q_1 - 3.9826 P_0 P_1 + 2.0016 Q_1 P_1$$

$$P_1 = 107 P_0 + P_3 + P_4$$



REVERSING DIRECTION OF ALL FORCES TO AGREE WITH SIGN CONVENTION OF THE WING & FUSELAGE ANALYSIS & DIVIDING BY $E = 10,000 \text{ KSI}$

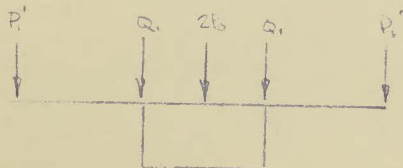
$$2V = P_0 C_{UV} P_V$$

$P_1' \quad Q_1 \quad P_0$

$$2V = \begin{vmatrix} P_1' \\ Q_1 \\ P_0 \end{vmatrix} \begin{vmatrix} 131.617 & .009531 & .018965 \\ .009531 & .005745 & .002283 \\ .018965 & .002283 & .010447 \end{vmatrix}$$

NOTE: P_1' here has a value

$$P_1' = P_1 / 100$$



A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT:

C-105

CENTRE SECTION

REPORT No. 7/0516/9

SHEET No 13-2

PREPARED BY

DATE

F. ANGLISTINE

11 OCT 1955

CHECKED BY

DATE

INTRODUCTION —

IN THE BODY OF THE
CENTRE FUSelage PASSENGER, FRAMES IN GENERAL
ARE ASSUMED RIGID UNDER DUCT LOADING
ARISING FROM THE TENSION & BENDING TYPES
OF REDUNDANT GACHES.

DUE TO THE FLEXIBLE NATURE OF
ATTACHING THE DUCT TO FRAMES AND, THE ABOVE
ASSUMPTION IS INVALID & THE IMPORTANCE OF
INVESTIGATING THE INFLUENCE OF THE DIRECTION
OF THE CONNECTING RING IS INDICATED.

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 710510/9

SHEET No. 13-5

AIRCRAFT:

C-105

CENTRE SECTION

PREPARED BY

DATE

H. AUGUSTINE

11/10/55

CHECKED BY

DATE

1.1 ACTUAL STRUCTURE -

- THE DUCT RING IS SO CONSTRUCTED THAT THE DUCT CIRCUMFERENTIAL JOINT IS A FREE SURFACE; I.E. THE DUCT SEGMENT FWD. OF THE RING CAN TRANSFER NO FORCES TO THE DUCT SEGMENT AFT OF RING.

ONLY THE FWD. RING - ATTACHED TO FWD. DUCT SEGMENT - IS CONNECTED TO A.R.S. VIA THE TWO PIN PICKUPS.

1.2. IDEALISED STRUCTURE -

- THE DISPLACEMENT OF THE DUCT $\frac{1}{4}$ FROM THE HORIZONTAL IS IGNORED.
- THE TWO PIN ATTACHMENTS ARE ASSUMED TO BE AT 45° EITHER SIDE OF DUCT $\frac{1}{4}$.
- THE TWO FRAMES ARE EACH THE SINGLE THICKNESS ($t = .0036$) TRAPEZOIDAL PANELS THAT

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/9

SHEET NO. 13-11

AIRCRAFT:

C-105

ENGINE ROOM

PREPARED BY

DATE

F. AUGUSTINE

11/OCT/55

CHECKED BY

DATE

ESSENTIALLY, THE METHOD CONSISTS OF —

- REMOVING THE CONCENTRATED LOAD TYPE OF PROBLEM, G₁, G₂ (REF. PG. 7), BY APPROXIMATING SHEAR FLOWS AS OBTAINED BY A FOURIER SERIES
- ASSUMING A GENERAL STRESS DISTRIBUTION IN THE DUCT (TRIGONOMETRIC SERIES).
- APPLYING THE BOUNDARY CONDITIONS (REF. PG. 5) & CALCULATING THE UNKNOWN IN THE TRIG. SERIES BY THEORY OF LEAST SQUARES.
- INCLUSION OF RESULTS IN THE BODY OF THE PROBLEM IN A MANNER CONSISTENT WITH EXISTING REDUNDANT GROUP DEFINITIONS.

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT No.

113519/7

SHEET No.

113-14

AIRCRAFT:

PREPARED BY

DATE

CHECKED BY

DATE

REPLACE THE CONVENTIONAL HOOPS - COLUMNS
 G_1, G_2 (REF. 9-13) BY FRAMES
SHEAR FRAMES

$$G_1 N \cdot G_1 = 2\pi R^2 (\tau_{411} \cdot t)$$

$$G_1 = 2\pi R (\tau_{411} \cdot t)$$

$$G_2 = \frac{\pi R^2 (\tau_{418} \cdot t)}{2t} - (\text{REF. 9-47})$$

SUBSCRIPTS -

N - SHEAR

T - TORSION

B - BENDING

R - TEAM OF SERVICE

REPLACING G_1 (TORSION) -

$$(r)_{NT} = 2\pi R \left[\frac{1}{2} + \sum_{n=2}^{\infty} \left(\cos \frac{n\pi}{4} - n \cdot \sin \frac{n\pi}{4} \right) \cos n\theta \right] (\tau_{411} \cdot t) - (1)$$

REPLACING G_2 (BENDING) -

$$(r)_{NB} = -\frac{2\pi R}{\pi} \left[\sum_{n=1}^{\infty} \left(\sin \frac{n\pi}{4} \right) \cdot \sin n\theta \right] \cdot G_2 - (2)$$

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0510/9

SHEET No. 13-16

AIRCRAFT:

C-105

CENTRE SECTION

PREPARED BY

DATE

F. BUSHNETT

1/0/25

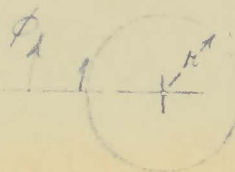
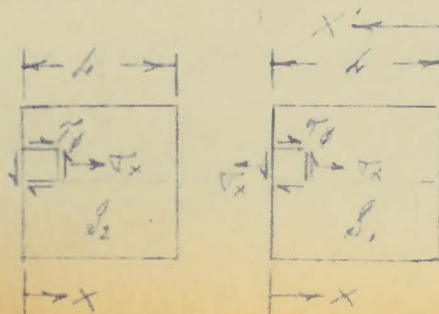
CHECKED BY

DATE

STRESS DISTRIBUTION IN DUCTS 1, 2 & 3.

DUE TO THE ANG DEFORMATION UNDER THE SHEAR FORCE (T_{WT} & T_{Wb}) & HENCE DUCT DEFORMATION, A STRESS DISTRIBUTION IS INVOLVED IN THE DUCT SEGMENTS 1, 2 & 3. (FOR CONSISTENCY WITH EXISTING STRESS DISTRIBUTIONS IN THE DUCT, THE SIGN CONVENTIONS & ANGULAR DISPLACEMENT ϕ ARE DEFINED AS ON PG. 12.)

A GENERAL FORM FOR THIS DISTRIBUTION IS ARRIVED AT FROM THE DIFFERENTIAL EQUATIONS OF EQUILIBRIUM FOR A CIRCULAR SHEAR PAIR THE BOUNDARY CONDITIONS.



VIEW ON
A. H. S. A/C
L. H. F. W. D.

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0510/9

SHEET No. 1326

AIRCRAFT:

C-105

ENGINE SECTION

PREPARED BY

DATE

CHECKED BY

DATE

V_{s1}, V_{s2} — STRAIN ENERGY OF DUCTS —

GENERAL —

$$V_s = \frac{\pi L}{2} \int_0^{2\pi} \int_0^h \left(\frac{G}{G_0} \right) \left(x^2 + r^2 \right) d\phi dx$$

WRITE $x = h - x'$ (ACF. PG. 16)

$$\text{THEN } (1 - x/h) = \frac{h - x}{h} = \frac{x'}{h}$$

SUBSTITUTING $(1 - x/h)$ IN T_{s0} (PG. 19), THE EXPRESSIONS FOR T_s & T_{s1} ARE SIMILAR FOR BOTH S. & S₁.

$$\therefore V_{s1} + V_{s2} = \left(\frac{\pi L}{G} \right) \int_0^{2\pi} \int_0^h \left(\frac{G}{G_0} \right) \left(x_{s1}^2 + x_{s2}^2 \right) d\phi dx + \left(\frac{\pi L}{G_0} \right) \int_0^{2\pi} \int_0^h \left(\frac{G}{G_0} \right) d\phi dx$$

($\because$ OF ORTHOGONALITY M DOES NOT COUPLE WITH HARMONICS)

$$V_{s1} + V_{s2} = \left(\frac{\pi L}{G_0} \right) \sum_{n=1}^{\infty} \left(\frac{G h^2}{3 G_0 n^2} + \frac{1}{n^2} \right) \left((T_{s1n})^2 + (T_{s2n})^2 \right) + (G_0) \int_0^{2\pi} \int_0^h d\phi dx$$

$$\text{FORM } \frac{\partial V_{s1+s2}}{\partial (T_{s1n})} \text{ \& } \frac{\partial V_{s1+s2}}{\partial (T_{s2n})} =$$

$$\frac{\partial V_{s1+s2}}{\partial (T_{s1n})} = \left(\frac{2 \pi L}{G_0} \right) \left(\frac{G h^2}{3 G_0 n^2} + \frac{1}{n^2} \right) (T_{s1n})$$

$$\frac{\partial V_{s1+s2}}{\partial (T_{s2n})} = \left(\frac{2 \pi L}{G_0} \right) \left(\frac{G h^2}{3 G_0 n^2} + \frac{1}{n^2} \right) (T_{s2n})$$

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0570/7

SHEET No. 1329

AIRCRAFT:

C-105

CEHRAE SECTION

PREPARED BY

R. AUGUSTINE

DATE

13/10/56

CHECKED BY

DATE

V_{f1} - STRAIN ENERGY OF FANNIE AS A RING
(REF. APPENDIX).

$$V_{f1} = \frac{\pi D^5}{2 E I_{f1}} \sum_{n=1}^{\infty} \frac{(L_n^2 + B_n^2)}{24 n^2 - 11^2} \quad (\text{REF. APPENDIX } \S 18.25)$$

WHERE -

$$L_n = 2 \pi \frac{D}{n}$$

$$B_n = 2 \pi \frac{D}{n}$$

$$V_{f1} = \left(\frac{4 \pi D^5}{4^2 E I_{f1}} \right) \sum_{n=1}^{\infty} \left(\frac{(T_{on}(t))^2 + (T_{on}(t))^2}{n^2 (n^2 - 11^2)} \right)$$

FORM $\frac{\partial V_{f1}}{\partial (T_{on}(t))}$ & $\frac{\partial V_{f1}}{\partial (T_{on}(t))}$ -

$$\frac{\partial V_{f1}}{\partial (T_{on}(t))} = \left(\frac{4 \pi D^5}{4^2 E I_{f1}} \right) \left(\frac{1}{n^2 (n^2 - 11^2)} \right) (T_{on}(t)).$$

$$\frac{\partial V_{f1}}{\partial (T_{on}(t))} = \left(\frac{4 \pi D^5}{4^2 E I_{f1}} \right) \left(\frac{1}{n^2 (n^2 - 11^2)} \right) (T_{on}(t)).$$

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0570/9

SHEET No. 13.28.

AIRCRAFT:

C-105

CENTRAL SECTION

PREPARED BY

F. AUGUSTINE

DATE

19/OCT/55

CHECKED BY

DATE

$$\text{FORM } \frac{\partial V}{\partial (a_n t)} = 0 \quad \& \quad \frac{\partial V}{\partial (b_n t)} = 0 \quad (\text{REF. PGS. 24, 25, 26, 27}).$$

$$\frac{\partial V}{\partial (a_n t)} = 0$$

$$\frac{n^2}{4I_n} \left(\frac{V_{a_n t}}{n} + a_n \cos \frac{n\pi}{2} (V_{a_n t}) + b_n \sin \frac{n\pi}{2} (V_{a_n t}) \right) \\ \frac{n^3(n^2-1)^2}{4I_n}$$

$$+ 2 \left(\frac{E}{G} \right) \left(\frac{4^2}{3n^2} \left(\frac{G}{E} \right) + \frac{1}{n^2} \right) (V_{a_n t})$$

$$+ \frac{4n^2}{4I_f} \left(\frac{1}{n^3(n^2-1)^2} \right) (V_{a_n t}) = 0$$

$$\frac{\partial V}{\partial (b_n t)} = 0$$

$$\frac{n^2}{4I_n} \left(\frac{V_{b_n t}}{n} + a_n \sin \frac{n\pi}{2} (V_{b_n t}) - b_n \cos \frac{n\pi}{2} (V_{b_n t}) \right) \\ \frac{n^3(n^2-1)^2}{4I_n}$$

$$+ 2 \left(\frac{E}{G} \right) \left(\frac{4^2}{3n^2} \left(\frac{G}{E} \right) + \frac{1}{n^2} \right) (V_{b_n t})$$

$$+ \frac{4n^2}{4I_f} \left(\frac{1}{n^3(n^2-1)^2} \right) (V_{b_n t}) = 0$$

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/570/9

SHEET NO. 013-29

AIRCRAFT:

C-105

CENTRE SECTION

PREPARED BY

E. AUGUSTINE

DATE

12/OCT/55

CHECKED BY

DATE

COLLECTING TERMS -

$$\int \frac{\left(\frac{D^2}{4}\right) \left(\frac{1}{I_n} + \frac{4}{I_f}\right)}{n^3(n^2-1)^2} + 2 \left(\frac{E}{Gt}\right) \left(\frac{4^2}{32n^2} \left(\frac{E}{E}\right) + \frac{1}{n^2}\right) \phi(\tau_{ni}, t)$$

$$= \frac{\left(\frac{D^2}{4 I_n}\right)}{n^3(n^2-1)^2} \int -a_n \cos \frac{n\pi}{2} (\tau_{ni}, t) - b_n \sin \frac{n\pi}{2} (\tau_{ni}, t) \phi$$

$$\int \frac{\left(\frac{D^2}{4}\right) \left(\frac{1}{I_n} + \frac{4}{I_f}\right)}{n^3(n^2-1)^2} + 2 \left(\frac{E}{Gt}\right) \left(\frac{4^2}{32n^2} \left(\frac{E}{E}\right) + \frac{1}{n^2}\right) \phi(\tau_{ni}, t)$$

$$= \frac{\left(\frac{D^2}{4 I_n}\right)}{n^3(n^2-1)^2} \int -a_n \sin \frac{n\pi}{2} (\tau_{ni}, t) + b_n \cos \frac{n\pi}{2} (\tau_{ni}, t) \phi$$

DEFORMATION OF P_i UNDER BENDING (τ_{ni}, t) IS ASSUMED NON-EXISTENT -

∴ THE SOLUTION FOR (τ_{ni}, t) & (τ_{bi}, t) MAY BE WRITTEN -

$$(\tau_{ni}, t) = -a_n \cdot a_n \cos \frac{n\pi}{2} (\tau_{ni}, t) - b_n \cdot b_n \sin \frac{n\pi}{2} (\tau_{ni}, t)$$

$$(\tau_{bi}, t) = -a_n \cdot a_n \sin \frac{n\pi}{2} (\tau_{ni}, t) + b_n \cdot b_n \cos \frac{n\pi}{2} (\tau_{ni}, t)$$

WHERE FOR THE CASE OF a_n , $I_f = \infty$.

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO.

7/0510/9

SHEET NO.

13-30

AIRCRAFT:

C-105

CENTRE SECTION

PREPARED BY

F. AUGUSTINE

DATE

18/05/55

CHECKED BY

DATE

CALCULATION OF T_B & T_{XB} —

SUBSTITUTE T_{an} & T_{bn} IN T_B & T_{XB} — PG. 17.

$$T_B = (1/4) \sum_{\substack{a: n=1 \\ b: n=2}}^{\infty} \frac{A_n \cdot a_n}{n} \cdot T_{an} \cdot \sin(\theta + \theta_n) + \frac{A_n \cdot b_n}{n} \cdot T_{bn} \cdot \cos(\theta + \theta_n) \quad \text{--- (1)}$$

$$T_{XB} = (1 - 1/4) \sum_{\substack{a: n=1 \\ b: n=2}}^{\infty} A_n \cdot a_n \cdot T_{an} \cdot \cos(\theta + \theta_n) - A_n \cdot b_n \cdot T_{bn} \cdot \sin(\theta + \theta_n) \quad \text{--- (2)}$$

WRITE —

$$A_n \cdot a_n = T_{nb} \quad (\text{b - BENDING})$$

$$A_n \cdot b_n = T_{nt} \quad (\text{T - TENSION})$$

THEN —

$$\frac{T_B}{T_{an}} = (1/4) \sum_{n=1}^{\infty} \frac{T_{nb}}{n} \cdot \sin(\theta + \theta_n); \quad \frac{T_B}{T_{bn}} = (1/4) \sum_{n=2}^{\infty} \frac{T_{nt}}{n} \cdot \cos(\theta + \theta_n) \quad \text{--- (3)}$$

$$\frac{T_{XB}}{T_{an}} = -(1 - 1/4) \sum_{n=1}^{\infty} T_{nb} \cdot \cos(\theta + \theta_n); \quad \frac{T_{XB}}{T_{bn}} = -(1 - 1/4) \sum_{n=2}^{\infty} T_{nt} \cdot \sin(\theta + \theta_n) \quad \text{--- (4)}$$

CALCULATION OF ENERGY COEFFICIENTS FOR INCLUSION IN C₁₂ MATRIX.

CALCULATION IS DONE IN TWO STAGES -

1. CONSIDERING TERMS IN T_{2q18} & T_{419} ONLY
2. CONSIDERING TERMS IN T_{2q18} & T_{419} THAT COUPLE WITH EXISTING STAGGER (REF. PG. 12)

- THESE TERMS WILL ONLY OCCUR FOR $n=1, 2$, ALL OTHERS VANISHING BY VIRTUE OF THE ORTHOGONALITY OF THE FUNCTIONS CONCERNED.

1. SUBSTITUTE T_{2q18} & T_{419} IN V_0, V_1, V_2, V_3 & EXPRESS IN TERMS OF $T_{2q18} = a_n$ & $T_{419} = b_n$ (REF. PG. 12)

V_n - STRAIN ENERGY OF RING - (REF. PG. 25, 28)

$$\frac{2 V_0}{T_{2q18}^2} = \frac{\pi R^4 t^2}{4 \mu I_n} \sum_{n=1}^{\infty} \frac{(T_{2q18} - a_n)^2}{n^2 (n^2 - 1)^2}$$

$$\frac{2 V_1}{(T_{419}^2)^2} = \frac{\pi R^4 t^2}{4 \mu I_n} \sum_{n=2}^{\infty} \frac{(T_{419} - b_n)^2}{n^2 (n^2 - 1)^2}$$

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO.

7/0510/9

SHEET NO.

1328

AIRCRAFT:

C-105

CENTRAL SECTION

PREPARED BY

G. ARGENTI

DATE

12/OCT/58

CHECKED BY

DATE

SUBSTITUTE NUMERICAL VALUES IN THE
EXPRESSIONS FOR V_n , V_{s1} , S_2 , E , V_{f1} -

$$V_n = 18.00 \text{ in.}$$

$$G = 4 \times 10^3 \text{ KIPS/in}^2$$

$$V_{s1} = 16.25 \text{ in.}$$

$$I_n = 2013 \text{ in}^4 - \text{REF.}$$

$$E = (10.6) 10^3 \text{ KIPS/in}^2$$

$$I_{f1} = 1.23 \text{ in}^4 - \text{APPENDIX.}$$

$$t = .032$$

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0510/7

SHEET No. 13-24

AIRCRAFT:

C-105

CENTRE SECTION

PREPARED BY

DATE

CHECKED BY

DATE

NUMERICAL CALCULATIONS - CONT'D.

(i) BENDING

$$A_n = \frac{\pi (P/p)}{.00058606681 n^4 (n^2-1)^2 + .00571679 n^2 (n^2-1)^2 + 1.10769231}$$

$$T_{nb} = A_n \cdot D_n$$

$$\frac{T_{nb}}{n} = \frac{\sqrt{2} (P/p) \sin \left(\frac{n\pi}{4} \right)}{.00058606681 n^4 (n^2-1)^2 + .00571679 n^2 (n^2-1)^2 + 1.10769231}$$

THEN -

$$\frac{2(V_3 + V_2)}{T_{a18}^2} = 3492.441270 \left[\sum_{n=1}^{\infty} \frac{\left(\frac{T_{nb}}{n} - \sqrt{2} \cdot \sin \frac{n\pi}{4} \right)^2}{n^2 (n^2-1)^2} \right] / (10)^{-3}$$

$$\frac{2(V_3 + V_2)}{T_{a18}^2} = \left[\sum_{n=1}^{\infty} (1.849394690 n^2 + 18.03993718) \left(\frac{T_{nb}}{n} \right)^2 \right] / (10)^{-3}$$

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0510/9

SHEET No. 13.88

AIRCRAFT:

PREPARED BY

DATE

F. A. WILKINSON

CHECKED BY

DATE

C-105

CENTRE SECTION

①	②	③	④	⑤	⑥	⑦	⑧	⑨	⑩
N	$N^2(1^2-1^2)$	$N(1^2-1^2)$	$\frac{N}{N(1^2-1^2)}$	$\frac{N}{N(1^2-1^2)}$	$\frac{N}{N(1^2-1^2)}$	$\frac{N}{N(1^2-1^2)}$	$\frac{N}{N(1^2-1^2)}$	$\frac{N}{N(1^2-1^2)}$	$\frac{N}{N(1^2-1^2)}$
1	0	0	1.000000	1.00	1.00	1.000000	1.000000	1.000000	1.000000
2	36	144	1.377800	1	1.000000	1.000000	1.000000	1.000000	1.000000
3	576	5184	1.428995	1/62	1.000000	1.000000	1.000000	1.000000	1.000000
4	3600	57600	1.428995	0	1.000000	1.000000	1.000000	1.000000	1.000000
5	14400	360000	1.428995	1/62	1.000000	1.000000	1.000000	1.000000	1.000000
6	44100	1577700	1.428995	1	1.000000	1.000000	1.000000	1.000000	1.000000
7	112896	5531200	1.428995	1/62	1.000000	1.000000	1.000000	1.000000	1.000000
8	259200	16457280	1.428995	0	1.000000	1.000000	1.000000	1.000000	1.000000
9	518400	41720448	1.428995	1/62	1.000000	1.000000	1.000000	1.000000	1.000000
10	980100	98010000	1.428995	1	1.000000	1.000000	1.000000	1.000000	1.000000
11	1742400	310336000	1.428995	1/62	1.000000	1.000000	1.000000	1.000000	1.000000
12	3149150	644242200	1.428995	0	1.000000	1.000000	1.000000	1.000000	1.000000
13	4789350	1061151600	1.428995	1/62	1.000000	1.000000	1.000000	1.000000	1.000000

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/9

SHEET NO. 13-28

AIRCRAFT.

PREPARED BY

DATE

F. AUGUSTINE

13/OCT/1955

CHECKED BY

DATE

C-105

CENTRE SECTION

$\textcircled{10}$ $= 1.10764 \times \textcircled{9}$ $(\frac{1}{n}) (\frac{\sum \text{Var}}{n})$	$\textcircled{11}$ $= (\textcircled{10} - \textcircled{7})^2$ $(\frac{1}{n}) (\frac{\sum \text{Var}}{n}) - (\frac{1}{n}) (\frac{\sum \text{Var}}{n})^2$	$\textcircled{12}$ $\frac{2.782,043}{2.782,043}$ $\frac{2.782,043}{2.782,043}$	$\textcircled{13}$ $= \textcircled{10} \times \textcircled{13}$ $\frac{2.782,043}{(2.782,043)^2}$	$\textcircled{14}$ 14.61751538 $\times (102.577 \frac{1}{2})$	$\textcircled{15}$ $= \textcircled{2} \times \textcircled{14}$ $\frac{2.15,12}{(94.0,5)^2}$	$\textcircled{16}$ $\frac{2.252,221}{(2.252,221)^2}$ $\frac{2.252,221}{2.252,221}$	$\textcircled{17}$ $= \textcircled{8}^2 \times \textcircled{16}$ $\frac{2.252,221}{(2.252,221)^2}$
.081165	.681928	77.279	229.536	.21060763	87.6958	.6453	220.913
.766735	.910780	4.82794	115.488	.76367575	16.2384	.434589	1.84503
.039435	.843813	.792790	2.970	.90725244	.059	.043019	.00090
.021430	.959655	.193197	6.136	.51024960	.023	.0062534	.00006
.011723	.730774	.0630849	9.067	.22553768	.00848	.044704	.00005
.002222	.974634	.0246425	2.153	.16567779	.009	.0040753	—
.000204	.999190	.0109523	.044	.0814326	.000004	.0013135	—
.000455	.987261	.0052666	.684	.02366174	.00003	.0002583	—
.000351	.985960	.0038853	1.135	.9223273	—	.0002435	—
.000141	.989679	.0015967	.460	.95172306	—	.0000286	—
.000082	.979665	.000944777	.004	.92899245	—	.0000523	—
		$\sum_{n=3}^{14}$	139.4355		16.2384		1.87667

$$2V_n = \frac{\pi \epsilon^2 t^2}{36 \pi^2 \epsilon I_n} \left(\left(\frac{V_{2n}}{2} - C_2 \right) (V_{2n}) (V_{2n}) + \frac{1}{2} (V_{4n})^2 \right)$$

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT No 7/0510/9

SHEET No 1340

AIRCRAFT:

C-105

CENTRE SECTION

PREPARED BY

E. AUGUSTINE

DATE

14/OCT/55

CHECKED BY

DATE

Coupling - Cont'd.

$V_{g1} + V_{g2}$ - DUCTS -

CONSIDER TERMS -

$$(T_{\theta})_{g1} = -\frac{\rho}{2h} \int b_2 \cdot h_{22} (T_{g11} \cdot t) + (T_{Wg11} \cdot t) \int \cos 2\phi$$

$$T_{Xg1} = (1 - 1/4) \int b_2 \cdot h_{22} (T_{g11} \cdot t) + T_{Wg11} \int \sin 2\phi$$

THEN -

$$2V_{g1} = \left(\frac{\pi R^3}{GAT} \right) \left(\frac{Gh_1^2}{2Gh^2} + \frac{1}{4} \right) \int b_2 \cdot h_{22} (T_{g11} \cdot t) + (T_{Wg11} \cdot t) \int \cos 2\phi$$

E' -

$$(T_{\theta})_{g2} = \frac{\rho}{2h} \int b_2 \cdot h_{22} (T_{g11} \cdot t) + (T_{Wg11} \cdot t - T_{Wg05} \cdot t) \int \cos 2\phi$$

$$+ \frac{\rho}{h} \int - (T_{g06} \cdot t) + a_1 h_{11} (T_{g018} \cdot t) \int \cos \phi$$

$$T_{Xg2} = \left[\frac{\rho}{h} \int T_{Wg11} - T_{Wg05} + b_2 \cdot h_{22} T_{g11} \int + T_{Wg05} \right] \sin 2\phi$$

$$+ \left[\frac{\rho}{h} \int - T_{g06} + a_1 h_{11} T_{g018} \right] \sin \phi$$

THEN - WRITING $a_1 \cdot h_{11} = T_{16}$

$$b_2 \cdot h_{22} = T_{27}$$

E' SUBSTITUTING IN FORMULAE - PGE. 26 -

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO.

7/0510/7

SHEET NO.

13-A1

AIRCRAFT:

C-105

CENTRE SECTION

PREPARED BY

E. AUGUSTINE

DATE

14/05/55

CHECKED BY

DATE

COMPAING - CONT'D. -

$$2V_1 = \left(\frac{\pi R^3 L}{Gh} \right) \left(\frac{Gh^2}{3EI^2} + \frac{1}{4} \right) \left\{ (\tau_{W419})^2 + 2\tau_{2T} (\tau_{W419}) (\tau_{419}) \right\}$$

$$2V_2 = \left(\frac{\pi R^3 L}{Gh} \right) \left[\left(\frac{Gh^2}{3EI^2} + \frac{1}{4} \right) (\tau_{W419}^2 + \tau_{W405}^2) \right.$$

$$+ \left(\frac{Gh^2}{3EI^2} + 1 \right) (\tau_{2106}^2)$$

$$+ \left(\frac{Gh^2}{3EI^2} - \frac{1}{4} \right) (\tau_{W419} \tau_{W405})$$

$$+ \left(\frac{2Gh^2}{3EI^2} + \frac{1}{4} \right) \tau_{2T} (\tau_{W419} \tau_{419})$$

$$+ \left(\frac{Gh^2}{3EI^2} - \frac{1}{4} \right) \tau_{2T} (\tau_{W405} \tau_{419})$$

$$+ \left(\frac{Gh^2}{3EI^2} - 2 \right) \tau_{2T} (\tau_{B415} \tau_{B406})$$

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO.

7/0370/9

SHEET NO

1342

PREPARED BY

DATE

E. AUGUSTINE

14 OCT. 1955

CHECKED BY

DATE

AIRCRAFT:

C-105

CENTRE SECTION

COMPARING - CONT'D -

V_f - FRAME F_1

CONSIDER SHEAR FROM TENSORS -

$$\tau_{11} = -\frac{Q_x}{A} \left(\tau_{1111} - \frac{1}{4} \tau_{1122} + b_2 \cdot b_{22} \tau_{1111} \right)$$

THEN - WAITING $b_2 \cdot b_{22} = \tau_{22}$

SUBSTITUTING IN FORMULA - PG. 27 -

$$2V_{f_1} = \frac{\pi A^2 t^2}{36 h^2 A I_A} \left(\left(\tau_{1111} \right)^2 + \frac{1}{4} \left(\tau_{1122} \right)^2 - \left(\tau_{1111} \cdot \tau_{1122} \right) \right. \\ \left. + 2 \tau_{22} \cdot \left(\tau_{1111} \cdot \tau_{1111} \right) - \tau_{22} \left(\tau_{1122} \cdot \tau_{1111} \right) \right)$$

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT No.

7/0510/9

SHEET No.

1343

PREPARED BY

DATE

F. AUGUSTINE

10/1/57

CHECKED BY

DATE

AIRCRAFT:

C-105

CENTRE SECTION

SUBSTITUTE NUMERICAL VALUES FOR
CALCULATION OF COMPARING, WARPING,
& BENDING (406) ENERGY COEFFICIENTS.

(REF. PG. 33)

(i) BENDING - 1ST HARMONIC ($N=1$) ONLY.

$$2V_2 = -2 \times .00855462 T_{418} \cdot T_{406} + .00994467 T_{400}^2$$

(ii) TEARSTICH -

$$2V_2 = 2 \times .08409830 T_{417} \cdot T_{407} + .02427386 T_{407}^2$$

$$2V_3 = -2 \times .01194876 T_{419} \cdot T_{417} + .00317183 T_{417}^2$$

$$2V_3 = -2 \times .01194876 T_{419} \cdot T_{417} + 2 \times .00673646 T_{417} \cdot T_{405} \\ + .00317993 T_{417}^2 + .00317183 T_{405}^2$$

$$- 2 \times .00199272 T_{419} \cdot T_{405}$$

$$2V_4 = -2 \times .06020079 T_{417} \cdot T_{407} + 2 \times .03010039 T_{417} \cdot T_{405} \\ + .01602079 T_{417}^2 + .00400517 T_{405}^2$$

$$- 2 \times .00801038 T_{417} \cdot T_{405}$$

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO.

7/0510/9

SHEET NO.

1344

AIRCRAFT:

C-105

CENTRE SECTION

PREPARED BY

F. AUGUSTINE

DATE

17/05/55

CHECKED BY

DATE

PREPARATION OF THE FORGONE RESULTS

FOR INTRODUCTION INTO PROBLEM #4.

THE BENDING MOMENT & SHEAR
COEFFICIENTS ARE TABULATED BELOW. SUM OF
TERMS FOR $n \geq 2$ & $n \geq 3$ RESPECTIVELY ARE
WRITTEN SEPARATELY.

(i). BENDING (AIR) - REF. PGS. 35, 43.

	$n \geq 2$	$n = 1$		
	$T_{n,18}^2$	$T_{n,18}^2$	$T_{n,18}^2$	$T_{n,18}^2 T_{n,20}^2$
$2V_n =$	0.0321139	0	0	0
$2V_3 =$	0.0321139	0	0	0
$2V_2 =$	0.0321139	0.01988758	0.00994467	2×0.00555769
$2V_1 =$	0	0	0	0

$$2V_{TOTAL} = 0.0658146 T_{n,18}^2 - 2 \times 0.00555769 T_{n,18}^2 T_{n,20}^2 + 0.00994467 T_{n,20}^2$$

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/9

SHEET NO. 13-46

AIRCRAFT:

PREPARED BY

DATE

A. RAVENHILL

14/01/55

CHECKED BY

DATE

C-105

CENTRE SECTION

(1C) TORSION -

THE FUNCTION $2V_4$, (REF. (1C) TORSION - PG. 43) PARTIALLY ANTICIPATES THE RESULTS OF PROBLEM #A SECTION. THIS, BY VIRTUE OF THE FACT THAT, IN GENERAL, THE FRAME ENERGY IS EXPRESSED IN THE G_{12} MATRIX AS A FUNCTION OF ONE STRESS ONLY (τ_{944} - THE AVERAGE SHEAR STRESS OF THE FRAME AS AN IDEALIZED TRIANGULAR PANEL). THIS STRESS (932) IN TURN IS EXPRESSED, THRU THE MEDIUM OF THE H_{12} MATRIX, AS A FUNCTION OF THE WRAPPING GROUPS (406) & (119). AND SINCE $\left\{ \begin{array}{l} \tau_{W411} = V_{406} \\ \tau_{W405} = V_{119} \end{array} \right.$ (REF. PG. 121), THE TERMS IN $2V_4$, IN $(\tau_{W411})^2$, $(\tau_{W405})^2$ & $(\tau_{W411} \tau_{W405})$ (REF. PG. 43) WILL APPEAR IN THE NORMAL SOLUTION OF PROBLEM #A.

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/9

SHEET NO. 1348

AIRCRAFT:

C-105

CENTRE SECTION

PREPARED BY

F. AUGUSTINE

DATE

19/01/55

CHECKED BY

DATE

$2V_f$ IS ACCORDINGLY CONVERTED BY WRITING -

$$\sigma_w = (\sigma_{w417} - \frac{1}{2} \sigma_{w495})$$

$$2V_f = .01602077 \sigma_w^2 - 2 \times .06020077 \tau_{419} \cdot \sigma_w$$

THEN -

- WRITE: $\sigma_w = -.327364528 \tau_{424}$ (REF.)

CHANGE 417 TO 434 - A NEW STRESS POINT ON THE FRAME TO REPRESENT FRAME STRESSES DUE TO THE 2ND TORSION HARMONIC

- ADD THE TERM IN τ_{434}^2 (REF. SHEET 45).
 $n=2$

THEN -

$$2V_f = .0071614 \tau_{434}^2 + 2 \times .01970718 \tau_{434} \cdot \tau_{434} + .22631473 \tau_{434}^2$$

FOR THE REMAINING TERMS -

GIVE $2V_2, 2V_3, 2V_4$ FOR $n \geq 2$ + $2V_f$ FOR $n \geq 3$

INTRODUCE THE RESULTS INTO THE

CIA ACCORDING TO EXISTING STRESS POINTS.
(SEE FOLLOWING PG. FOR TABULATION).

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/5570/9

SHEET NO. 13.42.

AIRCRAFT:

C-105

CENTRE SECTION

PREPARED BY

DATE

H. AUGUSTINE

17/05/55

CHECKED BY

DATE

APPENDIX

FRAME (F) DISTORTION AS INFLUENCED BY
DUCT SHEAR FLOWS.

THE EFFECT OF f ON DUCT SHEAR FLOWS IS EXPRESSED BY CONSIDERING THE FRAME AS A RING. FOR THIS PURPOSE IT IS NECESSARY TO FIND AN EQUIVALENT MOMENT OF INERTIA THAT GIVES THE SAME STRAIN ENERGY IN THE FRAME AS A RING AS IN THE FRAME ACTING UNDER THE ONLY LOADING THAT IS ASSUMED TO CAUSE DISTORTION AS A FRAME.

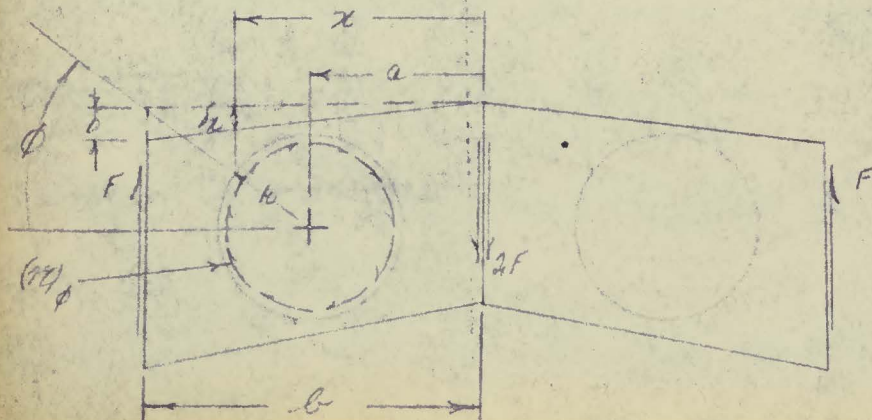


FIG. 1.

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0510/7

SHEET No. 13-49

AIRCRAFT:

C-105

CENTRE SECTION

PREPARED BY

DATE

F. ARGENTIUS

17/05/57

CHECKED BY

DATE

FROM TEST RESULTS IT HAS BEEN ESTABLISHED THAT FRAME BOWING OF THE RECTANGULAR TRAPEZOID TYPE SHOWN IN FIG. 1. ABOVE PRODUCES THE MAJOR FRAME DISTORTION. IT IS ASSUMED THEREFORE THAT IT IS THE ONLY TYPE OF LOADING CAUSING FRAME DISTORTION.

FROM TEST RESULTS-

$$F = A \cdot \delta \quad (A = 10 \text{ MIDS/INCH})$$

SO STRAIN ENERGY OF FRAME (ONE SIDE)-

$$V_{12} = \frac{1}{2} F \cdot \delta = \frac{1}{2} \cdot \frac{1}{A} \cdot F^2$$

APPLY TO THE DUCT CUT-OUT, ON THE FRAME, THE GENERAL SHEAR FORM (τ_{xy}) (REF. PG. 21.).

$$(\tau_{xy}) = \frac{2}{\pi} \sum_{n=1}^{\infty} \left(\frac{\tau_{0n} \cdot t}{n} \right) \cdot \sin n\phi + \left(\frac{\tau_{0n} \cdot t}{n} \right) \cdot \cos n\phi$$

NOTE: EFFECTS OF THE CONSTANT TERM K ARE IGNORED.

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT:

C-115

CENTRE SECTION

REPORT No. 7/0570/7

SHEET No. 13-50

PREPARED BY

DATE

H. AUGUSTINE

17/007/65

CHECKED BY

DATE

CALCULATE THE WORK DONE (V_{df}) BY THE SH. FLUX
(τ_{df}) DURING PERIPHERAL MOVEMENT OF THE
CIRCULAR DUST CUT DUE TO DISTORTION δ .

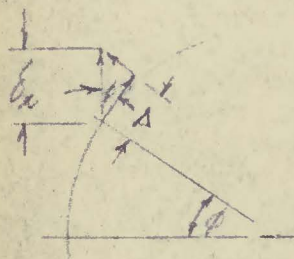


FIG. 2.

$$V_{df} = \frac{1}{2} \int_0^{2\pi} a(r, \phi) \cdot \Delta \phi \cdot d\phi \quad (\text{REF. FIG. 2}).$$

$$\Delta \phi = \delta_x \cdot \cos \phi = \frac{1}{2} b (a \cos \phi + r \cos^2 \phi) F.$$

$$\delta_x = \frac{1}{2} a \cdot \delta$$

$$x = a + r \cos \phi$$

$$\begin{aligned} V_{df} &= \left(\frac{\pi a^2}{2bh} \right) \cdot F \int_0^{2\pi} \left(\sum \left(\frac{\tau_{df}}{2} \right) \cdot \sin \phi + \left(\frac{\tau_{df}}{2} \right) \cdot \cos \phi \right) (a \cos \phi + r \cos^2 \phi) d\phi \\ &= \left(\frac{\pi a^2}{2bh} \right) \cdot F \left(a \left(\frac{\tau_{df}}{2} \right) + \frac{r}{2} \left(\frac{\tau_{df}}{2} \right) \right) \phi \end{aligned}$$

THE TERM τ_{df} ARISING FROM A DUST BOARDING
TYPE LOADING IS IGNORED (REF. 9-27 & 10).

$$V_{df} \approx \frac{\pi a^3}{2bh} \left(\frac{\tau_{df}}{2} \right) \cdot F$$

