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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

~~CONFIDENTIAL~~

Classification cancelled / Changed to UNCLASS
By authority of AVRS
Date 30 Sept 56
Signature
Unit / Rank / Appointment AVRS 5

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.

PREPARED BY C. F. Barrell DATE Dec'55
CHECKED BY E.C. Augustine. DATE Dec'55
SUPERVISED BY DATE Dec'55
APPROVED BY DATE

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

FORM 1316A

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
		C. E. Burrell	Dec'55
		CHECKED BY	DATE

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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		
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PREPARED BY		DATE
C. E. Burrell		Dec'55
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EXTRACTS FROM C 105 CENTRE FUSELAGE ANALYSIS

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

REPORT NO. 7/ 0510/9
SHEET NO. 3-1

AIRCRAFT:		PREPARED BY	DATE
C 105	Centre Fuselage	C. E. Burrell	Dec'55
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INTRODUCTION

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

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 $485.00^{in.}$.

In this solution the redundants total seventy one. Since it it 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 computors. 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 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 computor 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

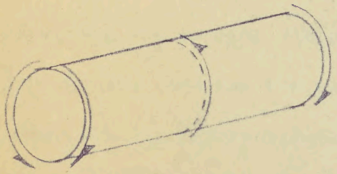
In order to reduce the number of redundants to seventy one, it has been necessary to lump the frames by twos, with the lumped frame located between the two. The frame at 469.50 in. is considered separately, as are, of course, the end bulkheads. Then each lumped frame is represented by a shear panel of equivalent thickness to provide the same strain energy. The load (in kips) transferred to each outer wall from the tank is

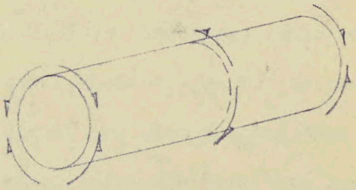
$0.131\ 238\ \tau_m$ for one frame

$0.262\ 475\ \tau_m$ for two frames

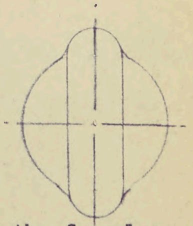
The side wall and the fuel tank are both represented by two booms and a shear web, following the usual lumping method.

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:

$$\sigma = [\sigma_{b1}(1-x/L)+\sigma_{b2}(x/L)]\sin\phi + [\sigma_{w1}(1-x/L)+\sigma_{w2}(x/L)]\sin 2\phi$$

$$\tau = (-\sigma_{b1}+\sigma_{b2})(r/L)\cos\phi + (-\sigma_{w1}+\sigma_{w2})(r/2L)\cos 2\phi + \tau_{d1}$$

where

σ = local axial stress k.s.i.
τ = local shear stress k.s.i.
r = radius of duct = $19.000^{\text{in.}}$
L = spacing between frames = $22.0^{\text{in.}}$
x = distance aft of fwd frame in bay
σ_{b1} = duct bending stress at fwd end of duct bay
σ_{b2} = duct bending stress at aft end of duct bay
σ_{w1} = duct warping stress at fwd end of duct bay
σ_{w2} = duct warping stress at aft end of duct bay
τ_{d1} = torsion stress in duct bay
ϕ = angle defined by figure two

By adjusting the spacing dimension to $16.25^{\text{in.}}$, the above would serve as approximate formulae for sub-problem 4. However, due to the ring at bulkhead 485.00, higher harmonics of stress are also present. For detail analysis, and stresses in the ring, reference should be made to section 13 of this report.

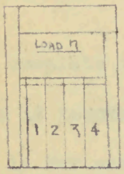
The magnesium webs and panels have been represented by the equivalent thickness of aluminum. This has been taken in the ratio of the two Moduli of Elasticity.

$$t_{alum} = t_{mg}\,\frac{6\times10^{6}}{10\times10^{6}}$$

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 computor work as a matrix (here designated S_{ia}) with the columns of stresses positoned as shown.



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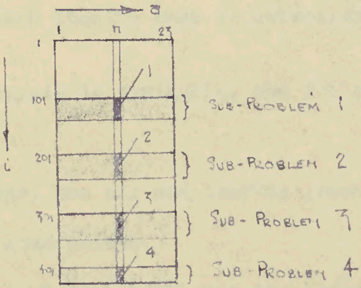
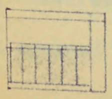


Deflections are also given for each of the twenty three load points due to all twenty three loads. All deflections are given in inches per kip per side. The set of pages of section 16 are arranged for computor work (here designated Z_{ab} or E^{w}_{ab} interchangably) with the columns of deflections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



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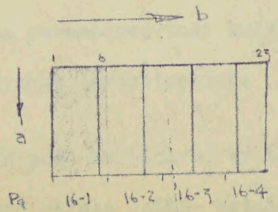


Zab

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.

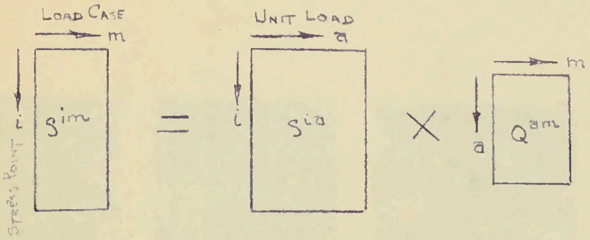
For detail stressing, stresses can be quickly established due to actual loading cases. For this, a Q^{am} matrix expressing the quantity of each unit load required for each loading case is established.

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 255.00in has been assumed rigid, pending further investigation.

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

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

REPORT NO.	7-0510-9
SHEET NO.	15-1
PREPARED BY	
DATE	Dec. 55
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DATE	
AIRCRAFT	C 105
	CENTRE FUSELAGE

LOAD No. 1

STRESSES DUE TO 1 KIP PER SIDE 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	6376	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	21888652
26	6007716 *	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 *	

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

REPORT NO. 7 0510 9
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C 105	CENTRE FUSELAGE		Dec. 55
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LOAD No. 1

STRESSES DUE TO 1 KIP PER SIDE 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*	208346*	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	1191560
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*	[illegible]3678*	364069*	11607156
29	148592	10[illegible]96*	22858*	14746365*
30	1480796	3996[illegible]7	4200942	
31	153994*	17791	246197	
32	1773456*	270339*	4373307*	
33	3524975	653013	8437056	2038721*
34	5093273*	786253*	10708127*	126330*
35	2666975	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*	

LOAD No. 3

STRESSES DUE TO 1 KIP PER SIDE AT LOCATION No. 3

STRESS POINT	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	1285694
16	433317*	29846*	243158*	433116*
17	1092878	6989*	45858*	5904*
18	2788234*	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	3686182	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*	

A. V. ROE CANADA LIMITED MALTON ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT No. 7-0510	
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C 105	CENTRE FUSELAGE		Dec. 55
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LOAD No. 4

STRESSES DUE TO 1 KIP PER SIDE AT LOCATION No. 4

STRESS POINT	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	66599*	10093*	73287*	9589*
18	1371144*	100769	2992293	863035
19	1146772	25578*	115786	131779*
20	181468	232458*	3381391*	4791782*
21	125579*	319824	5406474	8267150
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*	
		*		

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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	34000*	4141100	8073900
2	282100	165000	645300*	832000*
3	1067963*	334520	2680618	2564746
4	6657*	34710*	137152*	390850*
5	1001500*	105200	133700*	37600
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	598804	40375	859422*	638495*
15	1209003*	310556	2568312	1500745
16	118889*	22118*	189202*	386022*
17	223993*	6804	113917*	16570*
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*	

A. V. ROE CANADA LIMITED
MALTON ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7 0510 9
SHEET No. 15-6

AIRCRAFT: C 105

CENTRE FUSELAGE

PREPARED BY
DATE Dec. 55
CHECKED BY
DATE

LOAD No. 6

STRESSES DUE TO 1 KIP PER SIDE AT LOCATION NO. 6

STRESS POINT	PART 1	PART 2	PART 3	PART 4
1	543200	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*	4088031*
21	487600*	127539	3298243	6050246
22	457500*	151990*	4477915*	7662718*
23	942800	227281	5004072	4469033
24	759900	435348	2109042	4271993
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*	

A. V. ROE CANADA LIMITED
MALTON, ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO.	7/0510/9	
SHEET NO.	15-7	
AIRCRAFT	C 105	CENTRE FUSELAGE
PREPARED BY		DATE Dec. 55
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LOAD No. 7

STRESSES DUE TO 1 KIP PER SIDE AT LOCATION 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	3349436	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 *	5973359
29	108300 *	75576	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 MALTON - 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 KIP PER SIDE AT LOCATION No. 8

STRESS POINT	PART 1	PART 2	PART 3	PART 4
1	332400	029900	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*	336758*
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*	57476*	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*	172675*
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*	
			*	

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

AIRCRAFT C 105	CENTRE FUSELAGE	REPORT NO. 7-0510-9 SHEET NO. 15-9 PREPARED BY CHECKED BY DATE Dec. 55

LOAD No. 9

STRESSES DUE TO 1 KIP PER SIDE 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*	15220*
7	108800	27000	777788	46899
8	46200	4100*	505256*	2743312*
9	501600*	23000*	475900	2191170
10	52900*	20400*	846100*	3053180*
11	168300	47300	2347236	4885497
12		94900	3259604	3396823
13	315000	18800	540357*	3736613
14	101500	36200	1615527	58675
15	211100*	46900*	3412938	2360291
16	133700*	31500*	421409*	334507*
17	71000*	10000*	780149	82910*
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*	

A. V. ROE CANADA LIMITED
MALTON ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

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	CENTRE FUSELAGE
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LOAD No. 10

STRESSES DUE TO 1 KIP PER SIDE AT LOCATION 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 *	855200 *	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 *	
29	45900 *	13300 *	799157	2692816
30	98500 *	51700 *	2449294 *	3483590 *
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 *	

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

REPORT NO. 7 0510 9
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PREPARED BY
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LOAD No. 11

STRESSES DUE TO 1 KIP PER SIDE AT LOCATION NO. 11

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 *	13300 *	962500	
41	65900 *	37900 *	1814400 *	
42	224100 *	6700 *	737400	
43	22300 *	17700 *	867700 *	

A. V. ROE CANADA LIMITED
MALTON ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7-0519-9
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AIRCRAFT C 105

CENTRE FUSELAGE

LOAD No. 12

STRESSES DUE TO 1 KIP PER SIDE AT LOCATION No. 12

STRESS POINT	PART 1	PART 2	PART 3	PART 4
1	92500	19800	202800	381300 *
2	45400 *	6200 *	37900 *	530300
3	27400 *	4400 *	327400 *	1671171 *
4	17700 *	15700 *	275800 *	99462 *
5	42800 *	3100 *	114700 *	103000
6	1800 *	1000 *	87300 *	372200 *
7	21800	9600	348100	1099676
8	17100	8900	233600	3[illegible]1059
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 *	

A. V. ROE CANADA LIMITED
MALTON ONTARIO

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AIRCRAFT C 105

CENTRE FUSELAGE

LOAD No. 13

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

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	1191560
16	392016*	35311*	256185*	452969*
17	31391	286*	23930*	3819*
18	924529*	362593	4610659	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	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*	4373307*	
33	3524975	653013	8437056	2038721*
34	5093273*	786253*	10708127*	126330*
35	2666975	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*	

A. V. ROE CANADA LIMITED
MALTON ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

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LOAD No. 14

STRESSES DUE TO UNIT PRESSURE AT LOCATION 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 *	389717 *	506162 *
5	1057200	13000 *	36900	77600
6	1356700 *	11100 *	2572100	2146500
7	661566 *	43257	291408	407685
8	536292 *	142858 *	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	1252825 *	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	3920202	
31	441049	35489	307889	
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 *	
	*			

A V ROE CANADA LIMITED
MALTON ONTARIO
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LOAD No. 15

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

STRESS POINT	PART 1	PART 2	PART 3	PART 4
1	122000 *	60900	2717800	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	6926035
12		219595 *	1057686 *	1426003
13	7200 *	107905	3864819	7416746
14	693900	38433	306159 *	461009 *
15	611300	138069	2291113	1499191
16	1000	35043 *	446173 *	431762 *
17	156900	79290 *	57856	23543
18	541100 *	816226 *	1263929	629382
19	362000 *	59078	417966	113327 *
20	540500 *	134652 *	2577548 *	3771431 *
21	255100	26288 *	3455243	6297846
22	720100 *	85053 *	4518015 *	7887118 *
23	280900 *	428096	5573072	4620253
24	497600 *	333389 *	143558 *	3413593
25	244400	172830	5115291	8147467
26	689900	18344	473649 *	
27	933300	183097	2393752	
28	61500 *	42635 *	466244 *	7373040
29	93900	14564 *	29665	9126527 *
30	743700 *	77015 *	1598257	
31	529400 *	61613	413977	
32	850900 *	173276 *	3057428 *	
33	79500	143702	4399816	2314302 *
34	701200 *	226305 *	5623344 *	113327 *
35	264500 *	505862	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 *	251920	5423490	
43	747900 *	341787 *	6840040 *	
	*			

A. V. ROE CANADA LIMITED MALTON ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT No.	7/0510/9
		SHEET No.	15-16
AIRCRAFT		PREPARED BY	DATE
C 105	CENTRE FUSELAGE		Dec. 55
		CHECKED BY	DATE

LOAD No. 16

STRESSES DUE TO 1 KIP PER SIDE AT LOCATION No. 16

STRESS POINT	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	668076
8	112900*	65100*	1424095*	2559850*
9	222500	9300	57500*	3330310
10	425200*	44300*	817300*	4255970*
11	96200*	6300*	5233239	6331611
12		83500*	3383840*	795251
13	83000*	17500	1732836	4711092
14	451700	56200	290064	182686*
15	156900	64700	1797117	1525756
16	28700	3000*	391119*	393158*
17	69700	8700	71502*	60574
18	209500*	80500*	953138*	379431
19	135600*	42300*	699346	88275*
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	3269093*
34	399600*	64400*	3105806*	88275*
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*	
	*	*		

A. V. ROE CANADA LIMITED MALTON ONTARIO	REPORT NO.	7 0510
TECHNICAL DEPARTMENT (Aircraft)	SHEET NO.	15-17
AIRCRAFT	PREPARED BY	DATE
C 105 — CENTRE FUSELAGE		Dec. 55
	CHECKED BY	DATE

LOAD No. 17

STRESSES DUE TO 1 KIP PER SIDE 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	25200	6000	577338	1555705
4	2600	2600	51004 *	354834 *
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 *	1687090 *
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	1965663
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 *	639575 *	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	115640	
43	192700 *	37700 *	1687090 *	

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

REPORT No.	7 0510 9
SHEET No.	15-18
PREPARED BY	
DATE	Dec. 55
CHECKED BY	
DATE	
AIRCRAFT	C 105
	CENTRE FUSELAGE

LOAD No. 18

STRESSES DUE TO 1 KIP PER SIDE AT LOCATION No. 18

STRESS POINT	PART 1	PART 2	PART 3	PART 4
1	7300 *	1200 *	6100	218500
2	36400	5500	89300	83300
3	4000	400	61100	382894
4	400	500	6100 *	106028 *
5	7800	600	20700	19300 *
6	6000 *	1300 *	48400 *	264900 *
7	3900 *	1900 *	62600 *	197065 *
8	6700 *	4600 *	124100 *	276024 *
9	9600	500	9700 *	87100 *
10	27100 *	3000 *	50200 *	78300 *
11	4300 *	500 *	53500 *	222545 *
12		4300 *	203700 *	747979 *
13	9300 *	800 *	37800	378104
14	36900	6100	102500	23368
15	1900	1300	116500	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 *	267916 *
21	6700	200	23700 *	136714 *
22	25500 *	3400 *	59400 *	82346 *
23	3400 *	1400 *	80700 *	4877445
24	7900 *	7400 *	322000 *	1655681 *
25	11100 *	300 *	100100	505500
26	43300	7500	106500	
27	1300	2800	215400	
28	3500	200	54100 *	1118621
29	4200	1500	27100	1057753 *
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 *	

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

REPORT NO. 7/0510/9
SHEET NO. 15-19

AIRCRAFT
C 105

CENTRE FUSELAGE

PREPARED BY
DATE Dec. 53
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	15200	9400	142800	19400
2	64300 *	11000 *	216900 *	346500 *
3	2100 *	900	79000 *	665733 *
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	178594 *
14	67100 *	13100 *	270900 *	297598 *
15	6100	100	175800 *	1576635 *
16	22000 *	6400 *	145400 *	20845
17	9300 *	2600 *	85100 *	16160
18	6100	1600	20700	47010
19	13300	6900	265900	157386 *
20	31600	13500	376300	957828
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 *	338300 *	
28	40600 *	10600 *	144400 *	479542
29	11300 *	4300 *	90800 *	29126 *
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
MALTON ONTARIO
TECHNICAL DEPARTMENT (Aircraft)

REPORT No 7/0510/9
SHEET No 15-20
PREPARED BY
DATE Dec. 55
CHECKED BY
DATE

AIRCRAFT C 105

CENTRE FUSELAGE

LOAD No. 20

STRESSES DUE TO 1 KIP 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 *	55700 *
11	1200 *	300 *	66500 *	382326 *
12		3700 *	228500 *	939015 *
13	23900 *	4300 *	68800 *	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	8000	2800	17300 *	
41	6000 *	800 *	13600 *	
42	34600	3900	18300 *	
43	25200 *	4700 *	55700 *	

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

REPORT No.	7/0510/9
SHEET No.	15-21
PREPARED BY	DATE Dec. 55
CHECKED BY	DATE

AIRCRAFT: C 105 — CENTRE FUSELAGE

LOAD No. 21

STRESSES 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	142800 *
10	164900 *	20300 *	360100 *	432600 *
11	9600 *	2300 *	516400 *	2962962 *
12		29600 *	1769400 *	7277116 *
13	184300 *	33100 *	633700 *	616595
14	234100	45200	946100	1039176
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 *	2912200 *	16503936 *
25	261800 *	40700 *	434400 *	2552350
26	288100	59800	1128600	
27	44600 *	9000	1181100	
28	142100	37600	504500	1674171 *
29	39700	14700	317000	101573
30	31700 *	5100 *	82100 *	
31	86500 *	38900 *	1385200 *	
32	199200 *	67900 *	1768000 *	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 *	

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

REPORT No. 7/0519/9
SHEET No. 15-32

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

LOAD No. 22

STRESSES DUE TO 1 KIP [illegible] SIDE AT LOCATION No. 22

STRESS POINT	PART 1	PART 2	PART 3	PART 4
1	149500*	25100*	63500*	935800
2	467300*	60900*	1881600*	5791500*
3	74900	25900	79[illegible]800	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	2556285
23	59200*	18100*	600500*	499167*
24	51700*	30800*	598400*	276885*
25	248400*	13900*	403500	2262252
26	018700*	135200*	3960700*	10000000*
27	222400	55300	1645200	
28	225100	50500	658700	244763
29	11100	500	84000*	3252930
30	61400	19400	298600	
31	56700*	5600*	84400	
32	292500	101000	2926200	
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	235500	
42	426200	40900	339100	
43	332900	84600	2036600	

*

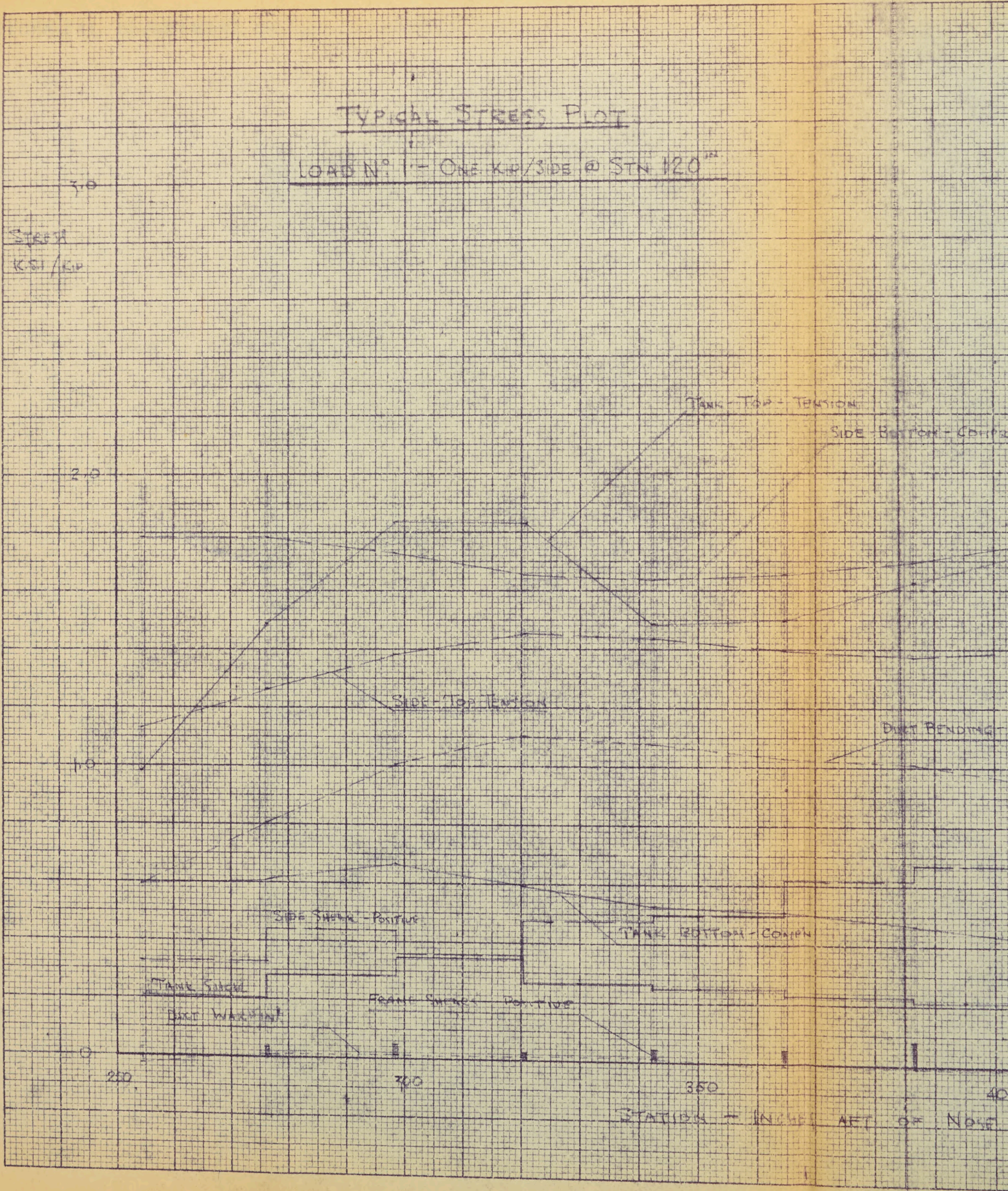
A. V. ROE CANADA LIMITED
MALTON ONTARIO
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO	7 0510 9
SHEET NO	15-23
AIRCRAFT	C 105
	CENTRE FUSELAGE
PREPARED BY	
DATE	Dec. 55
CHECKED BY	
DATE	

LOAD No. 23

STRESSES DUE TO 1 KIP PER SIDE AT LOCATION NO. 23

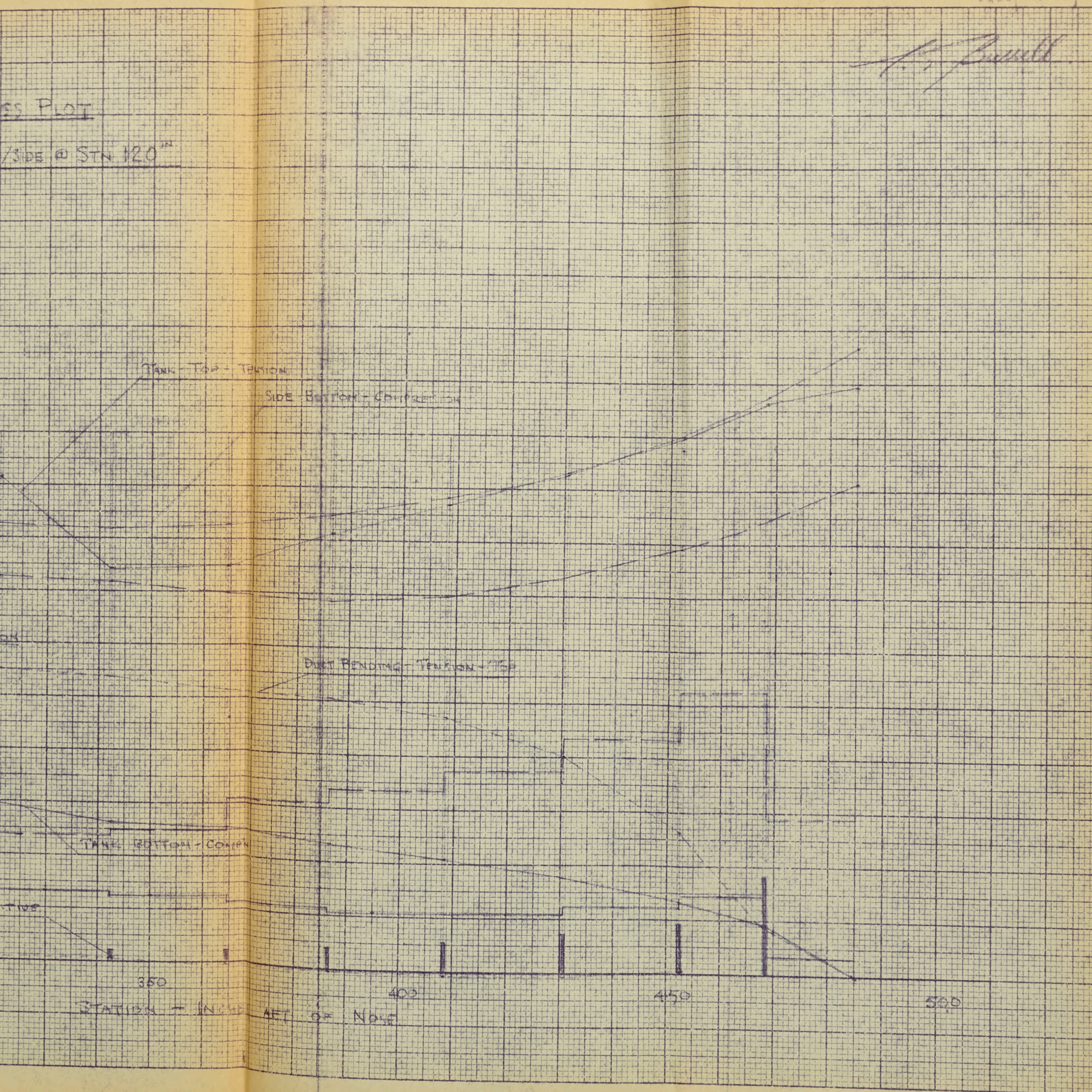
STRESS POINT	PART 1	PART 2	PART 3	PART 4
1	51100*	13500*	207300*	28000*
2	93100	15900	314800	502900
3	8700	1500*	114600	948758
4	9200	9100	187600	156237
5	10600	2600	89600	56500*
6	7800*	1900*	26100*	43600*
7	8500*	6200*	251900*	875536*
8	16200*	13400*	396200*	965561*
9	89500	11100	124000	59400*
10	68600*	8500*	149600*	179800*
11	4100*	900*	214700*	1231270*
12		12300*	735400*	3023786*
13	76900*	13600*	221700*	256293
14	97200	18900	393100	431705
15	8900*	100*	254900	2288000
16	31400	12100	210800	30206*
17	13400	4000	123600	23510*
18	8900*	2200*	30200*	68160*
19	19200*	9900*	386800*	228436
20	45800*	19600*	545800*	1389842*
21	87500	12300	104200	227904*
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	35900*	16300*	575700*	
32	82800*	28100*	734600*	
33	96300	12700	52000	6859396
34	72800*	12600*	184200*	228436
35	500	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*	



TYPICAL STRESS PLOT

LOAD N° 1 – ONE KIP/SIDE @ STN 120

7/0510/9
Sheet 15-24



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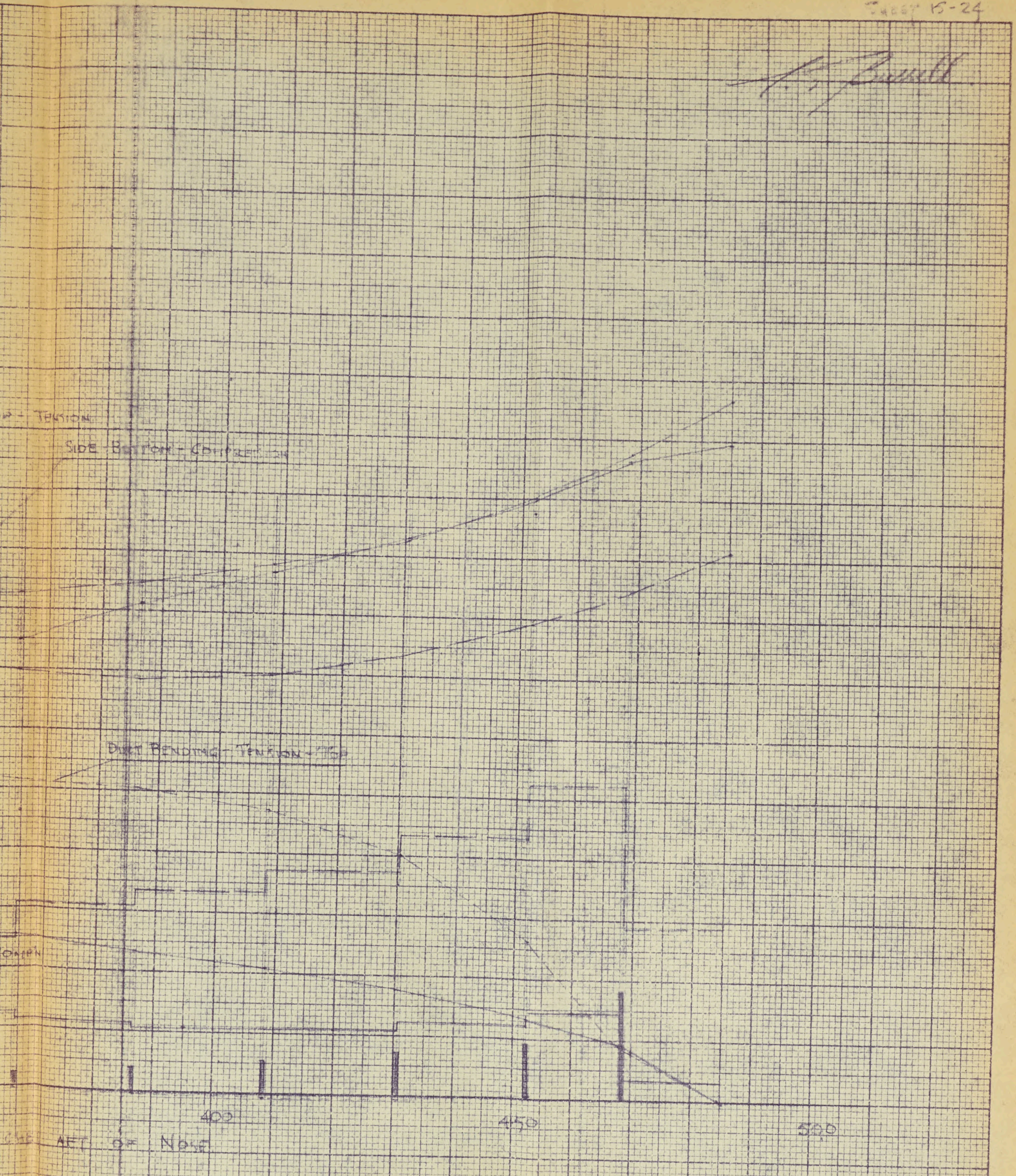
Sheet 15-24



A. V. ROE CANADA LIMITED
MALTON ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO	7/0510/9
SHEET NO	16-1
PREPARED BY	
DATE	Dec. 55
CHECKED BY	
DATE	

AIRCRAFT C 105

CENTRE FUSELAGE

DEFLECTIONS

DUE TO UNIT LOADS

INCHES PER KIP PER SIDE

(EIGHT PLACES OF DECIMALS)

AT LOAD POINT	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	20743967	12587361	11429716	9868911	8362357	6932886
4	17354466	10814442	9869094	8929216	7602005	6342139
5	14290956	9131718	8362528	7602283	6888984	5788796
6	11509706	7532574	6933150	6342028	5788805	5196540
7	9001790	6031683	5579213	5134864	4717382	4280412
8	6766835	4645851	4320996	4001514	3701726	3391976
9	4803512	3388413	3170796	2957771	2757507	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*	568390*	285150*	305700*

DEFLECTIONS

A. V. ROE CANADA LIMITED
MALTON ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7 0510/9
SHEET NO. 16-2

AIRCRAFT C 105

CENTRE FUSELAGE

PREPARED BY
DATE Dec. 55
CHECKED BY
DATE

DEFLECTIONS

DUE TO UNIT LOADS

INCHES PER KIP PER SIDE

EIGHT PLACES OF DECIMAL

AT LOAD POINT	LOAD 7	LOAD 8	LOAD 9	LOAD 10	LOAD 11	LOAD 1
1	9001230	6766207	4803214	3112983	1685806	790080
2	6031124	4645699	3388373	2272642	1304908	687560
3	5579068	4320549	3170750	2143677	1247428	673700
4	5134504	40017[illegible]5	2957718	2017790	1192419	661740
5	4717228	3701581	2757624	1900853	1142729	653580
6	4280317	3391971	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	1054751	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

A V ROE CANADA LIMITED
MALTON ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO	7 0510/9
SHEET NO	16-3
AIRCRAFT	C 105
	CENTRE FUSELAGE
PREPARED BY	
DATE	Dec. 55
CHECKED BY	
DATE	

DEFLECTIONS

DUE TO UNIT LOADS

INCHES PER KIP PER SIDE

(EIGHT PLACES OF DECIMAL)

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	12587361	9598431	6812392	4202978	1956986	366570
4	10814442	8402016	6109737	3838128	1815152	347120
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	250890
10	2272798	1976257	1657028	1274880	752519	198620
11	1305474	1168165	1014946	820537	531306	159370
12	687800	646360	593540	512110	364680	123230
13	14359819	10760011	7495932	4559527	2095898	385420
14	10760302	8768453	6266537	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

A. V. ROE CANADA LIMITED
MALTON ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT C 105	CENTRE FUSELAGE	REPORT NO 7/0510/9	SHEET NO 16-4
		PREPARED BY	DATE Dec. 55
		CHECKED BY	DATE

DEFLECTION

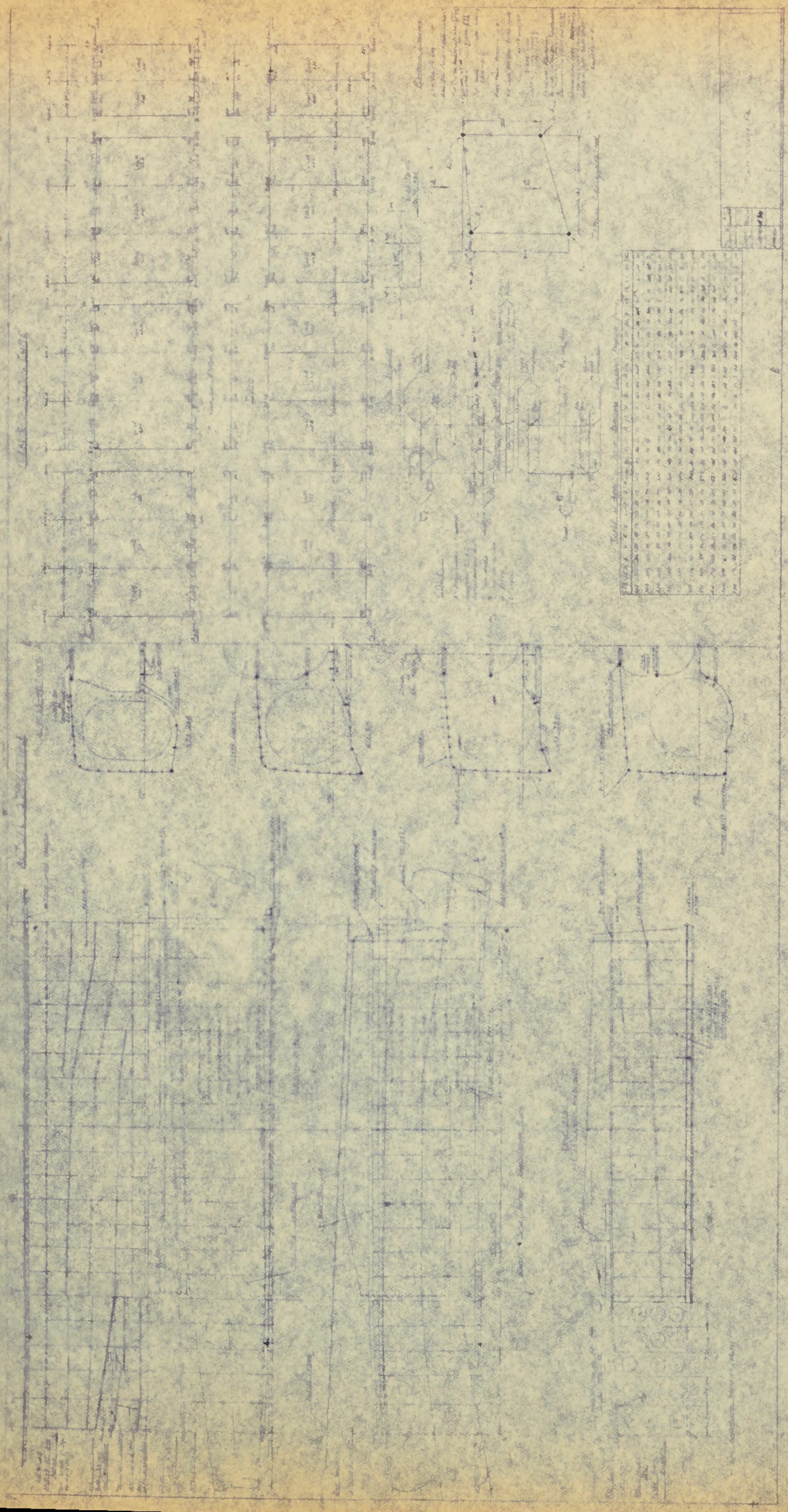
DUE TO UNIT LOADS

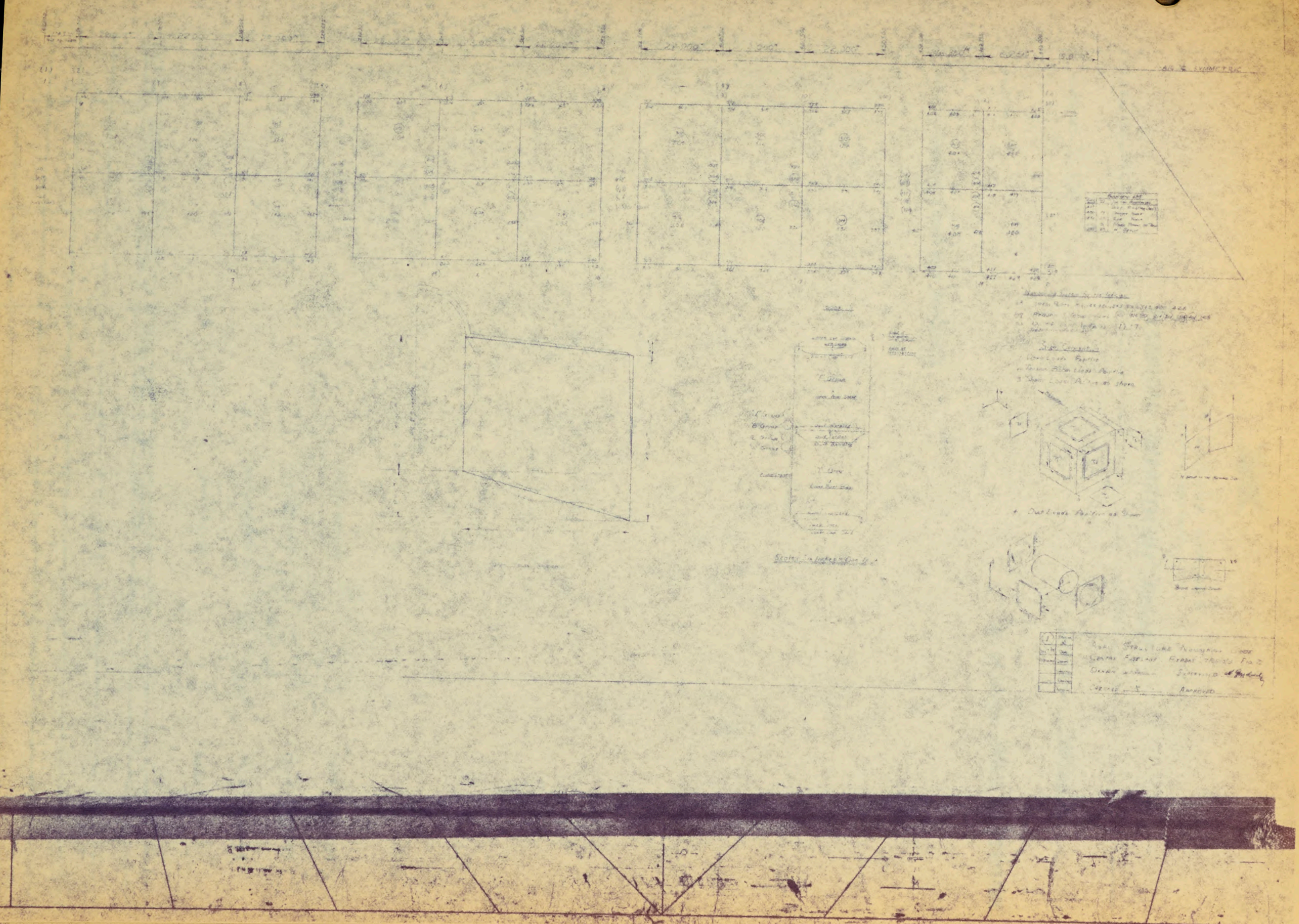
INCHES PER KIP PER SIDE

(EIGHT PLACES OF DECIMAL)

AT LOAD POINT	LOAD 19	LOAD 20	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*	736610*	579940*	305710*
7	228450	102880*	797660*	512290*	331360*
8	251610	113380*	878710*	440710*	365120*
9	283060	127590*	988560*	[illegible]62250*	410680*
10	326310	147100*	1140010*	[illegible]470*	473530*
11	386460	174100*	1349460*	[illegible]0*	560790*
12	449530	202530*	1570040*	[illegible]00*	652540*
13	166400	75040*	580900*	[illegible]010*	241480*
14	180200	81160*	629200*	[illegible]580*	261390*
15	188800	85070*	659510*	[illegible]640*	274110*
16	185210	83480*	646510*	[illegible]4010*	268720*
17	151110	68100*	527910*	[illegible]9770*	219430*
18	65880	29560*	230020*	31780*	95460*
19	868110	307900	1279820*	78860	559750*
20	307910	560270	675200	35620*	252340
21	1279820*	675210	11008210	275450*	1954820
22	78670	35430*	275460*	1121510	114550*
23	559740*	252340	1954810	114350*	812200

DEFLECTIONS




Structure
Drawn
Supervised
Checked
Approved

A. V. ROE CANADA LIMITED
MALTON · ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C 105

REPORT NO: 7/0510/9

FILE NO:

NO. OF SHEETS: 206 + 10 Drawings

TITLE: Centre Fuselage Analysis

PREPARED BY C. Burrell DATE Dec'55

CHECKED BY DATE Dec'55

SUPERVISED BY DATE Dec'55

APPROVED BY DATE

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

FORM 1316A

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No	7/0510/9
SHEET No	1-1
AIRCRAFT:	C105
	Centre Fuselage
PREPARED BY	C.B.
DATE	May 5.55
CHECKED BY	
DATE	

INDEX

FORM 1318A

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No.
SHEET No. 2-1
AIRCRAFT: 105
Analysis - Nose & Centre Section
PREPARED BY: C.B.
DATE: March '55
CHECKED BY:
DATE:

Centre Section - Drawings

7-4354-1	G.A. (Interchangeability)	Sht 1A.
7 0154-02	Skinning	
7-0152-37	Former 255	Shts 1, 2 & 3 Iss. 4.
7-1054-15	Former 315	Shts 1, 2, 3 & 4 Iss "A"
7-1054-23	Former 392	Shts 1, 2, 3 & 4 Iss "A"
7-1054-31.	Former 469	Shts 1, 2, 3 & 4 Iss "A"
7-0154-08	Tank	Shts 1 to 8
7-0154- 03-04 05-06	Duct Rear Sht 1 Fwd Sht 2	
7-0154-0	Armament Pack.	[illegible] P/L Hori[illegible] 295.93 V L 337 [illegible] 482.5 V L FS A 425 [illegible]
7-0154-06	Top Outer Longn	
" -07	Lower Longn	
7-0154-20	Armament Bay Roof	
7-0154-36	Top Inner Longl'n 255-315	

FORM 1319A

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No	7/0510/9
SHEET No	2-2
AIRCRAFT:	C105
	CENTRE FUSELAGE
PREPARED BY	CB
DATE	JULY 55
CHECKED BY	
DATE	

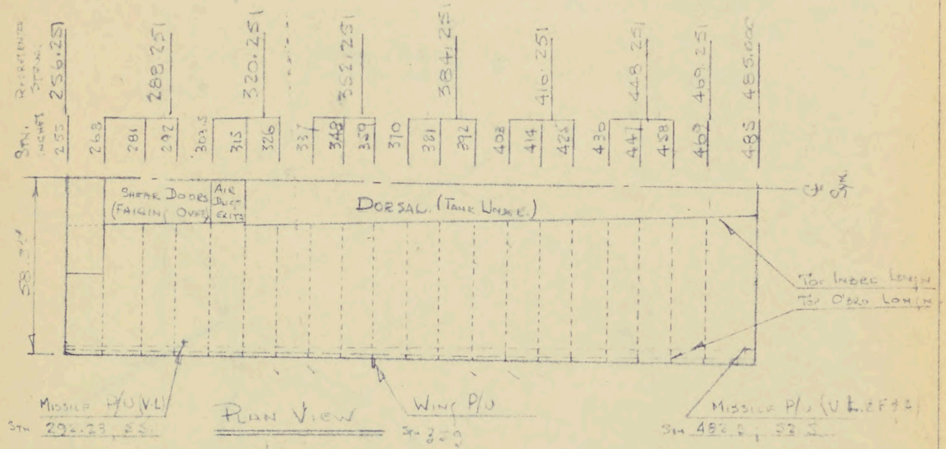
REFERENCES

	PAGE.
AVRO REPORT GEN/1090/336	3-1
FRONT-CENTRE SECTION GROUP AVRO REPORT No 7/0554/5	8-2
NACA ARR. L5B10	8-6
AVRO REPORT GEN/1090/311	8-29
TIMOSHENKO THEORY OF PLATES & SHELLS	9-2
AVRO REPORT GEN/1090/330	10-1 10-2
AVRO REPORT 7/0556/27	12-1

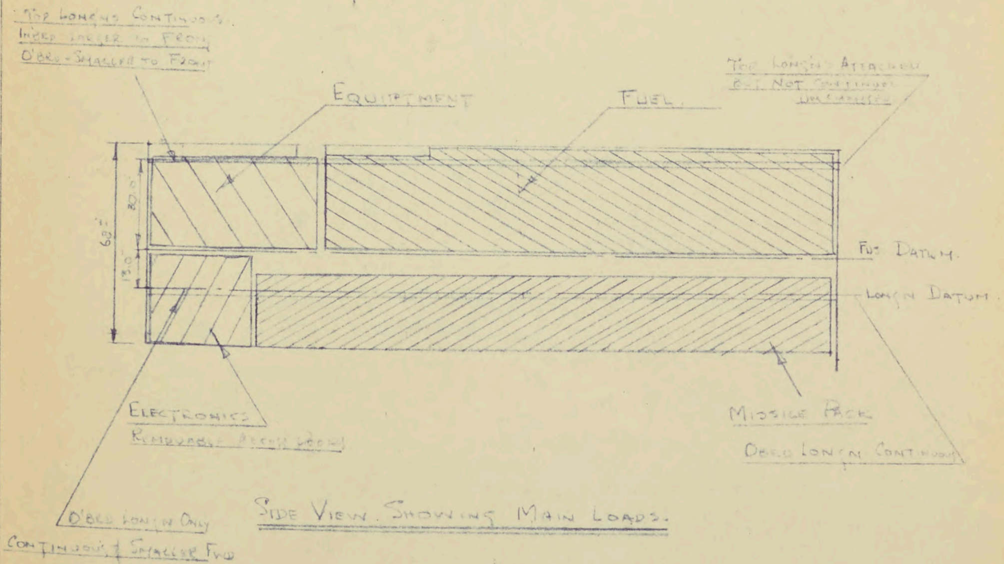
AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT		REPORT NO.	7/05101/3
105	c/s.	SHEET NO.	4-1
		PREPARED BY	DATE
		C.B.	Apr. 57
		CHECKED BY	DATE

Layout (1/50 Scale)



Plan View



Side View. Showing Main Loads.

1. Equip't & Fuel Inbrd of Ducts
2. Electronics & Pack across Full width
3. Pack Picks up at Four Points as shown by Plan View.

Ref. 7.4354-1 Sht 1 Iss A
D. Shone.
D. Wade.

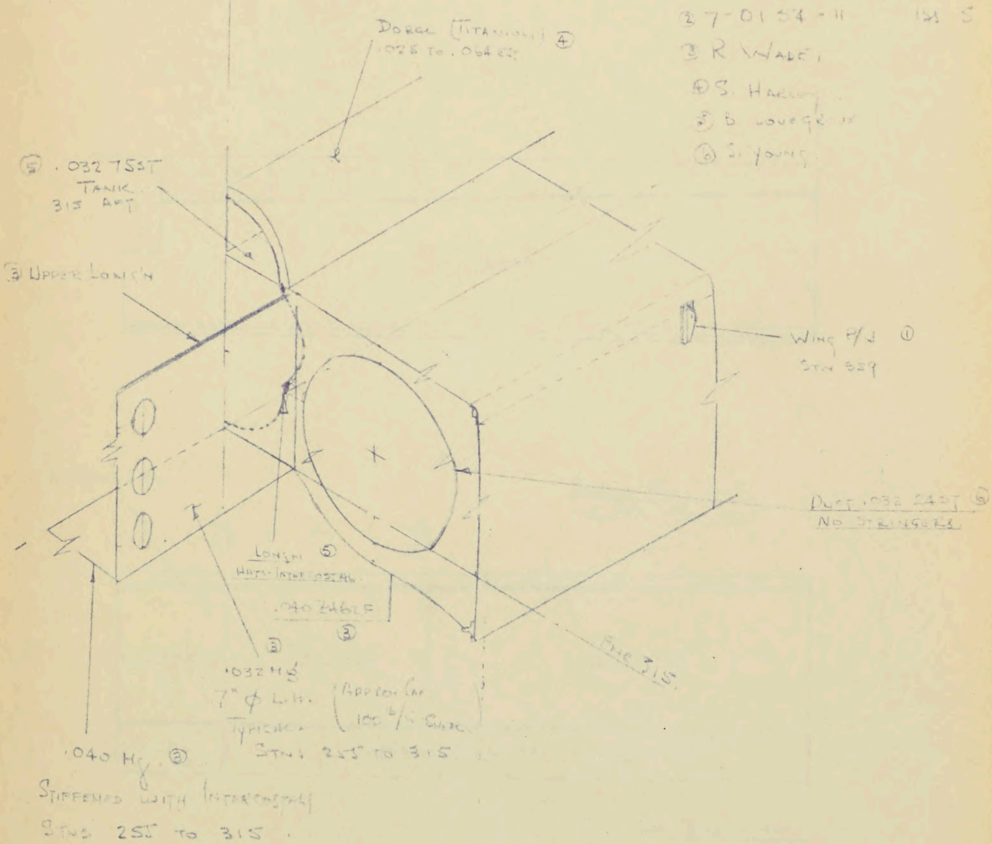
FORM 1319A

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO	7/0510/9		
SHEET NO	42		
AIRCRAFT	C105	C/S	
PREPARED BY	CB	DATE	APR 13 55
CHECKED BY		DATE	

LAYOUT. SCALE 1 30^IN = 1^IN

REF. ① 7-4354-1 SH 1 ISS. A.
② 7-0154-11 ISS 5
③ R WADE.
④ S. HARVEY
⑤ B LOVEGROVE
⑥ S. YOUNG



NOTES: 1. INTERCOSTALS BELOW TANK DO NOT LINE UP WITH ③
INTERCOSTALS BELOW EQ'PT BAY PANEL EITHER VERTICAL OR OBL.

2. EQ'PT BAY PANEL (.040 HG) STOPS @ STN 255 ⑤

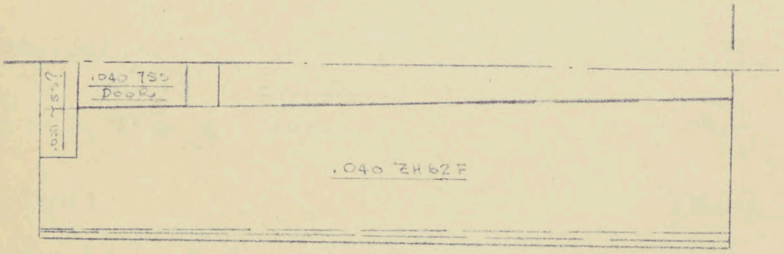
3. TANK HAS INTERCOSTAL STIFFENING STNS 315 TO 326 & STN 461 TO 482 ⑤

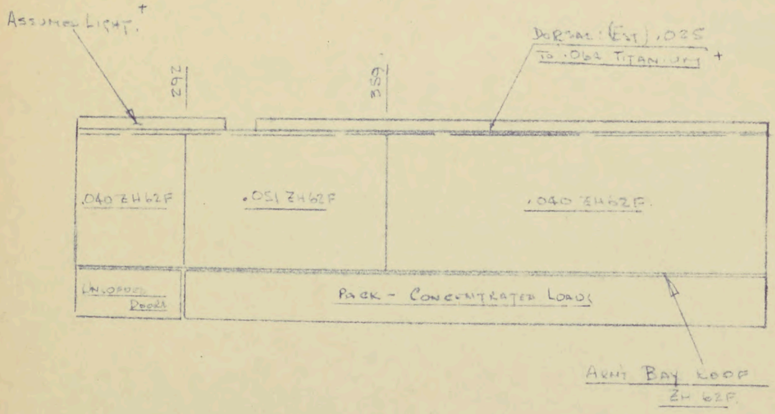
AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: 105	C/S	REPORT No 7/0510/9	SHEET No 4.3
		PREPARED BY: C.B.	DATE: Apr 13.55
		CHECKED BY:	DATE:

Skinning Layout (Approx 1/50 Scale)





Ref: R. Wade.
S. Harley +

FORM 1319A

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No 7/0510/9
SHEET No 4.4

AIRCRAFT		PREPARED BY	DATE
C105	Centre Fuselage	C.B.	Apr. 13 51
		CHECKED BY	DATE

Approximate Weights.

1. Structure	1 600 lb
2. Services	1 000
3. Missile Pack & Services	2000
4. Missiles	1000
5. Fuel 558 Gal @ 7.85 lb/G	4 400
	10,000 lb

Ref. E. Burnett

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

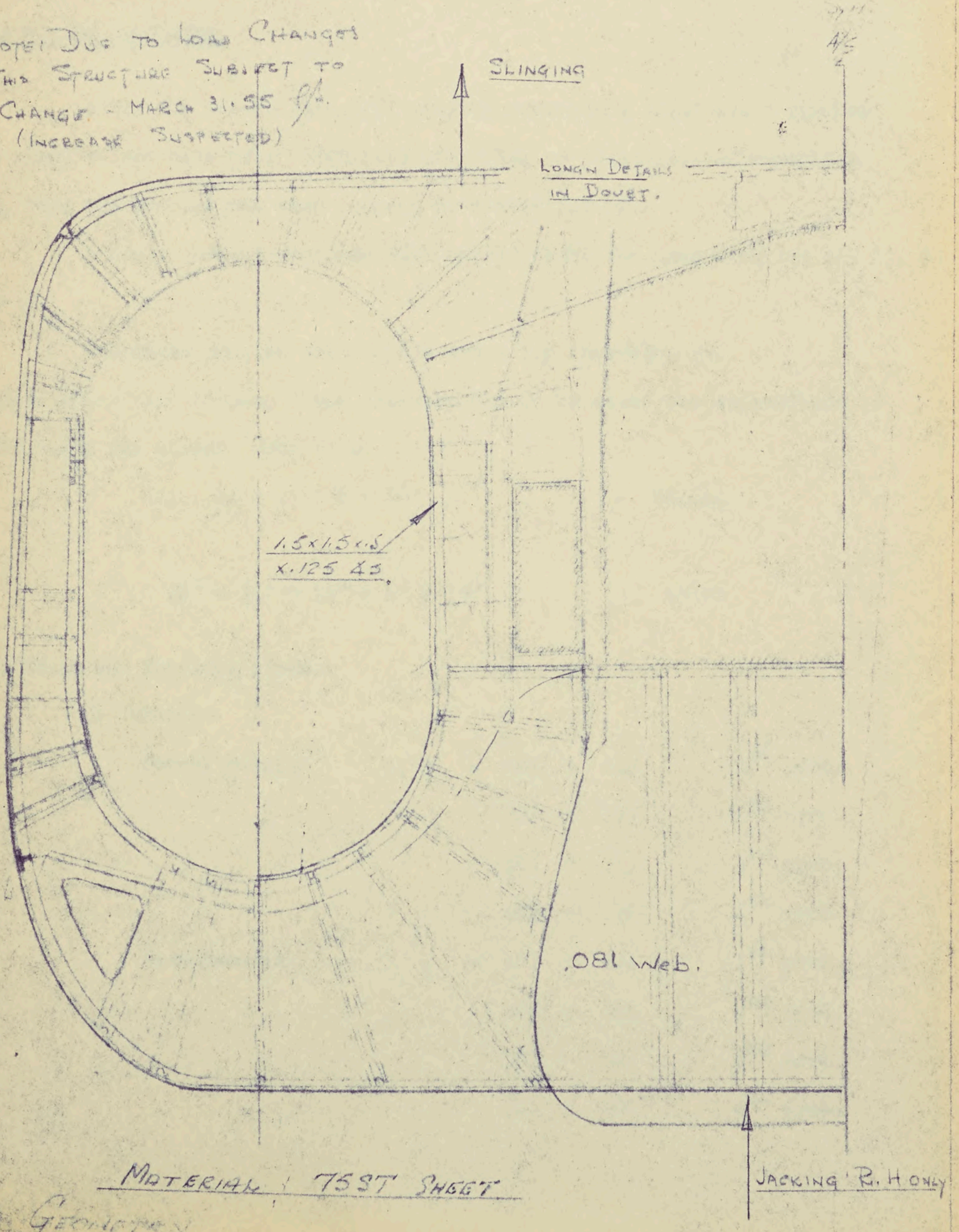
REPORT NO. 7/0510/9

SHEET NO. 4-5

AIRCRAFT		PREPARED BY	DATE
105	[illegible]		Feb '55
		CHECKED BY	DATE

Drg 7: 0152 · 0037 Shts 1, 2 & 3.

Note: Due to Load Changes
This Structure Subject to
Change - March 31.55
(Increase Suspected)



Material: 75ST Sheet

Geoffrey

Approx Scale 1/5

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

Loads a, b = 1, 2,..........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.

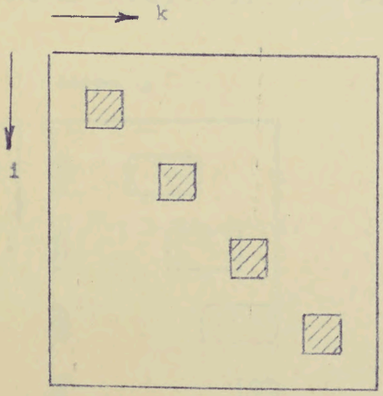
In the general problem

Loads a, b = 1, 2,..........,23

Reactions u, v = 1, 2,..................20.

2) Matrices given:

Overall Elastic Influence Coefficients C_{ik}



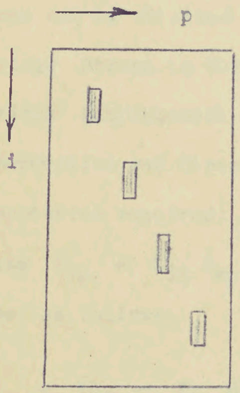
426 x 426

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

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

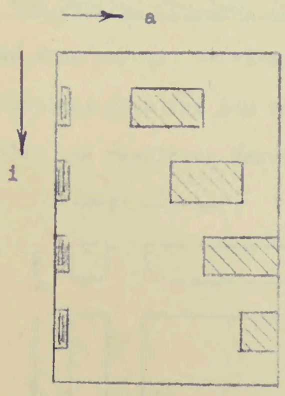
AIRCRAFT:	Centre Fuselage Analysis.	REPORT NO 7/0510/9	
		SHEET NO 5-3	
		PREPARED BY	DATE
		A.Grze.	
		CHECKED BY	DATE

Overall Stress to Redundant Load Relationship



426 x 406

Overall Stress to Unit Load Realtionship (partial problems)



426 x 120

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

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:

$$S_{ia} = (\delta_{ik} - K_{ip} D^{-1}_{pq} H_{qk}) T_{ka}$$

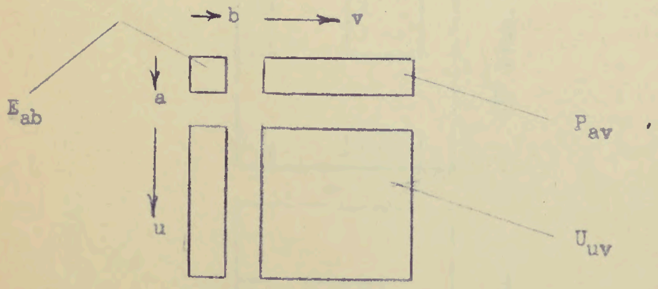
(print results)

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

C) Solution of the General Problem. Displacement Matrix.

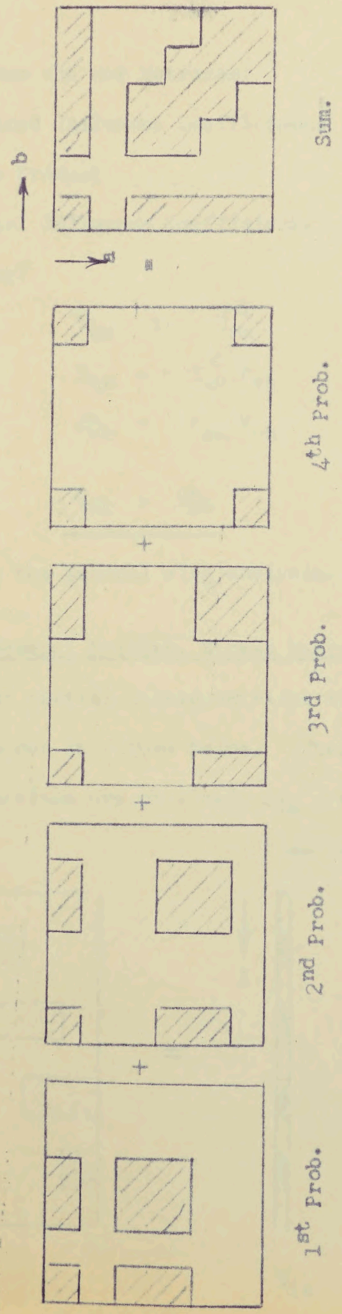
1) Add four partial Matrices Z_{ab} as shown on the next page.

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,.......120 has to be replaced by the range of u, v = 1, 2, 3,.......20. The range of a, b = 1, 2,...23 remains the same, however.

AIRCRAFT:	Centre Fuselage Analysis.	PREPARED BY	DATE
		A. Grze.	
		CHECKED BY	DATE



Sum.

4th Prob.

3rd Prob.

2nd Prob.

1st Prob.

Summation of partial Z_{ab} Matrices.

2) Consider the new Matrices:

Combined Influence Coefficients	U_{uv}
Cross Product	P_{ua}
Initial Influence Coefficients	E_{ab}

Compute the following:

a) Invert U_{uv} to U_{vu}^{-1}

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

c) Multiply $E^*_{ab} = P_{au} R_{ub}$ $(= P_{au} U_{uv}^{-1} P_{vb})$

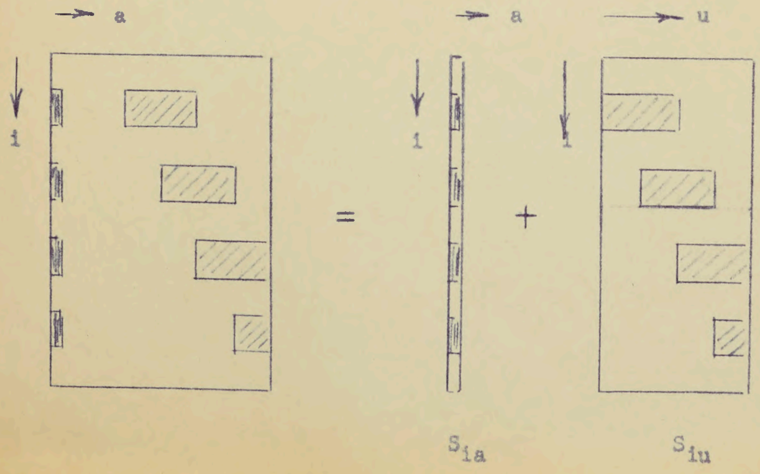
d) Add

$\underline{E_{ab} + E^*_{ab}}$ (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}.



2) Then compute the following

$$\underline{S_{ia} + S_{iu} R_{ua}} \qquad \text{(print)}$$

This is the final Stress of Unit Load Matrix. The last computation can be performed as four separate operations, and in each one only the indices are used:

u, v = 1,.......10 1st
6,.........15 2nd
11,............20 3rd
16,....20. 4th

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

AIRCRAFT: C105	CENTRE FUSELAGE	REPORT No. 7/0510-9	
		SHEET No. 5-9	
		PREPARED BY C.B.	DATE Oct '55
		CHECKED BY	DATE

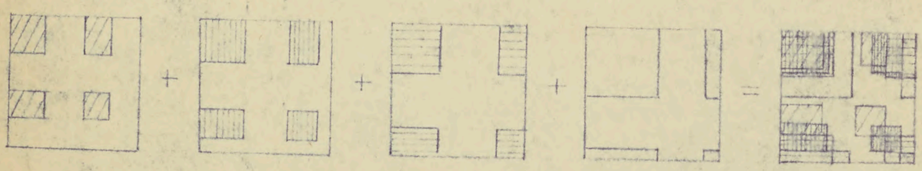
DETAIL PROCEDURE FOR COMBINING PROBLEMS.

1. RE-INDEX LOAD NUMBERS IN Zab MATRICES AS FOLLOWS:

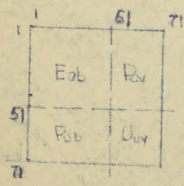
IN PROBLEM	COMPUTER N°	21	22	23	24	25	26	27	28	29	30	31
1	RE INDEX TO	51	52	53	54	55	56	57	58	59	60	72
2		56	57	58	59	60	61	62	63	64	65	72
3		61	62	63	64	65	66	67	68	69	70	72
4					66	67	68	69	70	72		

2. ADD

$$Zab1 + Zab2 + Zab3 + Zab4.$$



3. SPLIT TOTAL MATRIX AS



1. By First Index Into Two Groups
   a - First Index 1 to 23 Inclusive.
   b. First Index 51 to 70 Inclusive.
2. Split the above two Groups by the second Index
   c. Second Index 1 to 23 Inclusive
   d. Second Index 51 to 70 Inclusive

Giving 4 Matrices with Indices

	First Indices	Second Indices
Eab	1 to 23 Inc	1 to 23 Inc.
Pav	1 to 23 Inc.	51 to 70 Inc.
Pub	51 to 70 Inc	1 to 23 Inc.
Uuy	51 to 70 Inc.	51 to 70 Inc.

FORM 1319A

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

AIRCRAFT: C105	CENTRE FUSELAGE	REPORT NO. 7/0510/9	
		SHEET NO. 5·10	
		PREPARED BY	DATE
		C.G.	Oct. 55
		CHECKED BY	DATE

COMBINING PROBLEMS

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. INVERT U_{uv} TO U_{vu}^{-1} (20 by 20)

NOTE: ROW & COLUMN No. 55 ALL ZERO ELEMENTS

6. MULTIPLY $R_{va} = -U_{uv}^{-1}\,P_{va}$

$\left|\quad\right| \times \left|\quad\right|$ (20×20) × (20×23) = (20×23)

Note change of sign.

7. MULTIPLY $E^{*}_{ab} = P_{au}\cdot R_{ub} = -P_{au}\cdot U_{uv}^{-1}\cdot P_{vb}$. (SYMMETRIC)

(23×20) × (20×23) = (23×23)

8. ADD. $E^{w}_{ab} = E_{ab} + E^{*}_{ab}$ (SYMMETRIC – PRINT)

9. RE-INDEX LOAD NUMBERS IN S_{ia} MATRICES

AS IN STEP 1

10. SPLIT BY SECOND INDEX ONLY
INTO SECOND INDEX 1 TO 23 INCLUSIVE
SECOND INDEX 51 TO 70 INCLUSIVE $\left|S_{ia}\right|\left|S_{iu}\right|$

FORM 1319A

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

REPORT NO. 7/0510/9
SHEET NO. 5-11

AIRCRAFT:		PREPARED BY	DATE
C105	CENTRE FUSELAGE	C.B.	OCT '55
		CHECKED BY	DATE

COMBINING PROBLEMS.

11. CHECK SUMS.

ADD ELEMENTS FOR EACH OF THE FOUR S_{ia} MATRICES (BY ROWS ONLY) TO GIVE CHECK SUMS & PLACE AS FOLLOWS

SUM OF COL'NS 1 TO 23 INC. = COL'N 24
SUM OF COL'NS 51 TO 70 INC. = COL'N 71

ADD COL'N 72

12. MULTIPLY $S_{iu} \cdot R_{ua} = \Lambda_{ia}$

FOR EACH SUB-PROBLEM.

13 ADD $S_{ia} + \Lambda_{ia}$

FOR EACH SUB-PROBLEM.

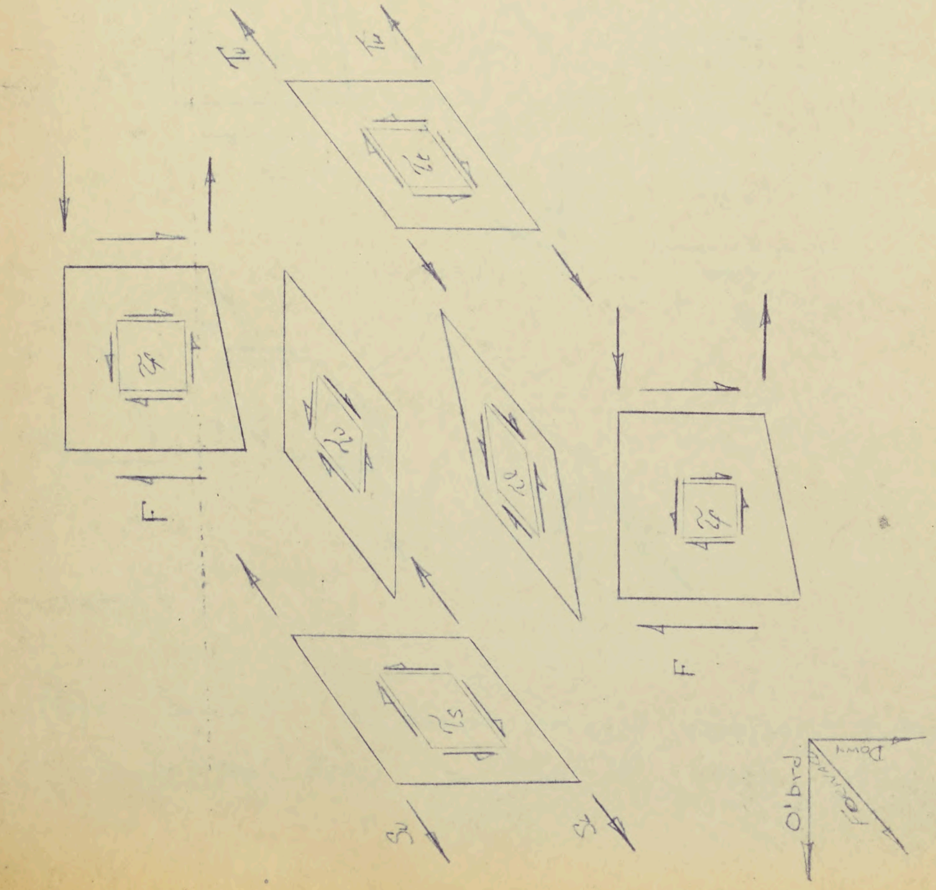
FORM 1319A

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C105	Centre Fuselage	REPORT No. 7/0.510/9	
		SHEET No. 6-1	
		PREPARED BY	DATE
		C.B.	May 3. 55
		CHECKED BY	DATE

Sign Convention: Outer Box.

Positive as shown



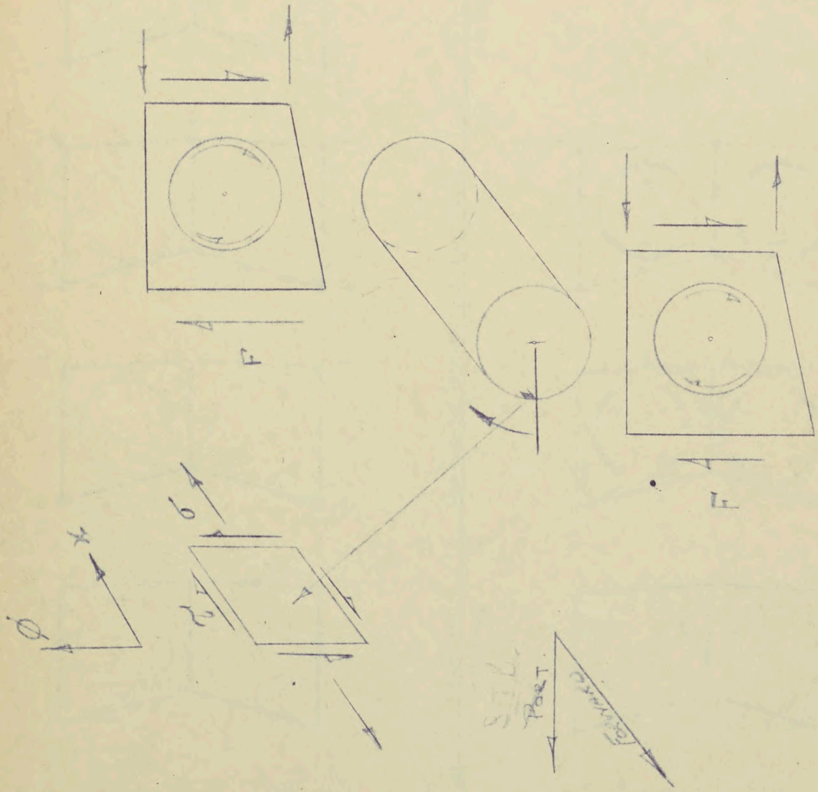
FORM 1118A

AIRCRAFT: C105	Centre Fuselage	PREPARED BY CB	DATE May 3 55
		CHECKED BY	DATE

REPORT No 7/0510/9
SHEET No 6-2

Sign Convention - Duct.

Positive as Shown.



Note: Shears on the frames produce equivalent energy in the frame to the "F" loads.

FORM 1319 A

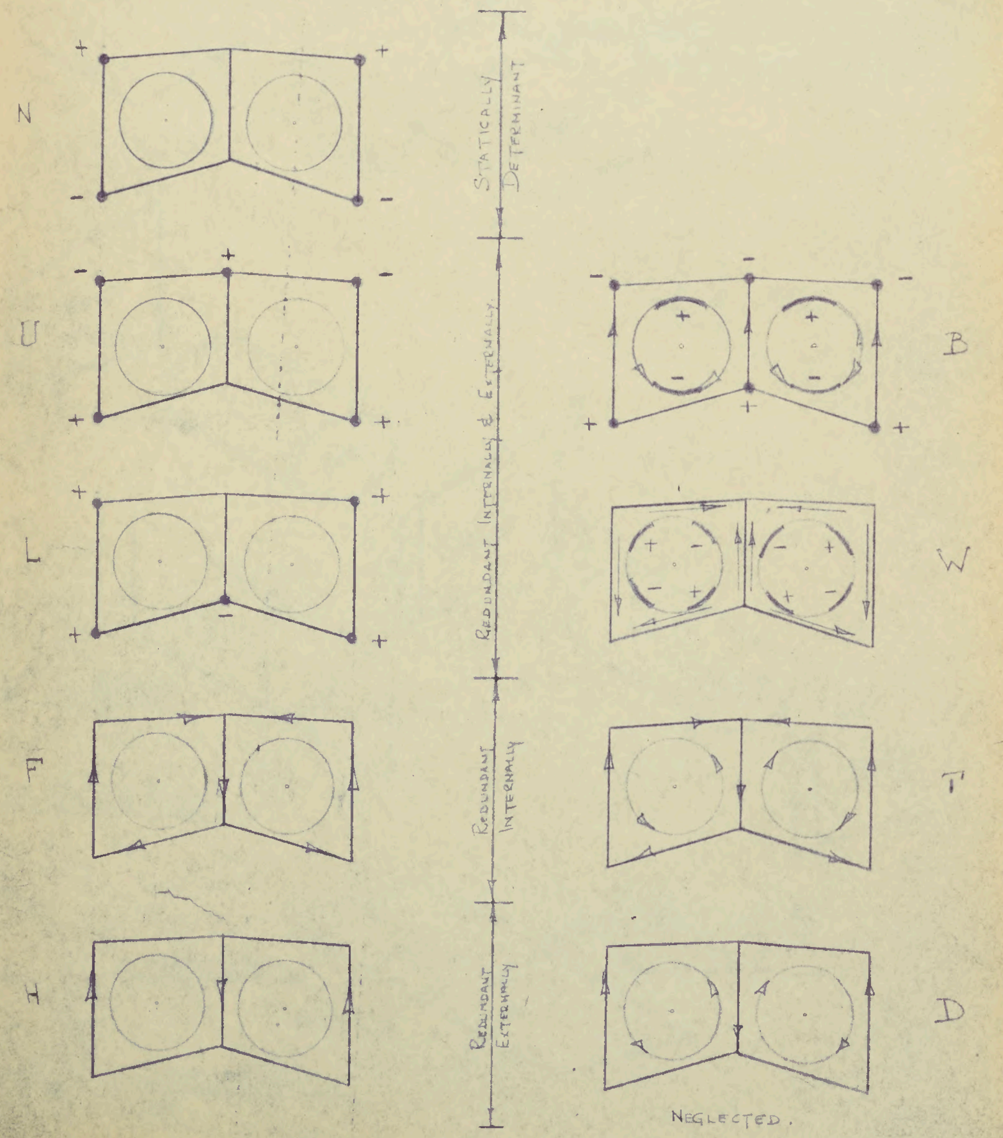
AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO 7/0510/9
SHEET NO 7-1

AIRCRAFT C105	CENTRE FUSELAGE	PREPARED BY C.B.	DATE May 4.55.
		CHECKED BY	DATE

REDUNDANT GROUPS.

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

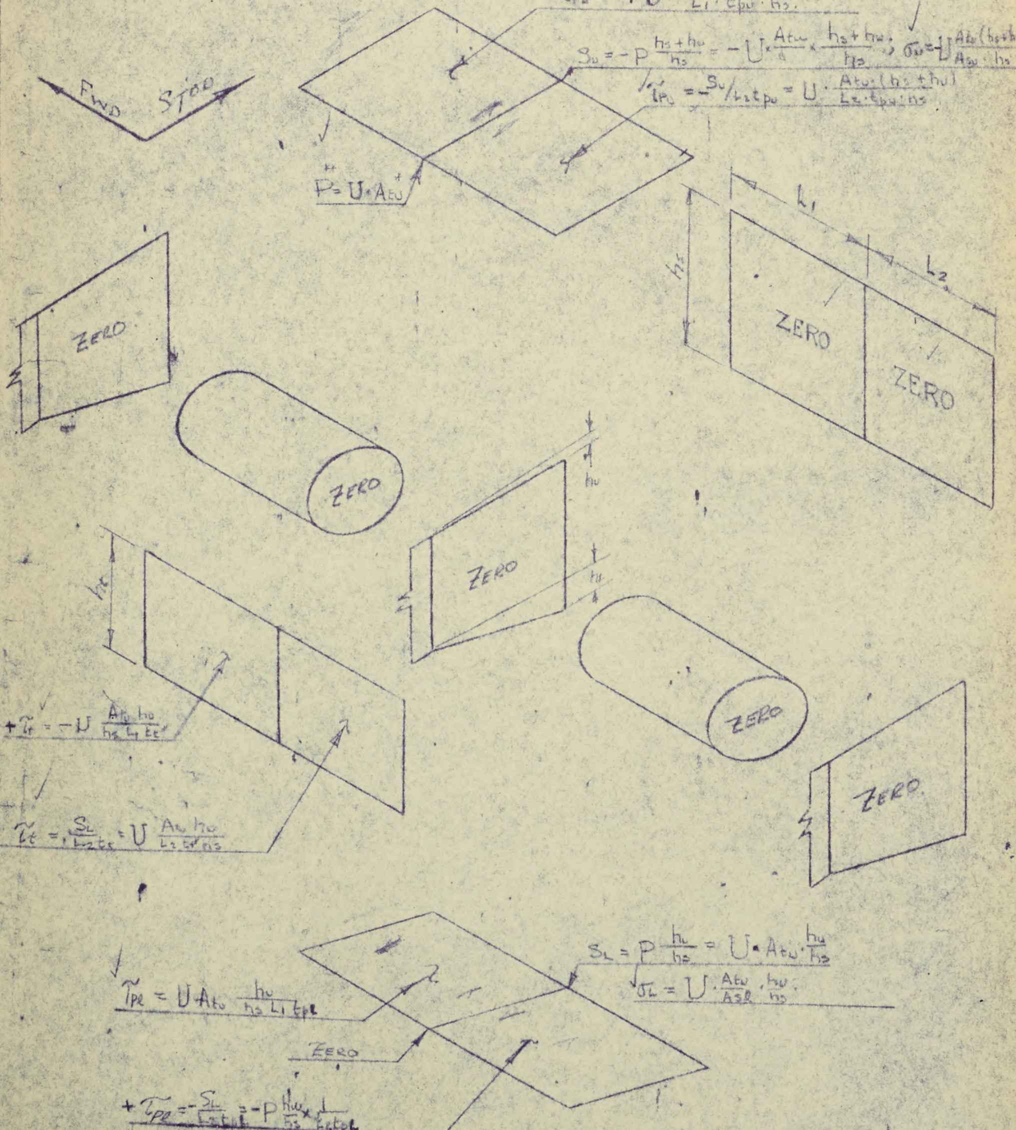


AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT C105	CENTRE FUSELAGE	REPORT NO 7/0510/9	SHEET NO 7-2
		PREPARED BY C.B.	DATE May 5/55
		CHECKED BY	DATE

"U" GROUP,

$-\tau_{pu} \cdot t_{pu} + \tau_t \cdot t_t = \mp (S_U + S_L)/L = +P/L$



$-\tau_{pu} = +U \cdot \frac{A_{tu}(h_s+h_u)}{L_1 \cdot t_{pu} \cdot h_s}$

$S_U = -P\frac{h_s+h_u}{h_s} = -U \times \frac{A_{tu}}{*} \times \frac{h_s+h_u}{h_s}$; $\sigma_U = -U\frac{A_{tu}(h_s+h_u)}{A_{su} \cdot h_s}$

$\tau_{pu} = -S_U/L_2 t_{pu} = U \cdot \frac{A_{tu}(h_s+h_u)}{L_2 \cdot t_{pu} \cdot h_s}$

$\overset{**}{P} = U \cdot \overset{*}{A_{tu}}$

$+\tau_t = -U\frac{A_{tu} h_u}{h_s L_1 t_t}$

$\tau_t = \frac{S_L}{L_2 t_t} = U\frac{A_{tu} h_u}{L_2 t_t h_s}$

$S_L = P\frac{h_u}{h_s} = U \cdot A_{tu} \cdot \frac{h_u}{h_s}$

$\sigma_L = U \cdot \frac{A_{tu}}{A_{sL}} \cdot \frac{h_u}{h_s}$

$\tau_{pL} = U A_{tu} \frac{h_u}{h_s L_1 t_{pL}}$

ZERO

$+\tau_{pL} = -\frac{S_L}{L_2 t_{pL}} = -P\frac{h_u}{h_s} \times \frac{1}{L_2 t_{pL}}$

$= -U A_{tu} \frac{h_u}{h_s L_2 t_{pL}}$

* ½ of the actual lumped area or thickness, ie for ½ of the aeroplane

** force on ½ of aeroplane.

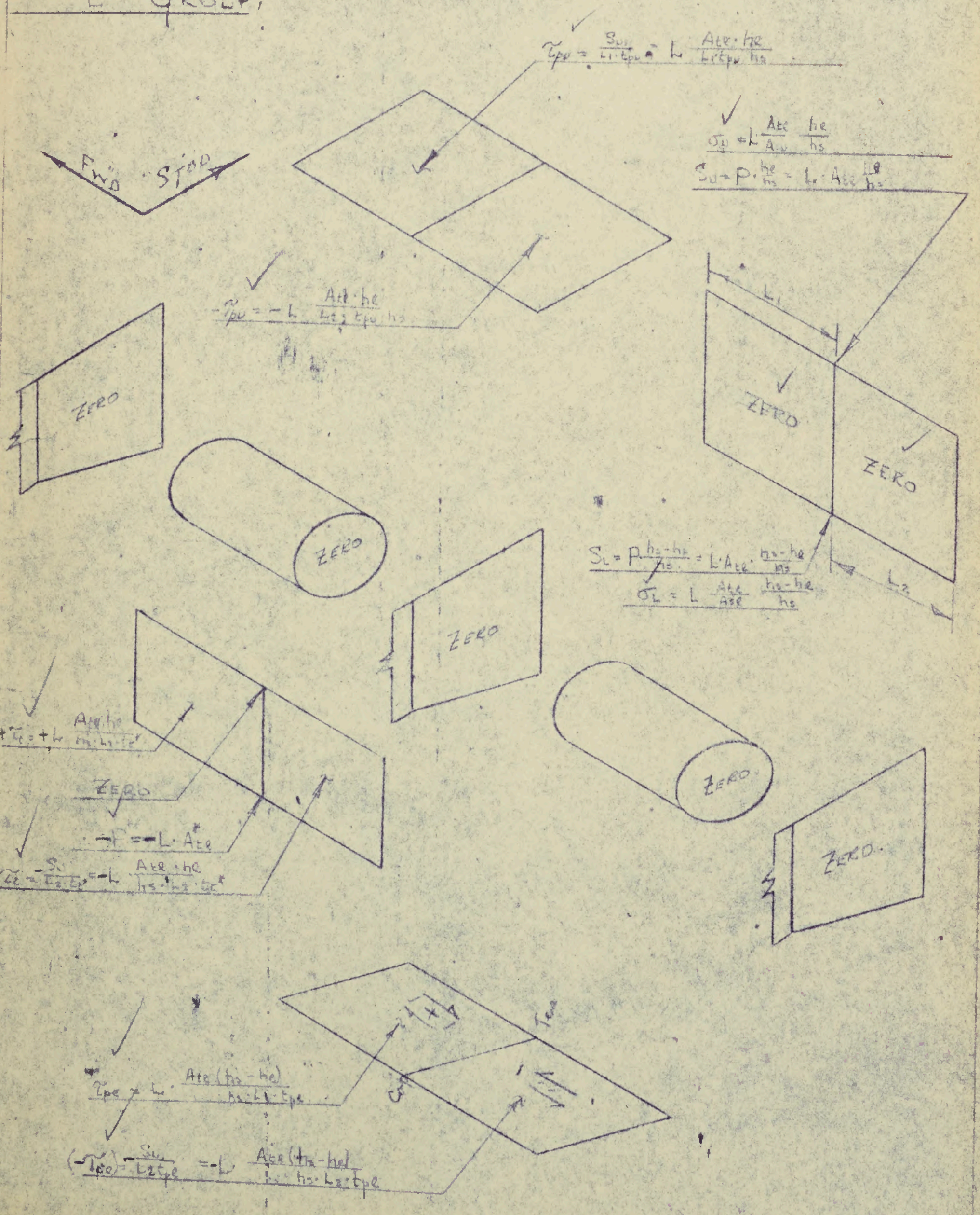
AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO 7/0510/9
SHEET NO 7-3

AIRCRAFT C105
CENTRE FUSELAGE

PREPARED BY C.B.
DATE May. 5. 55
CHECKED BY
DATE

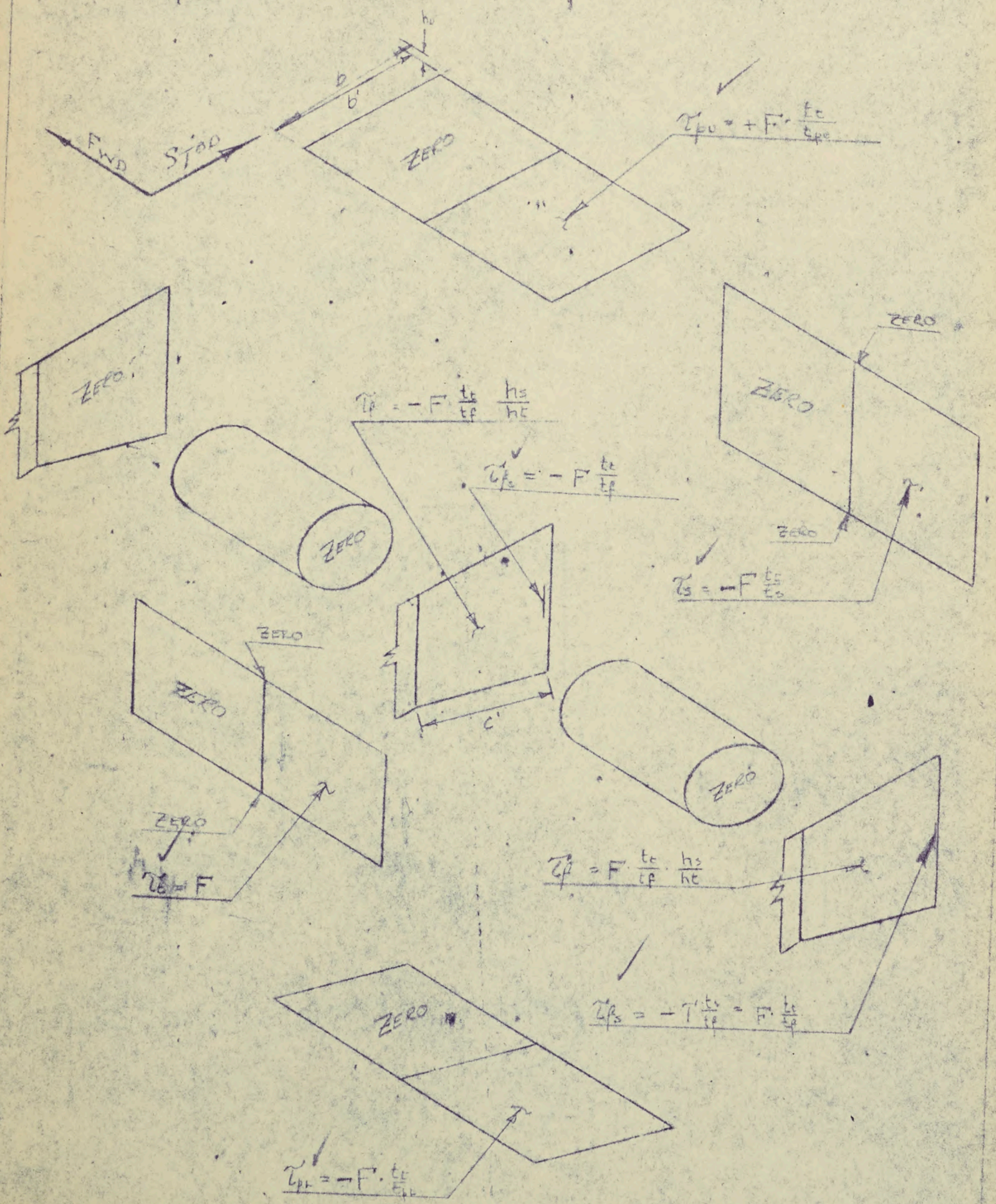
"L" GROUP.



$$\tau_{pu} \cdot t_{pu} + \tau_t \cdot t_t = \pm (S_u + S_L)/L = P/L$$

AIRCRAFT		REPORT No. 7/0510/9	
		SHEET No. 7-4	
C105	CENTRE FUSELAGE	PREPARED BY	DATE
		C.B.	May 5.55
		CHECKED BY	DATE

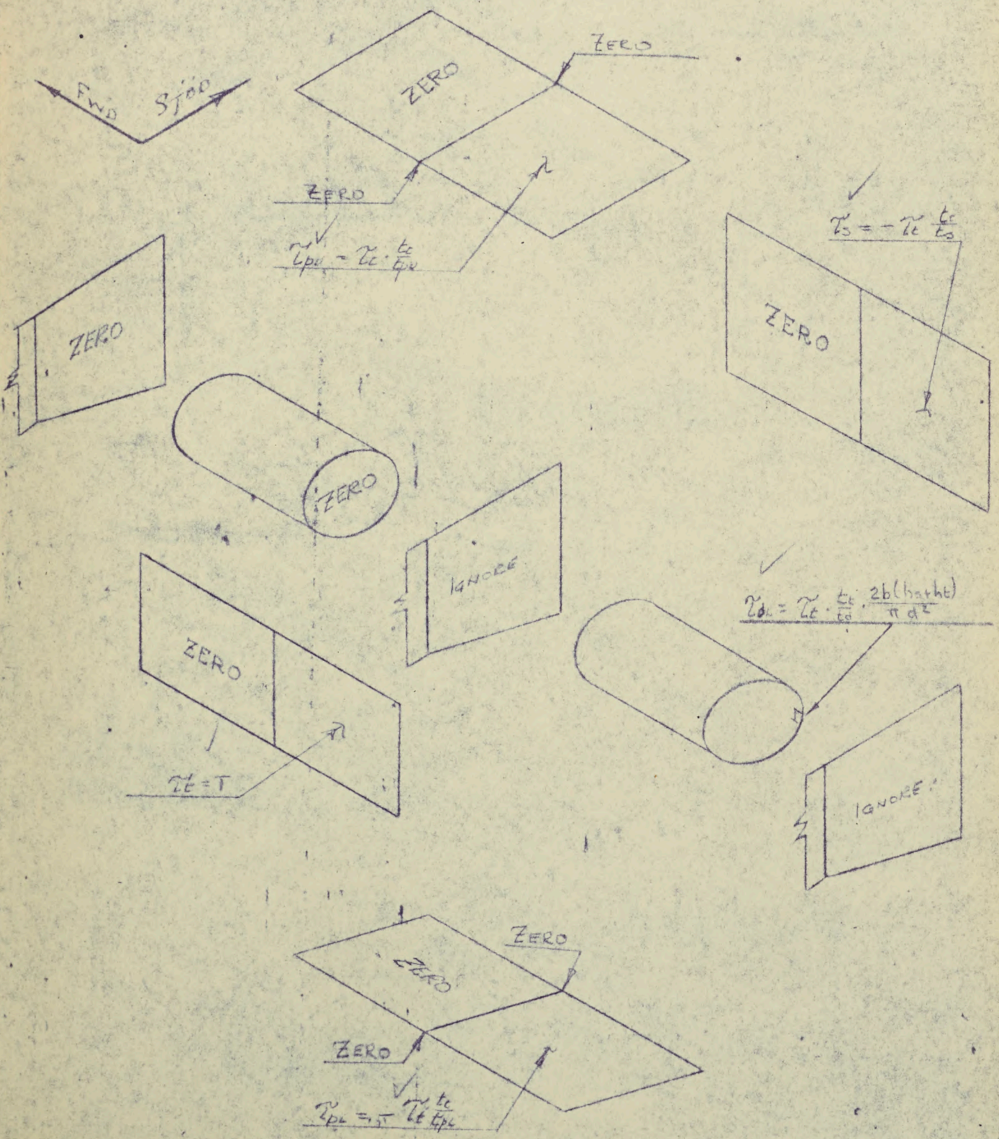
"F" GROUP.



AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO	7/0510/9	
SHEET NO	7-5	
AIRCRAFT	C105	CENTRE FUSELAGE
PREPARED BY	C.B.	DATE May 5 55
CHECKED BY		DATE

"T" GROUP.



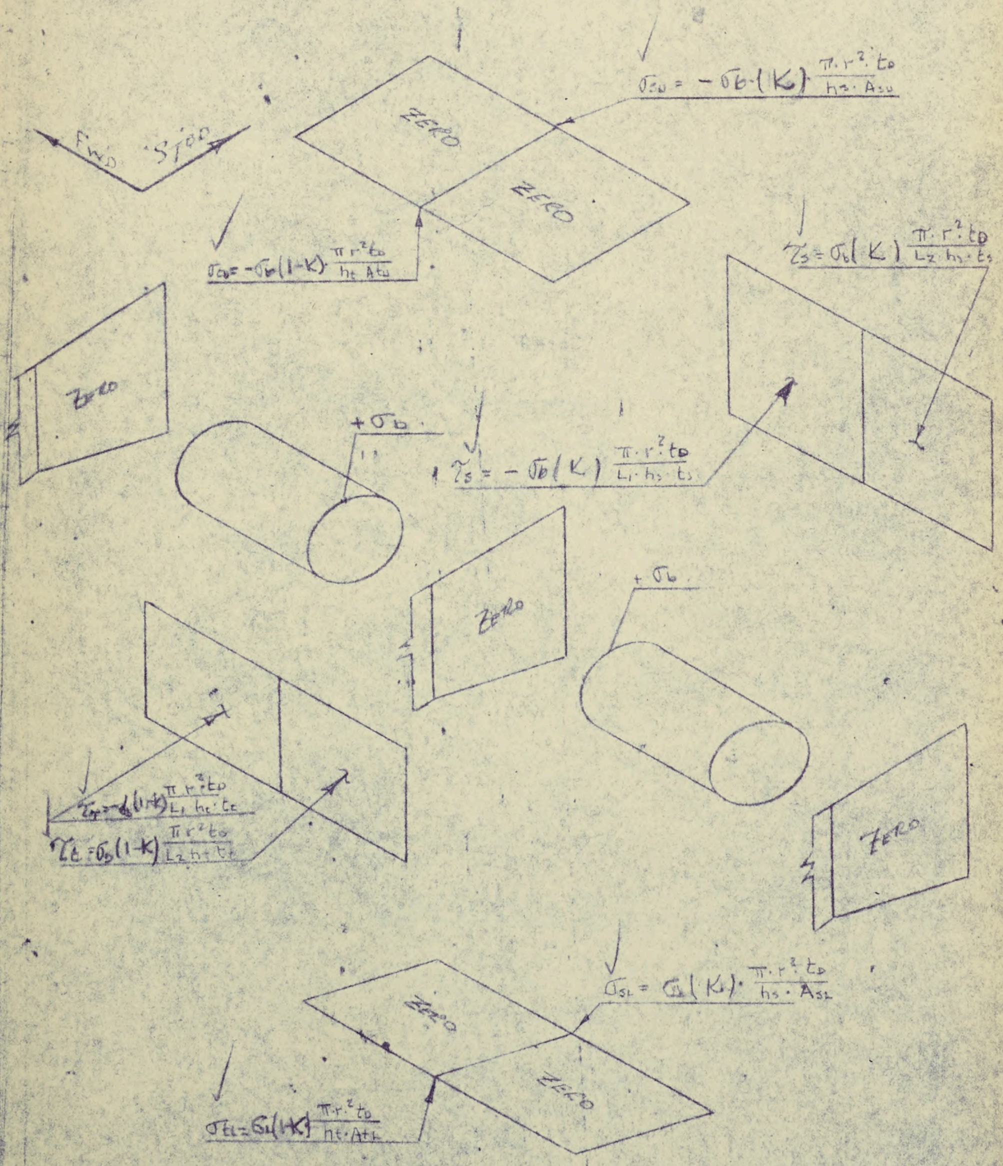
AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO 7/0510/9
SHEET NO 7-6

AIRCRAFT C105
CENTRE FUSELAGE

PREPARED BY C.B.
DATE MAY 5 '55
CHECKED BY
DATE

1. "B" GROUP.

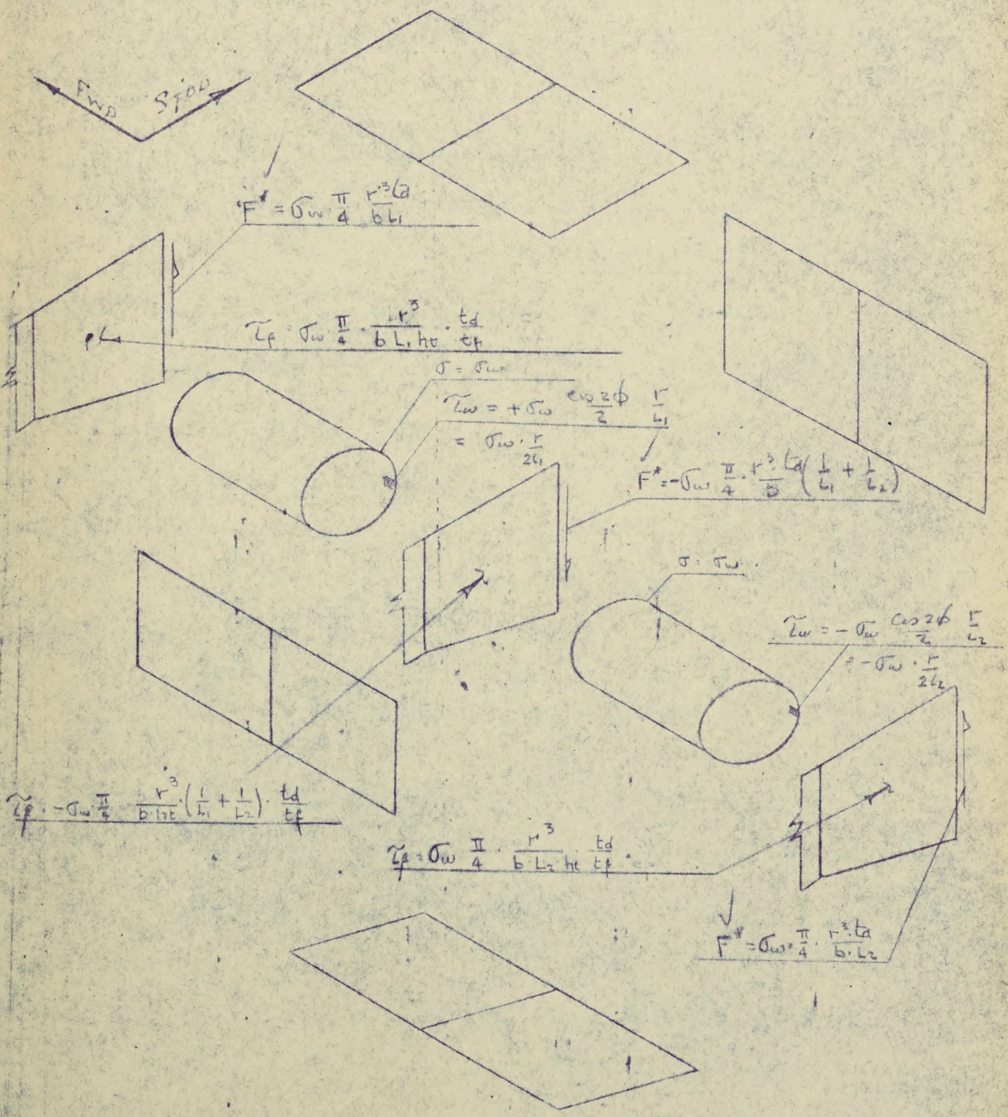


"K" BEING THE PROPORTION CARRIED ON THE FUSELAGE SIDE.

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

		REPORT NO 7/0510/9	
		SHEET NO 7-7	
AIRCRAFT		PREPARED BY	DATE
C105	CENTRE FUSELAGE	C.B.	May 5'55
		CHECKED BY	DATE

"W" GROUP.



AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/9
SHEET NO. 7-8

PREPARED BY: CF
DATE: May 4 55
CHECKED BY:
DATE:

AIRCRAFT: C105
CENTRE FUSELAGE

	U Pg 7-2	L Pg 7-3	F Pg 7-4	T Pg 7-5	B Pg 7-6	W Pg 7-7
σ_{sU}	$-\frac{A_{tU}}{A_{sU}}\cdot\frac{(h_s+h_U)}{h_s}$	$+\frac{A_{tL}}{A_{sU}}\cdot\frac{h_L}{h_s}$	•	•	$-(K)\frac{\pi\cdot r^2\cdot t_D}{h_s\cdot A_{sU}}$	•
σ_{sL}	$+\frac{A_{tU}}{A_{sL}}\cdot\frac{h_U}{h_s}$	$+\frac{A_{tL}}{A_{sL}}\cdot\frac{(h_s-h_L)}{h_s}$	•	•	$(K)\frac{\pi\cdot r^2\cdot t_D}{h_s\cdot A_{sL}}$	•
τ_s	•	•	$-\frac{t_t}{t_s}$	$-\frac{t_t}{t_s}$	$\mp(K)\frac{\pi\cdot r^2\cdot t_D}{L_1\cdot h_s\cdot t_s}$	•
σ_{tU}	+1.000 000	•	•	•	$-(1-K)\frac{\pi\cdot r^2\cdot t_D}{h_t\cdot A_{tU}}$	•
σ_{tL}	•	−1.000 000	•	•	$(1-K)\frac{\pi\cdot r^2\cdot t_D}{h_t\cdot A_{tL}}$	•
τ_t	Fwd $\mp\frac{A_{tU}}{L\cdot t_t}\cdot\frac{h_U}{h_s}$	$\pm\frac{A_{tL}}{L\cdot t_t}\cdot\frac{h_L}{h_s}$	+1.000 000	+1.000 000	$\mp(1-K)\frac{\pi\, r^2\, t_D}{L_1\cdot h_t\cdot t_t}$	•
τ_{pU}	$\mp\frac{A_{tU}(h_s+h_U)}{L\cdot t_{pU}\cdot h_s}$	$\pm\frac{A_{tL}}{L\cdot t_{pU}}\cdot\frac{h_L}{h_s}$	$+\frac{t_t}{t_{pU}}$	$+\frac{t_t}{t_{pU}}$	•	•
τ_{pL}	$\pm\frac{A_{tU}}{L\cdot t_{pL}}\cdot\frac{h_U}{h_s}$	$\pm\frac{A_{tL}}{L\cdot t_{pL}}\cdot\frac{(h_s-h_L)}{h_s}$	$-\frac{t_t}{t_{pL}}$	$-\frac{t_t}{t_{pL}}$	•	•
τ_f	•	•	$\mp\frac{t_t}{t_f}\cdot\frac{h_s}{h_t}$	•	•	$\pm\frac{\pi}{4}\cdot\frac{r^3}{b\cdot L\cdot h_t}\cdot\frac{t_d}{t_f}$ $\frac{1}{L}=\frac{1}{L_1}+\frac{1}{L_2}$ @ Centre
τ_{dL}	•	•	•	$\frac{t_t}{t_d}\cdot\frac{2b(h_s+h_t)}{\pi d^2}$	•	•
σ_{dw}	•	•	•	•	•	1.000 000
σ_{dt}	•	•	•	•	1.000 000	•

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C-105	CENTRE FUSELAGE	REPORT No. 7/0510/9 SHEET No. 8·1	
		PREPARED BY	DATE
		E. AUGUSTINE	18/[illegible]/55
		CHECKED BY	DATE
			JUNE /55

SECTION PROPERTIES.

REFERENCE SHOULD BE MADE TO DRAWING 7/0510/9 FIG. 1 SHT. SHOWING ACTUAL & IDEALIZED STRUCTURES.

CALCULATIONS HERE ARE CONCERNED ONLY WITH THE OUTER CELL OR ENVELOPE OF THE STRUCTURE. THIS ASPECT INVOLVES, ROUGHLY, A FOUR-SIDED BOX, DIVIDED BY FRAMES, FROM STATIONS 255 TO 485 & FORMED BY :

1. TOP FUSELAGE SKIN
2. OUTER SIDE " "
3. ARMAMENT - BAY ROOF
4. FUEL TANK SIDE WALL + AIR CONDITIONING SKIN.

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT		REPORT No. 7/0510/9	
C-105		SHEET No. 8-2	
		PREPARED BY	DATE
		E. AUGUSTINE	18/MAY/55
		CHECKED BY	DATE
		β	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 (NO'S. 2 → 8)
- SIDE " " (NO'S 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-REPT. NO. 0554/5). OF THE STRUCTURE CONCERNED. THE VALUES AVERAGED WERE FROM CASE 2.1(a) (ROOM TEMP. WT = 47000#, TENSION ON TOP) & CASE 3/4+V.E.B.M. (ROOM TEMP. WT. = 47000#, 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 TRAVERSED OUTBOARD TO THE FUSE. SIDE & THENCE FORE & AFT. HENCE AREAS ON TANK SIDE, BELOW U.I.L. & ON ARMAMENT

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C-105	REPORT No 7/0510/9	SHEET No 8-3
	PREPARED BY E. AUGUSTINE	DATE 18/MAY/55
	CHECKED BY B.	DATE JUNE 55

BAY FLOOR, OTHER THAN STRINGER #17, WERE CONSIDERED NON-EFFECTIVE.

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

- LONGITUDINAL MEMBER AREA + 30t WIDTH OF SKIN } SIMULATING COMPRESSION

- LONGITUDINAL MEMBER AREA + >30t WIDTH OF SKIN } SIMULATING TENSION.

(FACTOR ·6 TO CONVERT MAG. AREAS TO EQUIV. AL. AREAS)

1. INTERCOSTALS ON ARMAMENT BAY FLOOR. —
(DESIGNATED AS 18, 19, 20, 21, 22, REF. DWG. 7/0510/9 FIG 1)

THESE MEMBERS ARE 75ST Z SECTIONS WITH FLANGES BROKEN AT FORMER STATIONS, BUT HAVING CONTINUOUS SHEAR CONNECTIONS.

BASIC AREA IS CONSIDERED TO BE THE FLOOR CONNECTING FLANGE OF Z. SECTION.

BASIC AREA = .55 × .032 = .0176

30t WIDTH SKIN (MAG) (COMPRESSION) = 30 × .040 = 1.2 IN
WIDTH SKIN (TENSION) = 7.0 IN.
AREA SKIN (COMPRESSION) = 1.2 × .04 × .6 = ·0288
" " (TENSION) = 7.0 × .04 × .6 = ·1680

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

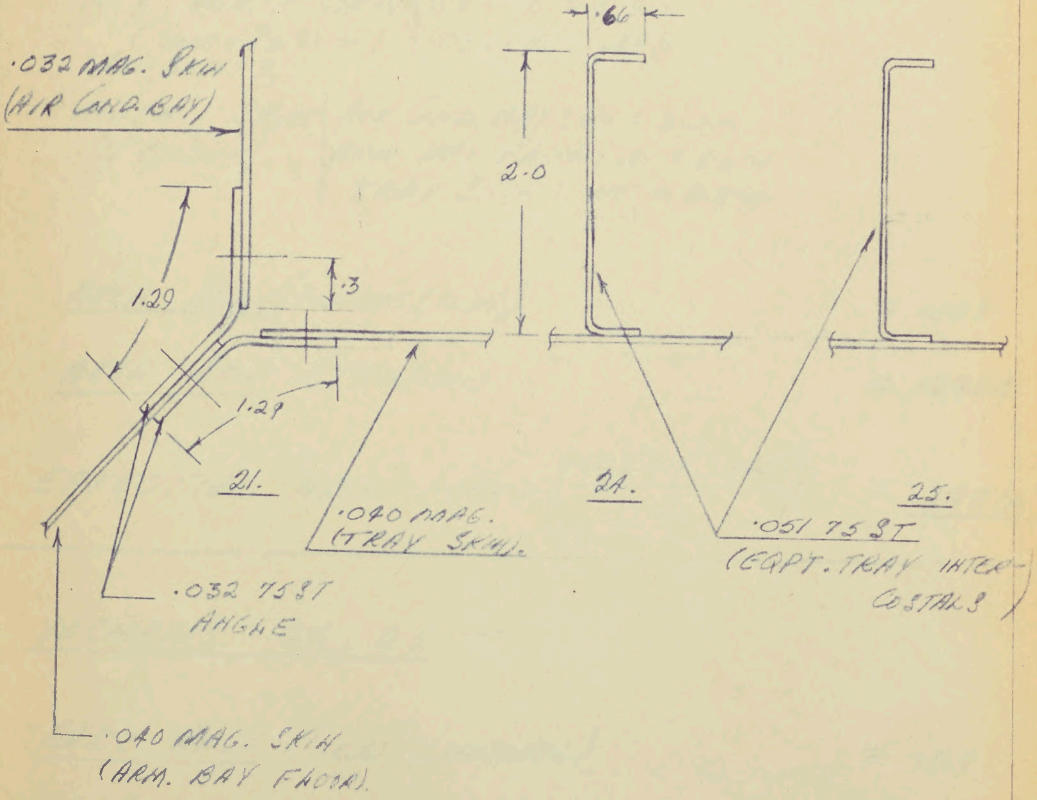
AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No	7/0510/9
SHEET No	8.4
PREPARED BY	E. AUGUSTINE
DATE	18/MAY/55
CHECKED BY	β
DATE	JUNE '55

AIRCRAFT: C-105

MEMBERS 19 & 20 ARE IN FACT BROKEN AT STATION 315 (REF. DWG. 7/0510/9 FIG. I), 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	19/MAY/55
CHECKED BY	DATE
β	JUNE '55

MEMBER 21 —

BASIC AREA (ANGLES + 3@.3IN SKIN) = .103

15 t WIDTH SKIN (COMPRESSION) (REF-SHEET 3). | .04 × 15 = 0.6IN
.032 × 15 = .48IN.

WIDTH SKIN (TENSION) | AIR CND. BAY : W = 2.0IN.
ARM. BAY FLOOR : W = 5.0IN.
TRAY SKIN : W = 3.5IN.

AREA. SKIN (COMPRESSION) = .038

AREA SKIN (TENSION) = .2424

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

MEMBERS 24, 25 —

BASIC AREA (.051 CHANNEL) = .169

30 t WIDTH SKIN (COMPRESSION) = 30 × .04 = 1.2 IN

WIDTH SKIN (TENSION) = 7.0 IN.

AREA SKIN (COMPRESSION) = .0288

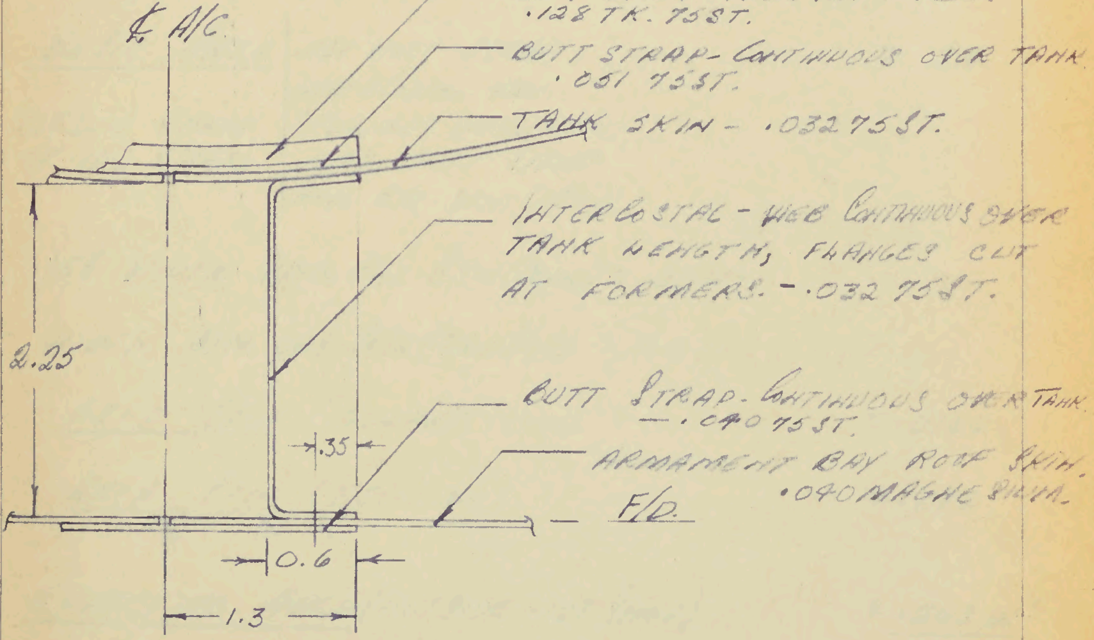
AREA SKIN (TENSION) = .168

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

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO	7/0510/9
SHEET NO	8-6
AIRCRAFT:	C-105
PREPARED BY	E. AUGUSTINE
DATE	19/MAY/55
CHECKED BY	β
DATE	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 #L5B10)

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

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No	7/0510/9
SHEET No	8-7
PREPARED BY	E. AUGUSTINE
DATE	19/MAY/55
CHECKED BY	
DATE	JUNE 55
AIRCRAFT	C-105

MEMBER 23. - CONT'D.

BASIC AREA (OUTSIDE RANGE OF ·128 STRAP).
- ·051 BUTT STRAP
- INTERCOSTAL WEB (FLANGES NEGLECTED)
- ·040 BUTT STRAP
- ARM. BAY SKIN (·95 IN).

= ·214

15t 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 IN^2

EFFECTIVE AREA (INCLUDING ·128 STRAP) = ·430 IN^2

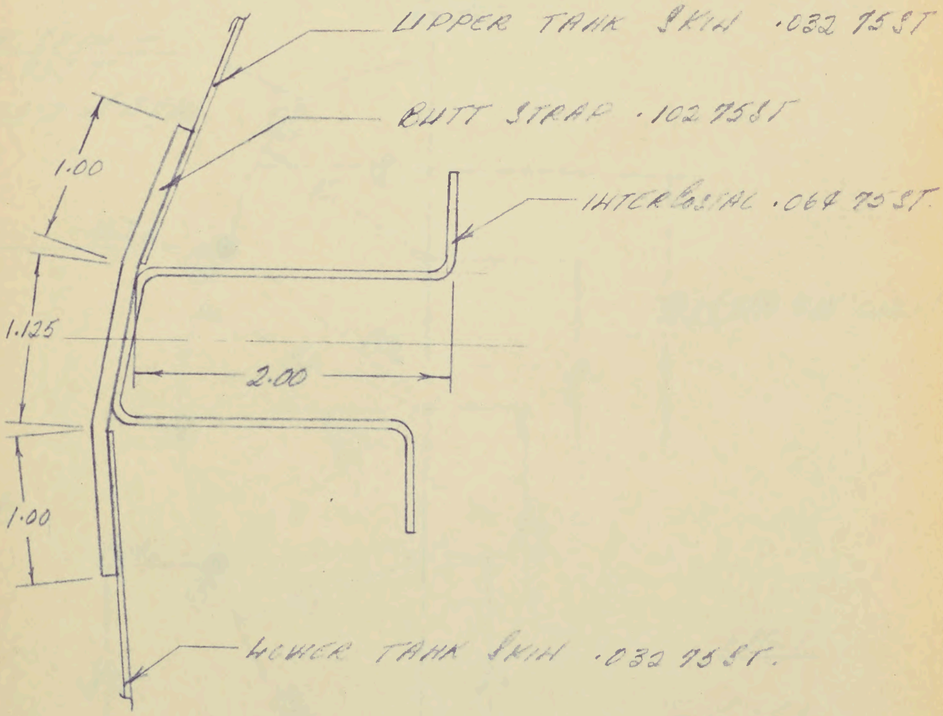
FORM 1DPA

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No	7/0510/9
SHEET No	8-8
AIRCRAFT:	C-105

PREPARED BY	DATE
E. AUGUSTINE	19/MAY/55
CHECKED BY	DATE
B	JUNE '55

TANK MIDDLE LONGERON. —



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

BUTT STRAP (.102) CONTINUOUS OVER TANK LENGTH.

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

EFFECTIVE AREA = .575 IN^2.

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C-105

REPORT No 7/5510/9
SHEET No 8-9

PREPARED BY	DATE
E. AUGUSTINE.	20/MAY/55
CHECKED BY	DATE
β.	JUNE '55

MOMENT OF INERTIA OF FUEL TANK WALL.

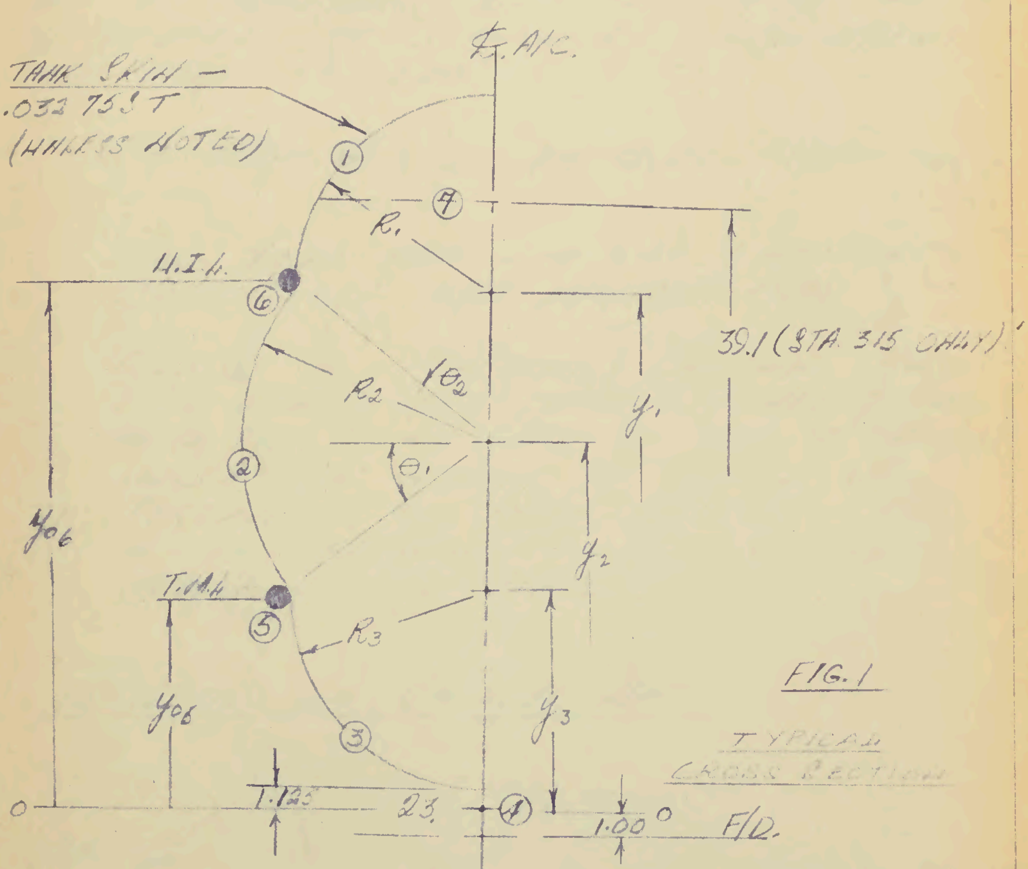


FIG. 1

TYPICAL CROSS SECTION

$$I = \sum_{i=1}^{i=7}\left(y_0^2\cdot\Delta A + i_0\right)_i - \bar{y}^2\cdot A$$

WHERE THE SUMMATION IS TAKEN OVER THE 7 ELEMENTS DESIGNATED AS ⊙.

y_0 – DIST. FROM REF. AXIS TO CENTROID OF ELEMENT
i_0 – MOM. OF INERTIA, ABOUT C.G. OF EACH ELEMENT.
$\bar{y}$ – DIST. FROM REF. AXIS TO TANK C.G.

REPORT NO. 7/0510/9
SHEET NO. B-10
AIRCRAFT: C-105

PREPARED BY	DATE
E. AUGUSTINE	20/MAY/55
CHECKED BY	DATE
β	JUNE '55

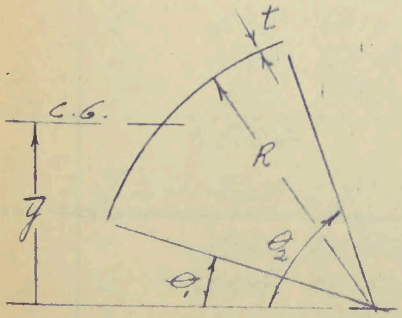
MOMENTS OF INERTIA ARE CALCULATED AT 3 STATIONS; 315, 392, & 478. ELEMENT ⑦ APPEARS AT STATION 315 ONLY WHERE THE TANK IS FLAT ON TOP.

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

TANK SKINS ARE ·03275 ST EXCEPT AT STATION 315 (REF. DWG 7/0510/9 FIG.1).

AREA ⑥ IS PRACTICALLY CONSTANT OVER TANK LENGTH (REF. TABLE OF AREAS - DWG. 7/0510/9 FIG. 1). AT EACH OF THE 3 STATIONS WHERE THE MOMENT OF INERTIA IS CALCULATED, THE AREA OF ⑤ IS TAKEN AS THE AVERAGE OF AREAS IN THE REGION OF THAT STATION.

PROPERTIES OF CIRCULAR ARCS.



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

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

$$i_0 = \left(\frac{R^2}{2} - \bar{y}^2\right)A - \frac{R^2 t}{2}(\sin\theta_2\cos\theta_2 - \sin\theta_1\cos\theta_1)$$

i_0 - MOMENT OF INERTIA ABOUT C.G.

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No	7/0510/9
SHEET No	8 - 11
PREPARED BY	E. AUGUSTINE
DATE	20 MAY '55
CHECKED BY	β
DATE	JUNE '55
AIRCRAFT:	C-105

CALCULATION OF $\bar{y}$ & i_o OF TANK 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_o ⑤	y_o ⑥
315	15.03	34.6	0	17.5	18.25	24.65	$\overline{45.5}$	34.	13.25	14.50	$\overline{90}$	$\overline{14}$	12.0	35.0
392	11.52	35.22	0	90	16.0	24.55	$\overline{35.}$	41.5	13.25	14.50	$\overline{90}$	0	16.0	36.0
478	7.93	35.6	0	90.	13.56	24.65	$\overline{24.5}$	55	13.25	14.50	$\overline{90}$	20	19.5	36.0

CALCULATION OF MOMENT OF INERTIA OF TANK.

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

STA.	ELEM.	ΔA	y_o	$y_o \Delta A$	y_o^2	$y_o^2 \Delta A$	i_o
315	1	.1836	36.8783		1360.0090		.3083
	2	.8103	22.9651		527.3108		38.9452
	3	.5625	4.8076		23.1130		5.2771
	4	.263	0		0		0
	5	.575	12.0		144.0		0
	6	.549	35.0		1225.0		0
	7	.560	39.1		1528.8100		0
		3.5034		76.0948		2301.5061	44.5316

$\bar{y} = 21.7203$; $\bar{y}^2 \cdot A = 1652.8640$; $I = 693.2337$.

STA.	ELEM.	ΔA	y_o	$y_o \Delta A$	y_o^2	$y_o^2 \Delta A$	i_o
392	1	.5790	42.5538		1810.8260		7.2784
	2	.6836	25.3912		644.7130		23.6962
	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
		3.3346		75.6832		2397.0187	42.0493

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

FORM 1119A

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0510/9
SHEET No. 8 - 12

AIRCRAFT: C-105

PREPARED BY	DATE
E. AUGUSTINE	29/MAY/55
CHECKED BY	DATE
β	JUNE '55

TANK MOMENT OF INERTIA (CONT'D.)

STA.	ELEM.	ΔA	y_0	$y_0\Delta A$	y_0^2	$y_0^2\Delta A$	i_0
	1	.3987	40.6480		1652.2724		2.3747
	2	.6021	27.9375		780.5039		15.0478
478	3	.8140	8.0147		64.2354		25.2555
	4	.430	0		0	0	0
	5	.575	19.5		380.2500		0
	6	.575	36.0		1296.0000		0
		3.3948		71.4642		2144.8417	42.6780

$\bar{y} = 21.0511$; $\bar{y}^2 A = 1504.4016$; $I = 683.1181$.

TABLE FOR CALCULATING CONTRIBUTION TO MOM. OF INERTIA OF TANK FROM CIRCULAR ELEMENTS.

RADII (R) & ANGLES (θ) TAKEN FROM DWGS. OF 3
PRESSURE BULKHEADS.

7/0154-0008 SHT 2 ISS 2, 7-0154-0008 SHT 3 ISS 5, SHT 4 ISS 7,

A.

A
W
C

		1	2	3	4	5	6	7	8	9	10
STA.	① ② ③	t	Rt	$(\theta_2-\theta_1)$ (RADIANS)	A ②×③	$\frac{R}{(\theta_2-\theta_1)}$ = ④/①	Cos θ_2	Cos θ_1	⑥-⑦	$\bar{y}$ -⑤×⑧	$\frac{R^2}{2}$
		·040	·6012	·30544	·1836	49 ·2077	·1537	1 ·0	— ·0463	2 ·2783	107 ·759
315		·032	·5840	1 ·38754	·8103	13 ·1528	·8290	·7009	·1281	$\overline{7}$ ·6849	16 ·692
		·032	·4240	1 ·32645	·5625	9 ·9891	·9703	0	·9703	$\overline{9}$ ·6924	$\overline{6}$ ·148
		·032	·3686	1 ·57080	·5790	11 ·3338	0	1 ·0	$\overline{1}$ ·0	7 ·3338	12 ·570
392		·032	·5120	1 ·33518	·6836	11 ·9834	·7490	·8192	— ·0702	·8412	127 ·292
		·032	·4240	1 ·57080	·6660	8 ·4352	1 ·0	0	1 ·0	$\overline{8}$ ·4352	16 ·62
		·032	·2538	1 ·57080	·3987	5 ·0484	0	1 ·0	$\overline{1}$ ·0	5 ·0484	5 ·95
478		·032	·4339	1 ·38754	·6021	9 ·7727	·5736	·9100	— ·3364	3 ·2875	81 ·12
		·032	·4240	1 ·91986	·8140	6 ·9015	·9397	0	·9397	$\overline{6}$ ·4853	45 ·72

FORM 1543

CULATING CONTRIBUTION TO MOM. OF INERTIA
OF TANK FROM CIRCULAR ELEMENTS.
GES (θ) TAKEN FROM DWG. OF 5
KHEADS.

0154-0008 SHT3 Iss 5, SHT4 Iss 7,

A. V. ROE CANADA LIMITED
MALTON, ONTARIO
TECHNICAL DEPT. (AIRFRAME)

AIRCRAFT -
WEIGHT -
C. G. POSITION -

REPORT NO. -
SHEET -
DATE -
CHECKED
PREPARED BY -

3	4	5	6	7	8	9	10	11	12	13	14	15	16	17
$(\theta_2-\theta_1)$ (RADIANS)	A ②×③	$\frac{R}{(\theta_2-\theta_1)}$ = ④/①	$\cos\theta_2$	$\cos\theta_1$	⑥−⑦	$\bar{y}$ −⑤×⑧	$(\frac{R^2}{2}-\bar{y}^2)$	$\frac{R^2\theta}{2}$	$\sin\theta_2$	$\sin\theta_1$	⑫ ⑥ −⑬×⑦	I_o ⑩×④ −⑪×⑭	y_o $(y+\bar{y})$	
.30544	.1836	49.2077	.9537	1.0	−.0463	2.2783	107.7598	67.9058	.3007	0	.2868	.3093	36.8783	
1.38754	.8103	13.1528	.8290	.7009	.1281	−1.6849	163.6924	97.2548	.5592	−.7133	.9635	38.9452	22.9651	
1.32645	.5625	9.9891	.9703	0	.9703	−9.6924	−6.1481	37.2193	−.2419	−1.0	−.2347	5.2771	4.8076	
1.57080	.5790	7.3338	0	1.0	−1.0	7.3338	12.5706	24.4612	1.0	0	0	7.2784	42.5338	
1.33518	.6836	11.9834	.7490	.8192	−.0702	.8412	127.2924	65.5360	.6626	−.5736	.9662	23.6962	25.3912	
1.57080	.6660	8.4352	1.0	0	1.0	−8.4352	16.6287	37.2193	0	−1.0	0	11.0747	6.0648	
1.57080	.3987	5.0484	0	1.0	−1.0	5.0484	5.9561	7.9801	1.0	0	0	2.3747	40.6484	
1.38754	.6021	9.7727	.5736	.9100	−.3364	3.2875	81.1291	39.8414	.8192	−.4147	.8473	15.0978	27.9315	
1.91986	.8140	6.9015	.9397	0	.9397	−6.4833	45.7221	37.2193	.3420	−1.0	.3214	23.2355	8.0197	

…a II Mom. of Inertia
…ular Elements.
…m Dis. of 3

…Ass. 7,

A. V. ROE CANADA LIMITED
MALTON, ONTARIO
TECHNICAL DEPT. (AIRFRAME)

AIRCRAFT -
WEIGHT -
C. G. POSITION -

REPORT NO. - 7/0340/9
SHEET - 8-13
DATE -
Checked [illegible] June 55
PREPARED BY - E.A.

6	7	8	9	10	11	12	13	14	15	16	17	18	19
θ_2	Cos θ_1	⑥ − ⑦	$\bar{y}$ −③×⑤	$(\frac{k}{2}-\bar{y})^2$	$\frac{k^2 t}{2}$	Sin θ_2	Sin θ_1	⑫×⑥ −⑬×⑦	i_o ⑩×⑨ −⑪×⑭	y_o $(y+\bar{y})$			
337	1 .0	— .0463	2 .2783	107 .7598	67 .9058	.3007	0	.2868	.3093	36 .8783			
90	.7009	.1281	$\overline{1}$ .6849	163 .6924	97 .2543	.5592	— .7133	.9635	38 .9452	22 .9651			
03	0	.9703	$\overline{9}$ .6924	$\overline{6}$ .1481	37 .2193	— .2419	$\overline{1}$ .0	— .2347	5 .2771	4 .8076			
	1 .0	$\overline{1}$ .0	7 .3338	12 .5706	24 .4612	1 .0	0	0	7 .2784	42 .5538			
490	.8192	— .0702	.8412	127 .2924	65 .5360	.6626	— .5736	.9662	23 .6962	25 .3912			
	0	1 .0	$\overline{8}$ .4352	16 .6287	37 .2193	0	$\overline{1}$ .0	0	11 .0747	6 .0648			
	1 .0	$\overline{1}$ .0	5 .0484	5 .9531	7 .9801	1 .0	0	0	2 .3747	40 .6484			
360	.9100	— .3364	3 .2875	81 .1291	39 .8414	.8192	— .4147	.8473	15 .0478	27 .9375			
377	0	.9397	$\overline{6}$ .4853	45 .7221	37 .2193	.3420	$\overline{1}$ .0	.3214	25 .2555	8 .0197			

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No	7/0510/9
SHEET No	8-14
PREPARED BY	F. AUGUSTINE
DATE	24 MAY '55
CHECKED BY	3
DATE	JUNE '55

AIRCRAFT: C-105

MOMENT OF INERTIA OF AIR-COND. BAY WALL.

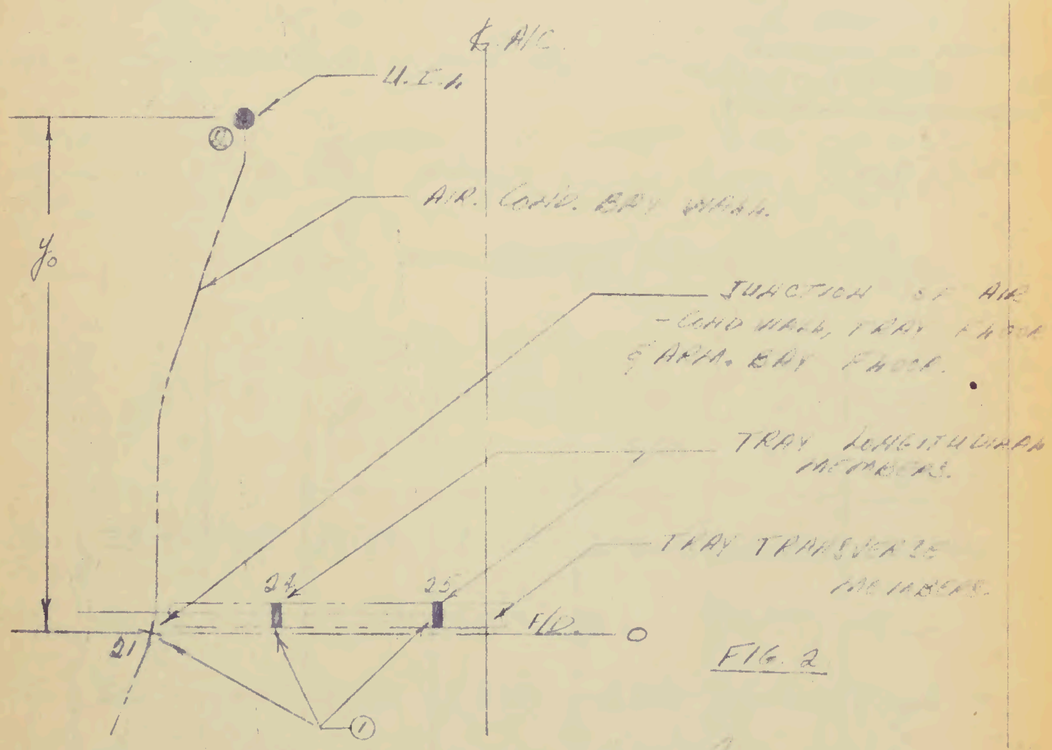


FIG. 2

TYPICAL CROSS SECTION

AIR CONDITIONING BAY WALL 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.

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

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C-105		REPORT No. 7/0510/9	
		SHEET No. 8-15	
		PREPARED BY	DATE
		E. AUGUSTINE	25/MAY/55
		CHECKED BY	DATE
			JUNE '55

MOMENT OF INERTIA OF AIR-COND. BAY WALL CONT'D.

REF. DWG. 7/0510/9 FIG. 1 FOR A_2 (U.I.L.) (A_2 TAKEN AS AVERAGE OF AREAS AT STAS. STRADDLING STA. WHERE I CALCULATED)

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

y_0 - SCALED FROM DWG. 7/0510/9 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 ①×③ − ⑥×④
255	33.5	1.360	45.56	2.137	21.3196	454.5253	554.9394
268	34.0	.896	30.4640	1.673	18.2092	331.5750	481.0510
292	35.0	.605	21.1750	1.382	15.3220	234.7637	416.6816
315	36.0	.549	19.7640	1.326	14.9050	222.1590	416.9212

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No 7/0510/9
SHEET No 8-16

AIRCRAFT: C-105

PREPARED BY	DATE
F. AUGUSTINE.	25/MAY/55
CHECKED BY	**DATE**
B	JUNE '55

MOMENT OF INERTIA OF OUTER FUSELAGE SIDE.

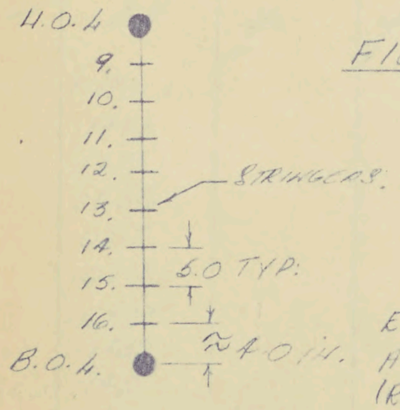


FIG. 3

TYPICAL CROSS SECTION.

ELEMENTAL AREAS ARE TAKEN AS THE AVERAGE OF THE AREAS (REF. TABLE - DWG. 7/0510/9 FIG. 1) IN THE REGION OF THE STATION CONCERNED.

STA. 255 AREAS - AVERAGE OF STAS. 255, 268.

ELEM	ΔA	y_0	$y_0 \Delta A$	y_0^2	$y_0^2 \Delta A$
B.O.L.	.9165	0	0	0	0
16	.197	4.0		16.0	
15	.1495	9.0		81.0	
14	0	14.0		196.0	
13	0	19.0		361.0	
12	0	24.0		576.0	
11	0	29.0		841.0	
10	.1425	34.0		1156.0	
9	.1395	39.0		1521.0	
U.O.L.	.1655	42.5		1806.25	
	1.6605		19.29275		690.3054

$\bar{y} = 11.6186$; $\bar{y}^2 A = 224.1542$; $I = 466.1512$

FORM 1310A

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C-105		REPORT NO. 7/0510/9	
		SHEET NO. 8-17	
		PREPARED BY E. AUGUSTINE	DATE 25/MAY/55
		CHECKED BY B	DATE JUNE '55

STA. 292. AREAS - AVERAGE OF STA'S. 268, 292, 315

ELEM.	ΔA	y_0	$y_0 \Delta A$	y_0^2	$y_0^2 \Delta A$
	1.193				
	.1443				
	.1473				
	.1056				
	.0817				
	.0818				
	.0997				
	.1937				
	.140				
	.263	44.5		1980.2500	
	2.4001		31.8374		1095.2589

$\bar{y} = 13.2650$; $\bar{y}^2 A = 422.3221$; $I = 672.9368$

STA 337 AREAS AVERAGE OF STA'S. 315, 337, 359.

	1.6757				
	.1393				
	.142				
	.148				
	.140				
	.1627				
	.1557				
	.1517				
	.148				
	.5753	45.0		2025.0	
	3.4384		51.8056		1883.3934

$\bar{y} = 15.0668$; $\bar{y}^2 \cdot A = 780.5460$; $I = 1102.8474$

FORM 1319A

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C-105		REPORT NO. 7/0510/9	
		SHEET NO. 8-18	
		PREPARED BY: A. AUGUSTINE	DATE: 25/MAY/55
		CHECKED BY: B	DATE: JUNE '55

STA. 392. AREAS AVERAGE OF STA'S. 359, 392, 414.

ELEM.	ΔA	y_0	$y_0 \Delta A$	y_0^2	$y_0^2 \Delta A$
	2.2447				
	.1357				
	.138				
	.1423				
	.106				
	.1457				
	.170				
	.166				
	.163				
	1.211	45.0		2025.0	
	4.6224		80.7138		3198.4932

$\bar{y} = 17.4614$; $\bar{y}^2 A = 1409.3721$; $I = 1789.1211$

STA. 469 AREAS AVERAGE OF STA S. 414, 469, 485.

	2.920				
	.0793				
	.0487				
	.0503				
	.0393				
	.0737				
	.183				
	.1803				
	.1463				
	1.745	45.0		2025.0	
	5.4659.		99.6431		4190.1879

$\bar{y} = 18.2300$; $\bar{y}^2 A = 1816.4984$; $I = 2373.6895$

FORM 1319A

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No 7/0510/9
SHEET No 8-19

AIRCRAFT: C-105

PREPARED BY	DATE
E. AUGUSTINE	26/MAY/55
CHECKED BY	DATE
β	JUNE '55

STA. 485 - AREAS - THOSE AT 485 ONLY.

ELEM	ΔA	y_0	$y_0 \Delta A$	y_0^2	$y_0^2 \Delta A$
	3.144				
	0				
	0				
	0				
	0				
	0				
	.137				
	.137				
	.137				
	1.850	44.5		1980.2500	
	5.4050		96.2190		4145.4285

$\bar{y} = 17.8167$; $\bar{y}^2 A = 1715.7351$; $I = 2429.6934$

FORM 1319A

2. IDEALIZED STRUCTURE

THIS SECTION IS CONCERNED WITH REDUCING THE FORGOING 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.4). THIS CROSS SECTION REMAINS CONSTANT THROUGHOUT THE STRUCTURE LENGTH.

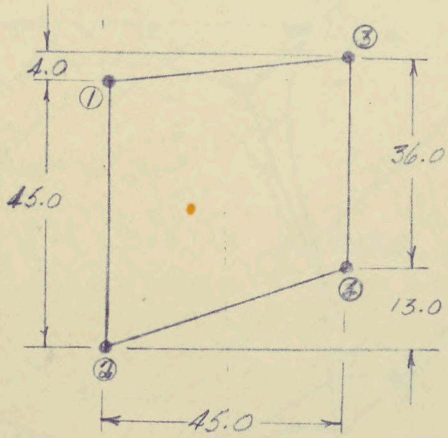


FIG. 4.

①-② SIMULATES FUSELAGE SIDE.

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

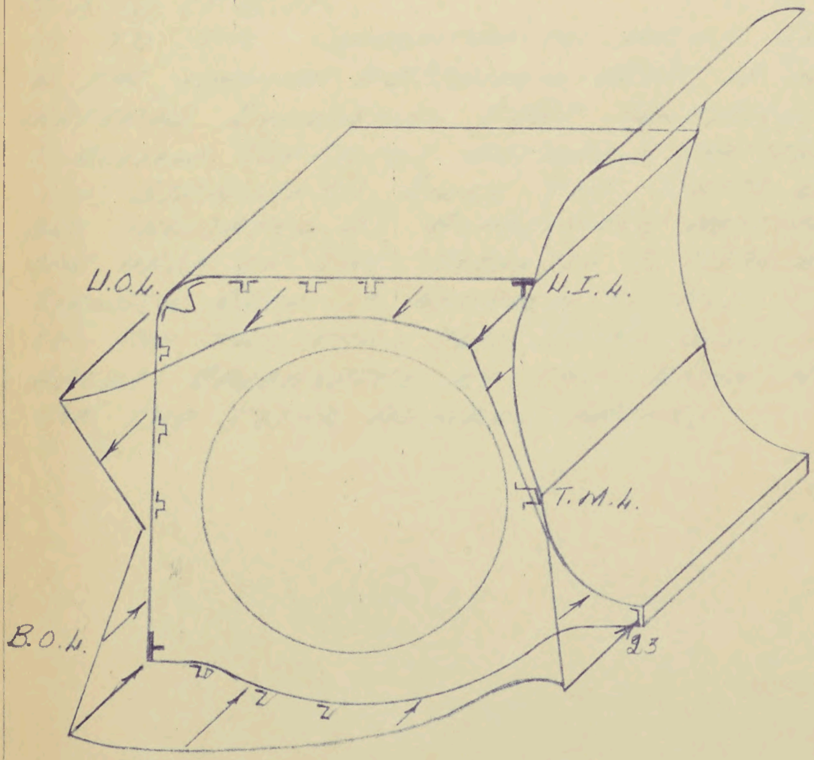


FIG. 5.

CONSEQUENTLY A RATIONAL PROCEDURE IS INDICATED FOR THE LUMPING OF AREAS. CONSIDERING THE OUTER FUSELAGE 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

OF INERTIA BUT HAVING CONCENTRATED CAPS IN THE LOCATIONS ① & ② AS SHOWN IN FIG. 1. THE TANK WALL*, HAVING THE SAME DISTRIBUTION IS LUMPED IN A SIMILAR MANNER, THE CONCENTRATED CAPS BEING NOW PLACED 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 WALLS. (i.e. STRINGERS). 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 ANALOGY, FOR THE TOTAL LUMPED AREAS.

---

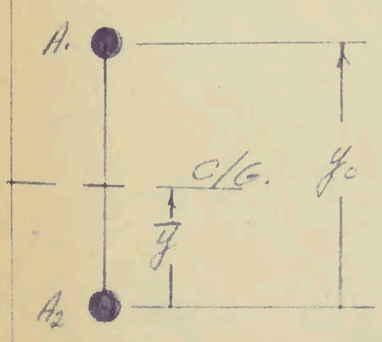
* & AIR CONDITIONING BAY WALL.

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No	7/0510/9
SHEET No	8-23

AIRCRAFT:	PREPARED BY	DATE
C-105	E. AUGUSTINE	21/MAY/55
	CHECKED BY	DATE
	B	JUNE '55

FORMULAE FOR CONVERTING ACTUAL MOMENTS OF INERTIA OF OUTER FUSELAGE SIDE WALL & TANK & AIR-COND. BAY WALL 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 BEAR 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 $y_0^2 A_1' - \bar{y}^2 (A_1' + A_2') = I$ (CALCULATED FOR ACTUAL STRUCTURE)

② $\bar{y} = k y_0 = \dfrac{y_0 A_1'}{(A_1' + A_2')}$ ($k = \bar{y}/y_0$ ACTUAL STRUCTURE).

REFERENCE CALCULATIONS - ACTUAL STRUCTURE — FOR I & k. (SHEETS 8-11-12; 15-16 & SEQ.).

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C-105	REPORT NO 7/0510/9	SHEET NO 8-24
	PREPARED BY E. Augustine	DATE 21/6/55
	CHECKED BY	DATE 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 —

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

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

&

		OUTER FUSE SIDE	TANK SIDE
$A_1' =$	$\dfrac{I}{(1-k) y_0^2} =$	$\dfrac{I}{2025(1-k)}$	$\dfrac{I}{1296(1-k)}$
$A_2' =$	$\dfrac{I}{k y_0^2} =$	$\dfrac{I}{2025 k}$	$\dfrac{I}{1296 k}$

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

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

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

WHERE —

REF. DWG. 7/0510/9 FIG. 1

$A_T =$ SUM OF AREAS OF MEMBERS 2 → 8 ON TOP SKIN.

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

FORM 1319A

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No	7/0510/9
SHEET No	8-25
AIRCRAFT:	C-105
PREPARED BY	E. ALLENSTINE
DATE	26/MAY/55
CHECKED BY	β
DATE	JUNE '55

REF. DWG. 7/0610/9 FIG. 1 & SHEET 8-24 FOR ORIGIN OF THESE VALUES

STA.	A_T	A_B
255	1.040	.601
268	.960	.601
292	.967	.610
315	.963	.601
337	.990	.832
359	.985	.832
392	.988	.832
414	.992	.832
469	.987	.832
485	.985	.832

A①, A②, A③, A④ - LUMPED AREAS ARE CALCULATED ON FOLLOWING SHEET

- INTERPOLATION BETWEEN STATIONS IS MADE, WHERE NECESSARY, TO ARRIVE AT APPLICABLE VALUES OF A_T & A_B.

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C-105

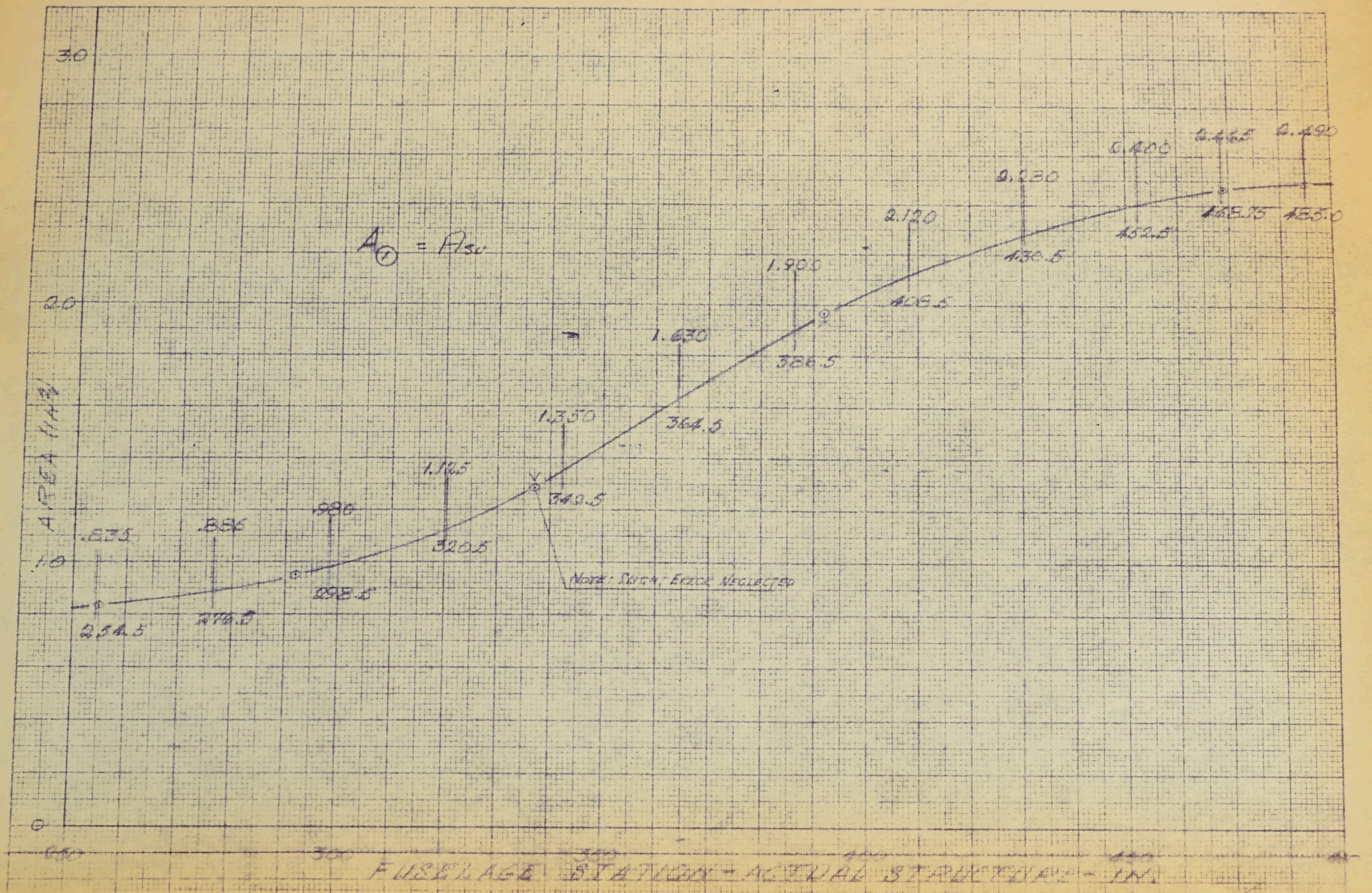
REPORT No 7/0510/9
SHEET No 8-26

PREPARED BY: F. AUGUSTINE
DATE: 10/JUNE/55.
CHECKED BY: B
DATE: JUNE/55

① STA.	② I	③ k	④ (1-k)	⑤ 2025(1-k)	⑥ 2025 k	⑦ $A'_1 = ②/⑤$	⑧ $A'_2 = ②/⑥$	⑨ $A_T/2$	⑩ $A_B/2$	⑪ $A_1 = A_{3U} = ⑦+⑨$	⑫ $A_2 = A_{3L} = ⑧+⑩$
255	466.1512	.274	.726	1470.1500	554.8500	.3171	.8401	.520	.3005	.8371	1.1406
292	672.9368	.298	.702	1421.5500	603.4500	.4734	1.1151	.4335	.305	.9569	1.4201
357	1102.8474	.335	.665	1346.6250	678.3750	.8190	1.6257	.495	.416	1.3140	2.0417
392	1789.1211	.388	.612	1239.3000	785.7000	1.4437	2.2771	.494	.416	1.9377	2.6931
469	2373.6815	.405	.595	1204.8750	820.1250	1.9701	2.8943	.4935	.416	2.4636	3.3103
485											

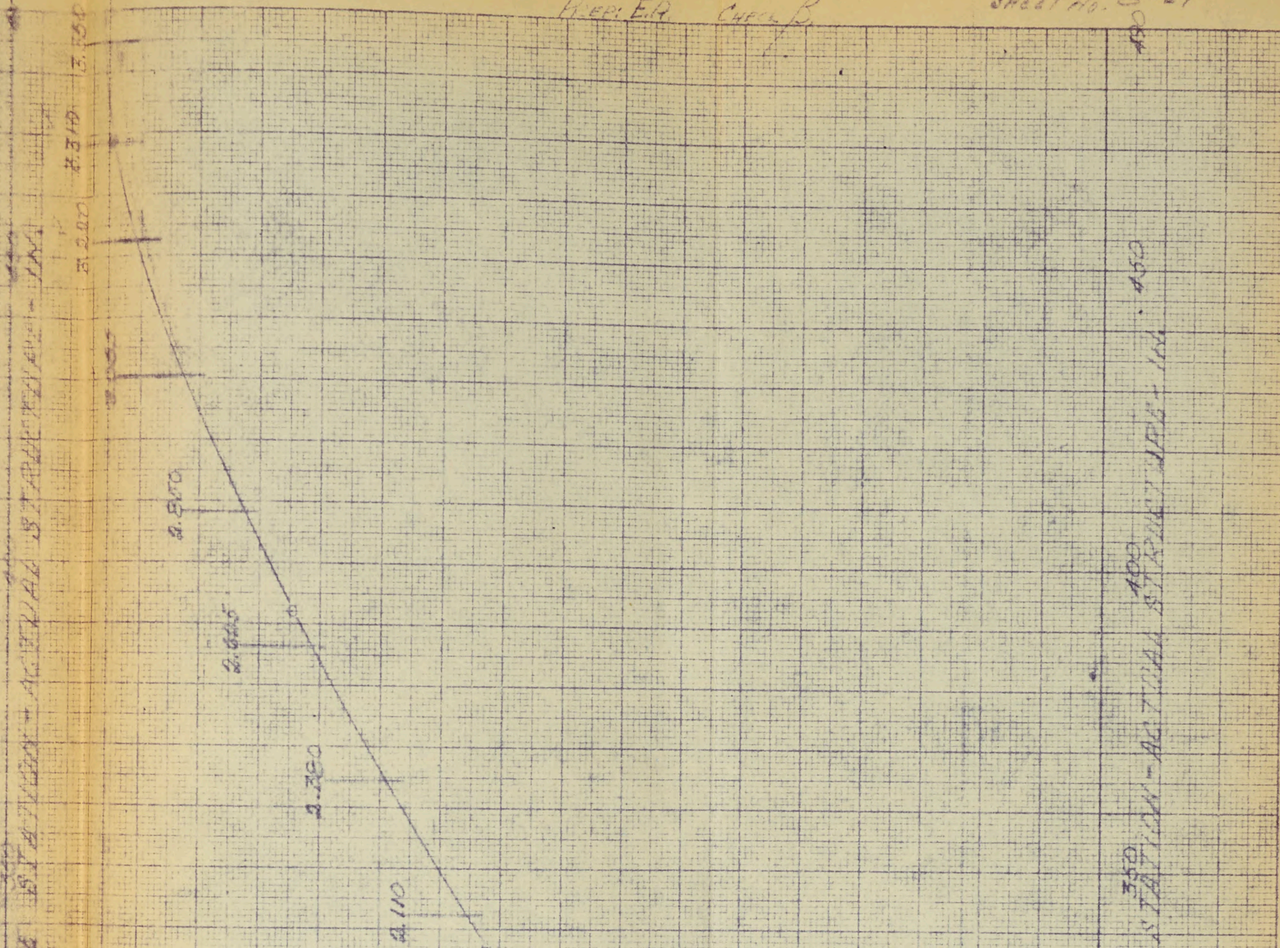
① STA.	② I	③ k	④ (1-k)	⑤ 1296(1-k)	⑥ 1296 k	⑦ $A'_3 = ②/⑤$	⑧ $A'_4 = ②/⑥$	⑨ $A_T/2$	⑩ $A_B/2$	⑪ $A_3 = A_{TU} = ⑦+⑨$	⑫ $A_4 = A_{TL} = ⑧+⑩$
255	554.9394	.636	.364	471.744	824.2560	1.1764	.6733	.520	.3005	1.6964	.9738
268	481.0510	.536	.464	601.3440	694.6560	.8000	.6925	.480	.3005	1.2800	.9930
292	416.6846	.438	.562	728.3520	567.6480	.5721	.7340	.4835	.305	1.0556	1.0390
315	416.9212	.415	.585	758.1600	537.8400	.5500	.7752	.4815	.3005	1.0315	1.0757
	693.2337	.555	.445	576.7200	719.2800	1.2020	.9638	.4815	.3005	1.6835	1.2643
392	721.3421	.485	.515	667.4400	628.5600	1.0808	1.1476	.494	.416	1.5748	1.5636
478	633.1181	.483	.517	670.0320	625.9680	1.0195	1.0913	.493	.416	1.5125	1.5073
485											

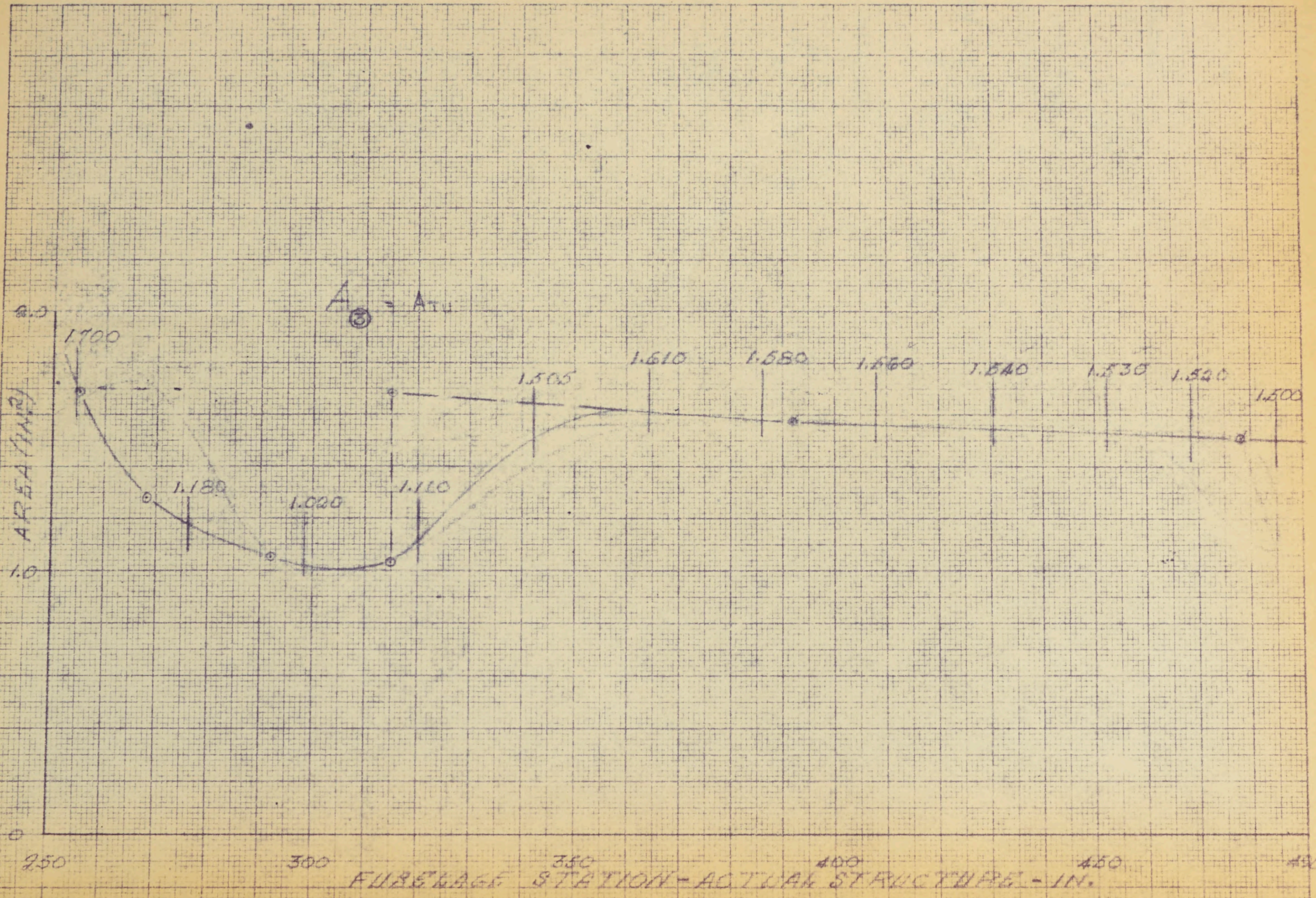
359-14L KEUFFEL & ESSER CO.
Millimeters, 5 mm. lines accented, cm. lines heavy.
MADE IN U.S.A.

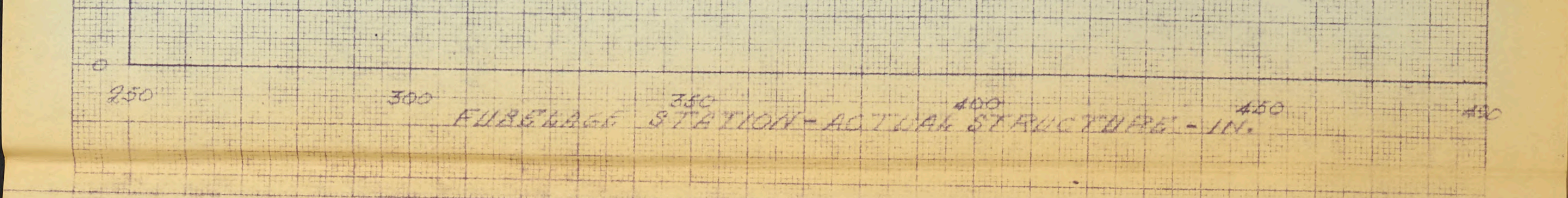


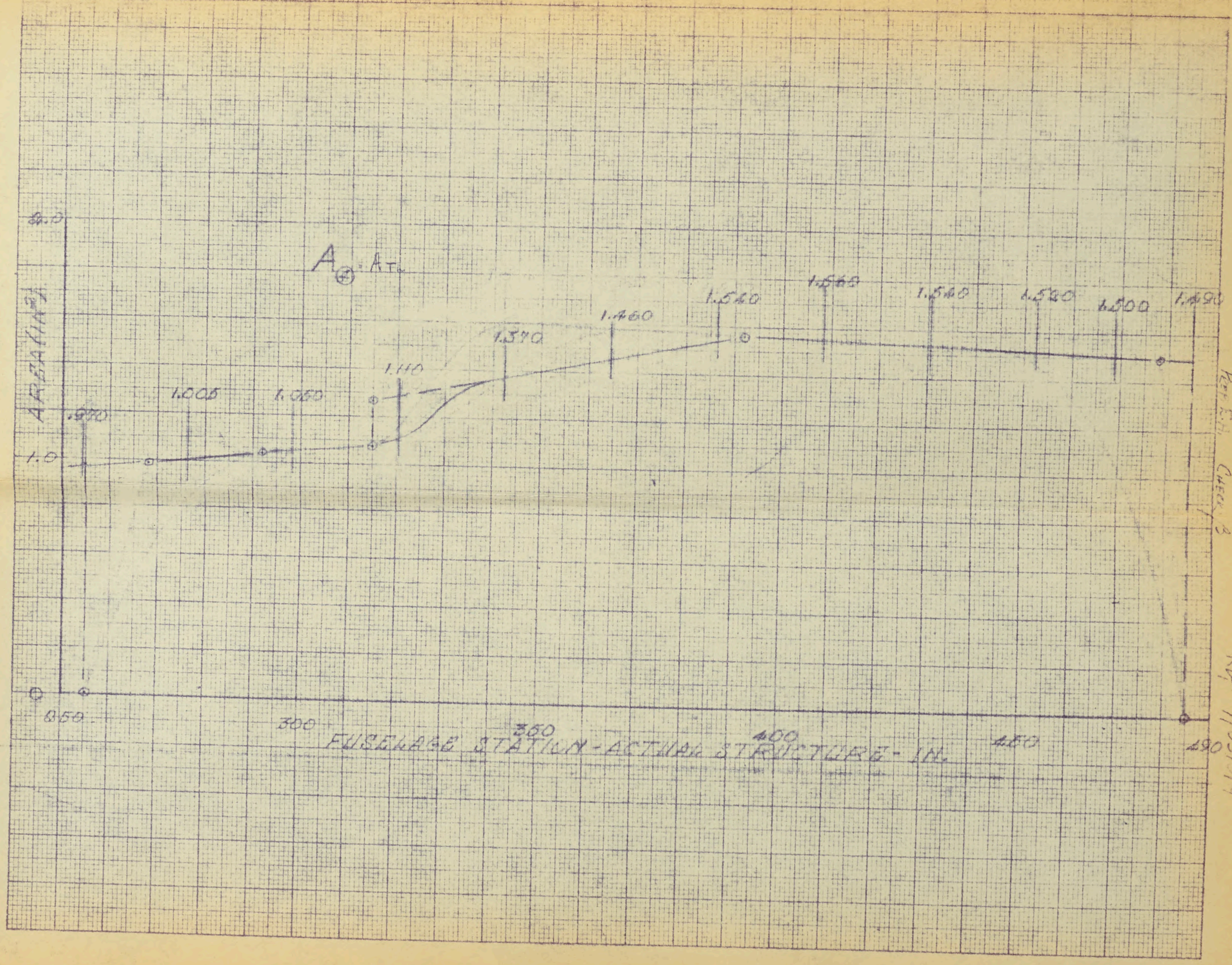
REPT. NO. 7/0510/9
SHEET NO. B-27

PREP: E.R. CHECK: B.




$A_{③} = A_{TU}$
AREA (IN.²)
2.0
1.0
0
1.700
1.180
1.020
1.110
1.505
1.610
1.580
1.560
1.540
1.530
1.520
1.500
250
300
350
400
450
FUSELAGE STATION - ACTUAL STRUCTURE - IN.





Prep E.H. Check B

Rq 7/05/0/9

REPORT NO 7/0510/9
SHEET NO 8 29

AIRCRAFT: C105
CENTRE FUSELAGE

PREPARED BY: C.B. DATE: May 31. 55
CHECKED BY: EO DATE:

LONGERONS:

$$2V = \frac{1}{E}\int \frac{(S_x)^2}{A_x}\,dx$$

FOR LINEAR VARIATION OF S_x & A_x
(GEN/1090/311 SHT 902)

$$2V \doteq \frac{L}{E}\left| \frac{S_i^2}{3\left(\frac{3}{4}A_i + \frac{1}{4}A_k\right)} + \frac{S_i S_k}{3\left(\frac{1}{2}A_i + \frac{1}{2}A_k\right)} + \frac{S_k^2}{3\left(\frac{1}{4}A_i + \frac{3}{4}A_k\right)} \right|$$

$$2V = \frac{L}{E}\left| \frac{\sigma_i^2 A_i^2}{3\left(\frac{3}{4}A_i + \frac{1}{4}A_k\right)} + \frac{2\sigma_i\sigma_k A_i A_k}{3(A_i + A_k)} + \frac{\sigma_k^2 A_k^2}{3\left(\frac{1}{4}A_i + \frac{3}{4}A_k\right)} \right|$$

CLK COEFFICIENTS

	σ_1	σ_2	σ_3	σ_4
σ_1	$\frac{L_{12}}{3E}\cdot\frac{A_1^2}{\left(\frac{3}{4}A_1 + \frac{1}{4}A_2\right)}$	$\frac{L_{12}}{3E}\cdot\frac{A_1 A_2}{(A_1 + A_2)}$		
σ_2	$\frac{L_{12}}{3E}\cdot\frac{A_1 A_2}{(A_1 + A_2)}$	$\frac{L_{12}}{3E}\cdot\frac{A_2^2}{\left(\frac{1}{4}A_1 + \frac{3}{4}A_2\right)}$ $\frac{L_{23}}{3E}\cdot\frac{A_2^2}{\left(\frac{3}{4}A_2 + \frac{1}{4}A_3\right)}$	$\frac{L_{23}}{3E}\cdot\frac{A_2 A_3}{(A_2 + A_3)}$	
σ_3		$\frac{L_{23}}{3E}\cdot\frac{A_2 A_3}{(A_2 + A_3)}$	$\frac{L_{23}}{3E}\cdot\frac{A_3^2}{\left(\frac{1}{4}A_2 + \frac{3}{4}A_3\right)}$ $\frac{L_{34}}{3E}\cdot\frac{A_3^2}{\left(\frac{1}{4}A_4 + \frac{3}{4}A_3\right)}$	$\frac{L_{34}}{3E}\cdot\frac{A_3 A_4}{(A_3 + A_4)}$
σ_4			$\frac{L_{34}}{3E}\cdot\frac{A_3 A_4}{(A_3 + A_4)}$	$\frac{L_{34}}{3E}\cdot\frac{A_4^2}{\left(\frac{1}{4}A_3 + \frac{3}{4}A_4\right)}$

$$\frac{4L}{3E} \quad \frac{4 \times 22.001}{3 \times 10,600} = \qquad \frac{L}{3E} = .000\,691\,823\,90$$

$$\frac{4L}{3E} \quad \frac{4 \times 16.215}{3 \times 10,600} = .002\,044\,025\,156 \qquad \frac{L}{3E} = .000\,511\,006\,289$$

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO 7/0510/9
SHEET NO 8-30

PREPARED BY	DATE
W. Grever	June 13/55
CHECKED BY	DATE

AIRCRAFT: C 105 — Centre Fuselage

Webs:

$$2V = \frac{A \cdot t}{G}\gamma^2 \qquad G = 3{,}200$$

Tank Side:

$A_{TW} = 22 \times 36 = 792$ sq.in

or

$16.25 \times 36 = 585$ sq.in

$t = .0192$ or $.032$ Drawing: Fig 1

$\therefore A_{TW}\,t = 15.2064$ & 25.344

or

$= 18.72$

Outer Side:

$A_{SW} = 22 \times 45 = 990$ sq.in

or

$= 16.25 \times 45 = 731.25$ sq.in

$t = .024$ or $.0306$ or $.040$

$\therefore A_{SW}\,t = 23.76$ & 30.294

or

$= 17.55$ & 29.25

Coeff's:

		22 IN		16.25 IN	
		.0192	.032	.032	
2V =	Tank	4.75200×10^{-3}	7.92000×10^{-3}	5.85000×10^{-3}	
		.024	.0306	.024	.040
	Side	7.42500×10^{-3}	9.466875×10^{-3}	5.484375×10^{-3}	9.140625×10^{-3}

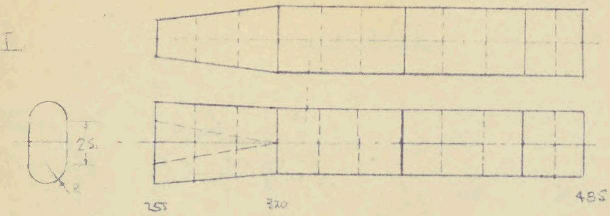
AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO 7/0510/9			
SHEET NO 9 - 0			
AIRCRAFT C105	Centre Section	PREPARED BY CB	DATE May 30 '55
		CHECKED BY	DATE

Duct

Matl .032 24ST.



Sth	R & S1	Bend S.P.	Warp S.P.	Torsion S.P.	Bend. Cir Coeff.	Warping Cir Coef	Torsion Cir Coef	Warping Torsion Cir Coef
254.500	12.24 in	106	105		.007 841	.002 796		
	8.88 in			107	-.005 969	-.001 022	.019 807	±.000 945
276.500		118	117		.015 462	.005 707		
	Linear Variation			119	-.005 811	-.001 029	.019 846	±.000 524
298.500		130	129		.015 027	.005 799		
				131	-.005 624	-.001 036	.019 885	±.000 159
320.500	18.000	141 206	140 205		.007 376 .007 914	.002 913 .002 918		
	0			207	-.006 036	-.001 040	.019 905	0
342.500	18.000	218	217		.015 828	.005 836		
	0			219	-.006 036	-.001 040	.019 905	0
364.500	18.000	230	229		.015 828	.005 836		
	0			231	-.006 036	-.001 040	.019 905	0
386.500	18.000	241 306	240 305		.007 914	.002 918		
	0			307	-.006 036	-.001 040	.019 905	0
408.500	18.000	318	317		.015 828	.005 836		
	0			319	-.006 036	-.001 040	.019 905	0
430.500	18.000	330	329		.015 828	.005 836		
	0			331	-.006 036	-.001 040	.019 905	0
452.500	18.000	341 406	340 405		.007 914	.002 918 .003 180		
	0			407		-.001 793	.014 703	0
468.750	18.000							
	0							
485.000	18.000							
	.0							
Ref Page					9-16 & 9-49	9-43 & 9-50	9-19-1 & 9-49	9-46

FORM 1319A

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT No 7/0510/9

SHEET No 9-1

AIRCRAFT: C105

CENTRE FUSELAGE

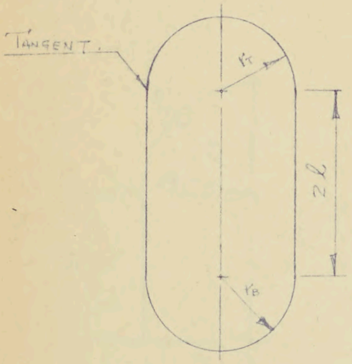
PREPARED BY: C.B.

DATE: May 2. 55

CHECKED BY:

DATE: June 55

DUCT



$r_T \doteqdot r_B$

t = .032 24ST CL4

S_{TN}	$r_{AVG.}$	$2l$
255.0	12.24	17.76
268.	13.3	14.5
281.	14.3	11.4
292.	15.25	8.2
303.	16.15	5.3
315.	17.1	2.25
331 & AFT.	18.00 INSIDE	0

Approximate only

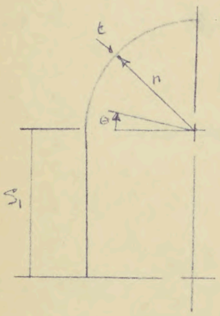
REF: S. YOUNG.

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0510/9			
SHEET No. 9-2			
AIRCRAFT: C105	CENTRE FUSELAGE	PREPARED BY: C.B.	DATE: May '55
		CHECKED BY: J F Gardener	DATE: June 55

DUCT BENDING.



$$\sigma = \frac{\sigma_b x}{(S_1+r)L}\left(s \;,\; S_1 + r \sin\theta \right)$$

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

Timoshenko Plates & Shells P. 384

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

ON STRAIGHT PORTION

ON CURVED PORTION

$$\frac{\partial\tau}{\partial s} = \frac{\partial\tau}{r\,\partial\theta} = -\frac{\sigma_b}{(S_1+r)L}\left(s \;,\; S_1 + r \sin\theta \right)$$

$$\tau_s = -\frac{\sigma_b}{(S_1+r)L}\left(\frac{s^2}{2} + C_1\right) ; \quad \tau_\theta = -\frac{\sigma_b}{(S_1+r)L}\left(S_1 r\theta - r^2\cos\theta + C_2 \right)$$

$\tau = 0$ @ $\theta = 90^\circ$ $\quad C_2 = -S_1 r \frac{\pi}{2}.$

$$\tau_s = -\frac{\sigma_b}{(S_1+r)L}\left(\frac{s^2}{2} + C_1\right) \qquad \tau_\theta = -\frac{\sigma_b}{(S_1+r)L}\left(S_1 r\theta - r^2\cos\theta - S_1 r \frac{\pi}{2}\right)$$

SET EQUAL AT $s = S_1$, $\theta = 0^\circ$; THEN $C_1 = -r^2 - S_1 r\frac{\pi}{2} - \frac{S_1^2}{2}$

EVALUATE σ, & τ for $0, \frac{S_1}{3}, \frac{2S_1}{3}, S_1$, 15° 30° 45° 60° 75° 90°.

CALCULATE FRAME FORCE
" BENDING ENERGY
" SHEAR ENERGY
} USING SIMPSON'S & WEDDLE'S RULES

AXIALLY

$$2V = \frac{1}{E}\iint (\sigma_1 + \sigma_2)^2\, t\, ds\cdot dx = \frac{t}{E(S_1+r)^2}\iint\left[\sigma_1\left(1-\frac{x}{L}\right) + \sigma_2\frac{x}{L}\right]^2 y^2 \cdot ds\cdot dx$$

$$= \frac{tL}{3E(S_1+r)^2}\left[\sigma_{b_1}^2 + \sigma_{b_1}\sigma_{b_2} + \sigma_{b_2}^2\right]\int_0^S y^2\, ds$$

where y is the amount shown in the bracket above.

Duct Bending.

Shear

$$2V = \frac{1}{G}\iint (\tau_1 - \tau_2)^2 t\, ds\, dx = \frac{t}{G}\iint \left[\sigma_1^2 - 2\sigma_1\sigma_2 + \sigma_2^2\right]\frac{\bar{y}^2}{(S_1+r)^2 L^2}\, ds\, dx$$

$$= \left[\sigma_1^2 - 2\sigma_1\sigma_2 + \sigma_2^2\right]\frac{t}{(S_1+r)^2 G L}\int \bar{y}^2\, ds.$$

Frame Load:

$$\partial P = \tau \cdot t \cdot ds \cdot \cos\Theta.$$

$$P = \int \tau \cdot t \cos\Theta \cdot ds,$$

$$= \frac{-\sigma_b t}{(S_1+r)L}\int_0^{S_1}\left(\frac{S^2}{2} - r^2 - \frac{S_1 r \pi}{2} - \frac{S_1^2}{2}\right)(1{,}000\,000)\, ds.$$

$$+ \frac{-\sigma_b t}{(S_1+r)L}\int_0^{\pi/2}\left(S_1 r\Theta - r^2\cos\Theta - \frac{S_1 r \pi}{2}\right)\cos\Theta\; r\, d\Theta$$

where $\sigma_b = (\sigma_{b_1} - \sigma_{b_2})$

Duct Bending - Sta 255.00

S_1 = 8.88000; S_2 = 19.226 547 S = 28.106 55 r = 12.240 000

	1	2	3	4	5	6	7	8	9
s	θ	Sin θ	Cos θ	S_j; $S_1 + r\sin\theta$	$\frac{S^2}{2} - r^2 - S_1 r\frac{\pi}{2} - \frac{S_1^2}{2}$ or $S_1 r\theta - r^2\cos\theta - S_1 r\frac{\pi}{2}$	③⑤	$④^2$	$⑤^2$	Coeff. [illegible]
0	.000 000	.000 000	1.000 000	0	-359.976 556	-359.976 556	.000 000	129,583. 120 870	+1.000 000
2.960 000	.000 000	.000 000	1.000 000	2.960 000	-355.595 756	-355.595 756	8.761 600	126,448. 341 685	3
5.920 000	.000 000	.000 000	1.000 000	5.920 000	-342.453 356	-342.453 356	35.046 400	117 274. 301 036	3
S_1 = 8.880 000	.000 000	.000 000	1.000 000	8.880 000	-320.549 356	-320.549 356	78.854 400	102 751. 889 632	+1. 1.
12.034 424	.261 799	.258 819	.965 926	12.047 945	-286.989 224	-277.210353	145.152 979	82 362. 814 692	5.
15.288 848	.523 599	.500 000	.866 025	15.000000	-243,566 939	-210.935088	225,000 000	59 324. 853 774	1.
18.493 272	.785 398	.707 107	.707 107	17.534 990	-191,302 579	-135.271 676	307.475 874	36 596. 829 774	6.
21.697 696	1.047 198	.866025	.500 000	19.480 146	-131.819 342	-65.909614	379.476 088	17 376. 340 771	1.
24.902 120	1.308 997	.965 926	.258 819	20.702 934	-67.230 943	-17.400 645	428.611 476	4 519. 999 697	5.
S = 28.10655	1.570 796	1.000 000	.000 000	21.120 000	.000000	.000 000	446.054 400	.000 000	1.
						1.110000		$\frac{1}{8}$ × 8.880	000
						.961 3273		19.226547	
								30	
								To	TA..

FORM 1543

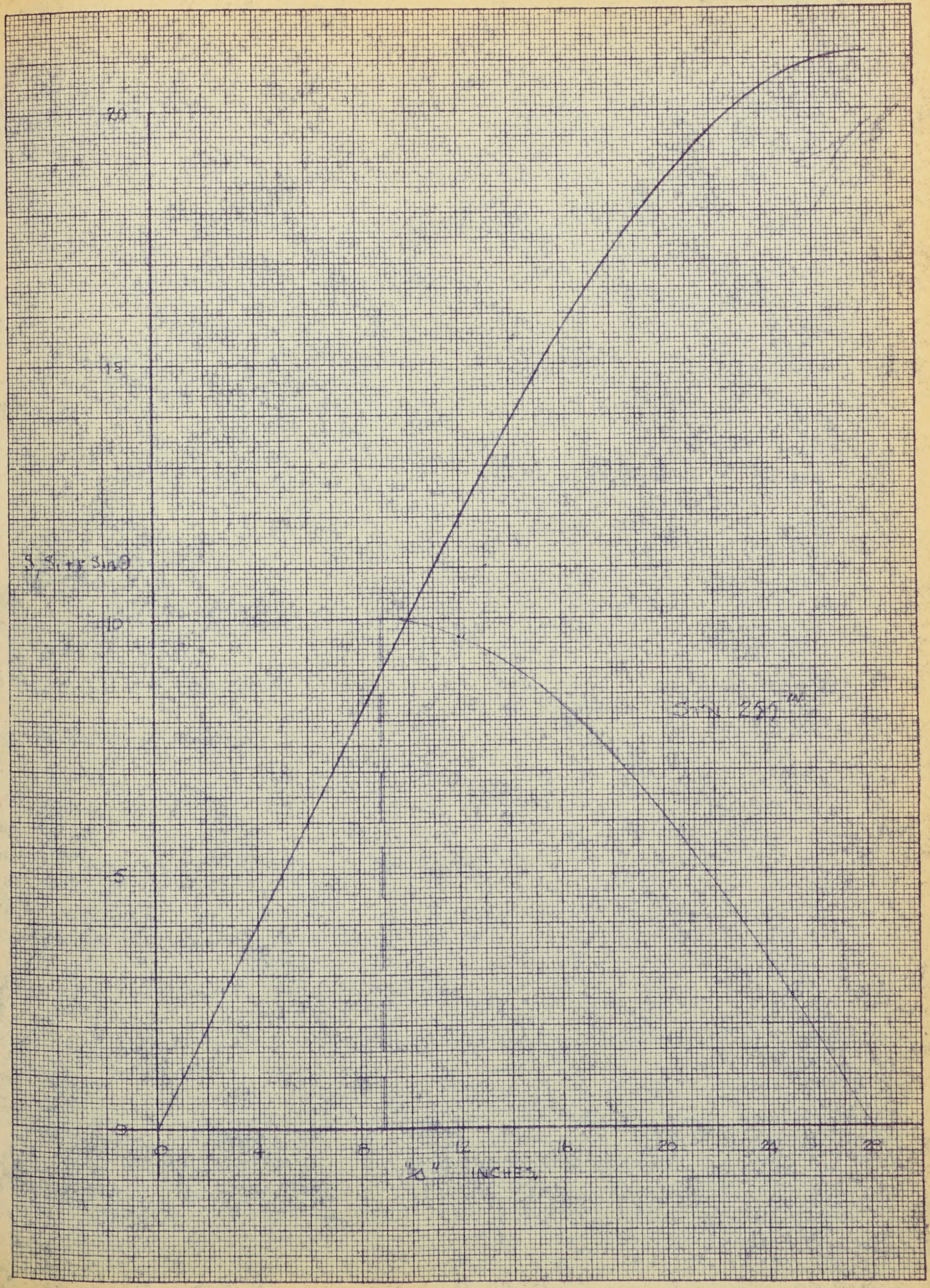
A. V. ROE CANADA LIMITED
MALTON, ONTARIO
TECHNICAL DEPT. (AIRFRAME)

AIRCRAFT . . C105
WEIGHT . .
C. G. POSITION . .

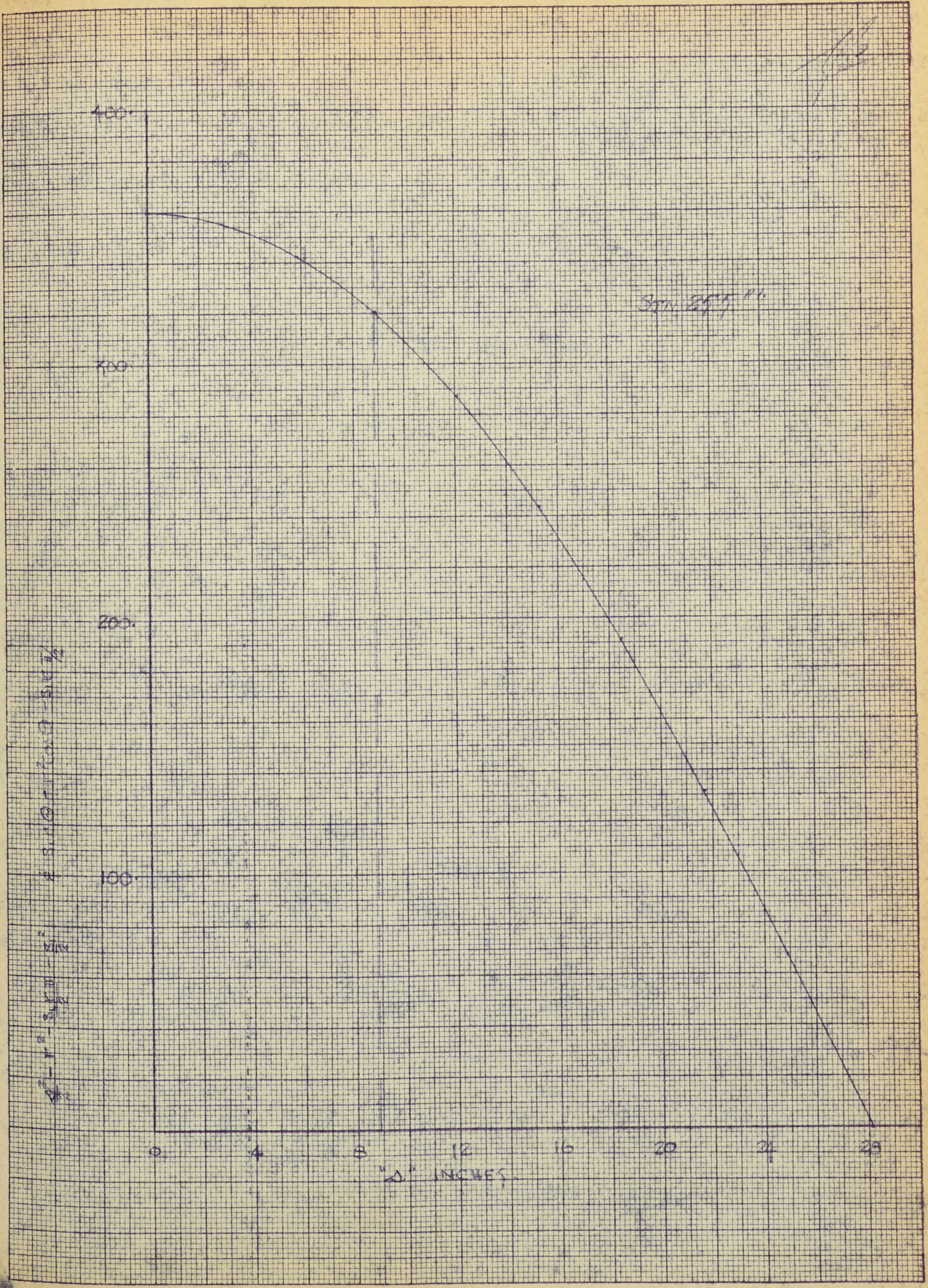
REPORT NO. . 7/0510/9
SHEET . . 9-4
DATE . . May 18 55
PREPARED BY . C.E.

6	7	8	9	10	11	12	13	14	15	16	17	18	19
③ ⑤	$④^2$	$⑤^2$	Coeff. [illegible]	⑥ ⑨	⑦ ⑨	⑧ ⑨					Unit f_b Notched	Unit f_s Notched	
59.976 556	.000 000	129,583.120 870	+1.000 000	-359.976 556	.000 000	129,583.120 870					0	-.775	
355.595 756	8.761 600	126,448.341 685	3	-1066.787 268	26.284 800	379,345.025 035					.140	$\overline{.766}$	
342.453 356	35.046 400	117 274.301 036	3	-1027.360 068	105.139 200	351 822.903 108					.280	$\overline{.727}$	
-320.549 356	78.854 400	102 751.989 632	+1. / 1.	-320.549 356	78.854 400	102 751.889 632					.420	.690	
277.210 353	145.152 979	82 362.814 692	5.	-1386.051 765	725.764 89	411 814.073 460					.570	.599	
-210.935 088	225.000 000	59 324.853 774	1.	-210.935 058	225.000 000	59 324.853 774					.710	$\overline{.524}$	
-135.271 676	307.475 874	36 596.829 774	6.	-811.630 056	1844.855 244	219 580.978 644					.830	$\overline{.412}$	
-65.909 674	379.476 088	17 376.340 771	1.	-65.909 674	379.476 088	17 376.340 771					.922	$\overline{.284}$	
-17.400 645	428.611 476	4 519.999 697	5.	-87.003 225	2143.057 38	22 599.998 481					.980	$\overline{.145}$	
.000 000	446.054 400	.000 000	1.	.000 000	446.054 400	.000 000					1.000	0	
1.110 000		$\frac{1}{8}$ × 8.880	000 ×	-2 774.673 248	210.278 400	963 202.938 665							
.961 3273		$\frac{19.226\,547}{30}$ ×		-2 882.079 134	5843.062 347	833,448.134 766							
			TOTAL	-5 850.508 658	5 850.504 085	1 810 704.7071							

7/0610/3
9-5



7/05/019
9-6



AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C105	Centre Fuselage	REPORT NO 7/0510/9	
		SHEET NO 9-7	
		PREPARED BY	DATE
		C B.	May 18 55
		CHECKED BY	DATE
		J. Gardner	June 55

Duct Bending Stn 255.000 in.

Axial Energy $= \dfrac{tL}{3E(S_1+r)^2}\left[\sigma_{b_1}^2 + \sigma_{b_1}\sigma_{b_2} + \sigma_{b_2}^2\right]\int y^2\,ds.$

$$= \frac{.032 \times 22.000 \times 5850.504\,095}{3 \times 10{,}600 \times 21.12^2}\left[\sigma_{b_1}^2 + \sigma_{b_1}\sigma_{b_2} + \sigma_{b_2}^2\right]$$

$$= .000\,290\,37\left[\sigma_{b_1}^2 + \sigma_{b_1}\sigma_{b_2} + \sigma_{b_2}^2\right]$$

Shear Energy $= \dfrac{t}{GL(S_1+r)^2}\int \bar{y}^2\,ds\left[\sigma_{b_1}^2 - 2\sigma_{b_1}\sigma_{b_2} + \sigma_{b_2}^2\right]$

$$= \frac{.032 \times 1\,870\,704.7071}{4\,000 \times 22.0 \times 21.12^2}\left[\sigma_{b_1}^2 - 2\sigma_{b_1}\sigma_{b_2} + \sigma_{b_2}^2\right]$$

$$= .001\,525\,05\left[\sigma_{b_1}^2 - 2\sigma_{b_1}\sigma_{b_2} + \sigma_{b_2}^2\right]$$

Total 2V $= 4\left(.000\,290\,37\left[\sigma_{b_1}^2 + \sigma_{b_1}\sigma_{b_2} + \sigma_{b_2}^2\right] + .001\,525\,05\left[\sigma_{b_1}^2 - 2\sigma_{b_1}\sigma_{b_2} + \sigma_{b_2}^2\right]\right)$

$$= \underline{.007\,261\,68\left[\sigma_{b_1}^2 + \sigma_{b_2}^2\right] - .005\,519\,47 \times 2\,\sigma_{b_1}\sigma_{b_2}}$$

Frame Load (From One Side)

$$P = -\frac{\sigma_b t}{(S_1+r)L}\int \quad = \frac{\sigma_b \times .032 \times 5850.508\,658}{21.12 \times 22.000} \times 4$$

$$= \underline{1.611\,711\,\sigma_b}$$ Total Down on Duct For Fwd & Aft Frame

$$= \underline{3.223\,421\,\sigma_b}$$ Total Up on Duct For Centre Frame

FORM 1319A

A. V. ROE C
MALT
TECHNICA

AIRCRAFT ·
WEIGHT ·
C. G. POSITION

Duct Bending – Midway between Sta's 285.0" & 320"

S_1 = 4.440 00" S_2 = 23.750 443 S = 28.904"

r = 15.120 00"

	1	2	3	4	5	6	7	8	9	10	11
s	θ	Sin θ	Cos θ	s, S_1+r Sin θ	$\frac{s^2}{2}-r^2-\frac{S_1 s}{2}+\frac{S_1^2}{2}$ or $S_1\theta-r^2\cos\theta-\frac{S_1 r\pi}{2}$	(3)(5)	$(4)^2$	$(5)^2$	Coeff 3/8 Rule & Weddle	(6)(9)	(7)(9)
0.0	0	0	1.000	0	-343.923 160	-343.923 160	0	118 28· 139 98	1	- 343· 923 160	0
1.480 000	0	0	1.000	1.480 000	-342.827 760	-342.827 760	2.190 400	117 53 010 15	3	- 1 028· 483 880	6· 571 200
2.960 000	0	0	1.000	2.960 000	-339,542 360	-339,542 360	8.761 600	115 289· 014 234	3	-1018· 627 080	26· 284 800
S_1 = 4.440 000	0	0	1.000	4.440 000	-334.066 360	-334.066 360	19.713 600	111 600 332 8	1	- 334· 066 360	19· 713 600
8.398 407	.261 799	.258 819	.965 926	8.753 343	-308.701 253	- 298· 182 566	69.778 339	95 296· 463 6	5	- 1 490· 912 800	348· 891 695
12.356 814	.523 599	.500 000	.866 025	12.000 000	-268.297 079	- 232· 343 318	144.000 000	71 97· 956 7	1	- 232· 343 318	144· 000 000
16.315 221	.785 398	.707 107	.707 107	15.131 458	- 214.780 876	- 151· 590 190	228.961 021	45 959· 142 84	6	- 909· 541 140	1 373· 766 126
20.273 629	1.047 198	.866 025	.500 000	17.534 298	-149.457 826	- 74· 728 913	307.451 606	22 337· 641 758	1	- 74· 728 913	307· 451 606
24.232 036	1.308 997	.965 926	.258 819	19.044 801	- 76.745 077	- 19· 863 084	362.704 445	5 889· 806 844	5	- 99· 315 420	1 813· 522 225
S = 28.190 443	1.570 796	1.000 000	.000 000	19.560 000	.000 026	0	382.593 600	0·	1	0	382· 593 600
					(.555 000)	$\frac{1}{8}$	× 4.44			- 2 725· 100 480	52· 569 600
					(1.187 522)		$\frac{23.750\ 443}{20}$			- 3, 140· 907 951	4 359· 938 85
							TOTAL			- 5 242· 328 058	5 242· 325 093

FORM 1543

Span 2550 in & 320

50 443 S: 28.704 in

A. V. ROE CANADA LIMITED

MALTON, ONTARIO

TECHNICAL DEPT. (AIRFRAME)

AIRCRAFT . . C105

WEIGHT . .

C. G. POSITION .

REPORT NO. . 7/0510/9

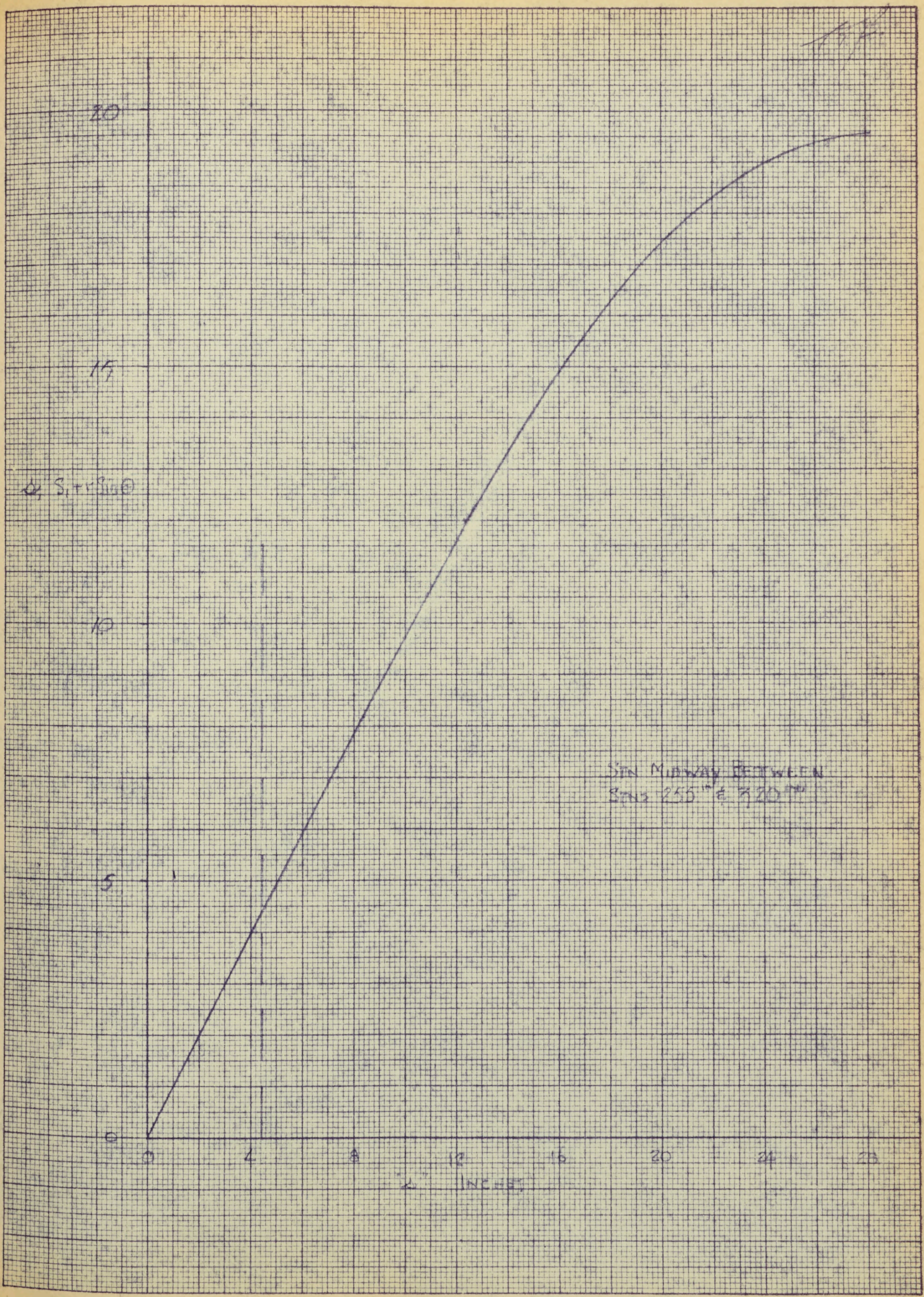
SHEET . . 9-3

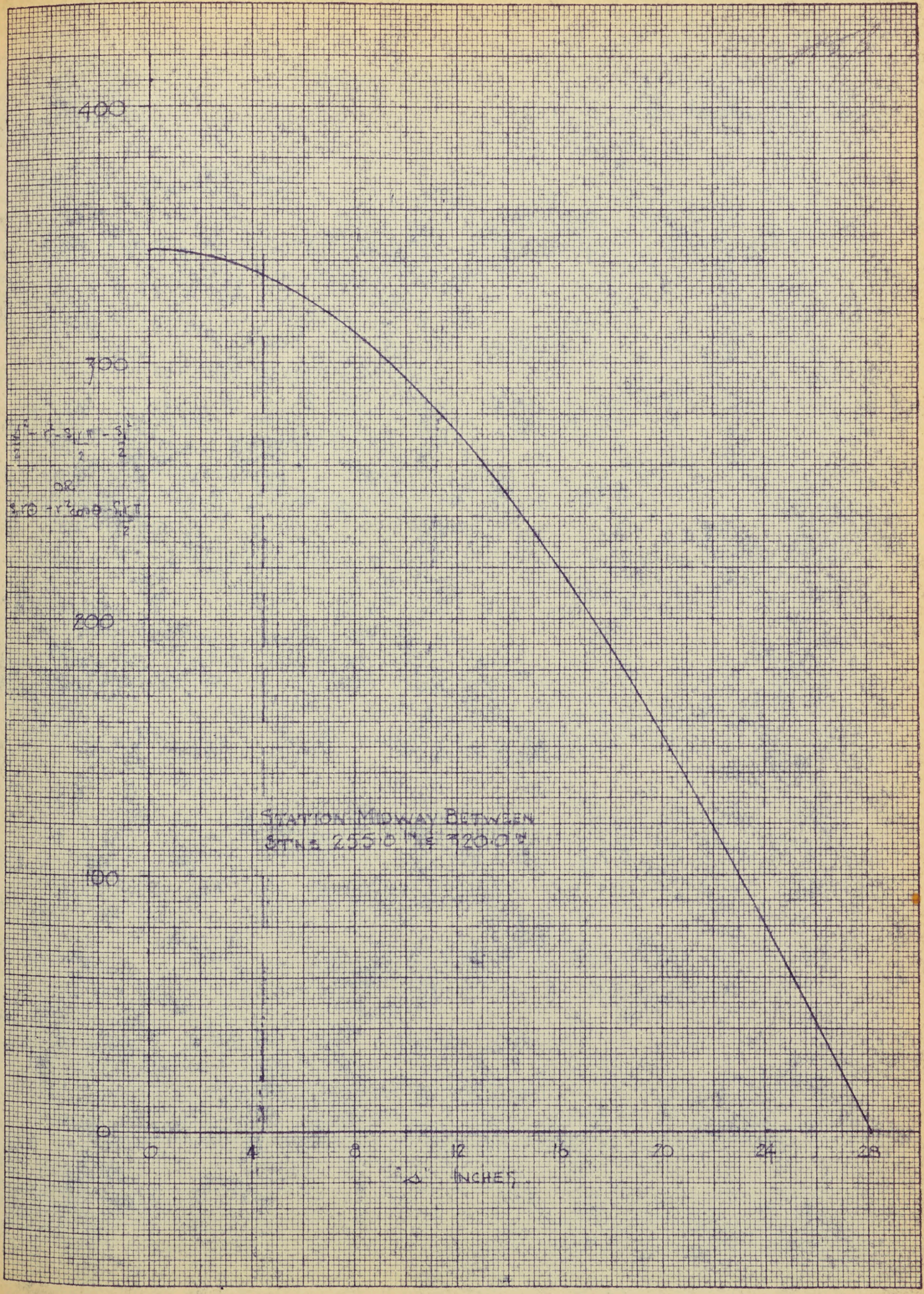
DATE . . May 20/55

PREPARED BY . C.B.

6	7	8	9	10	11	12	13	14	15	16	17	18	19
(3)(5)	(4)²	(5)²	Co[illegible] ³/₈ [illegible] & Web[illegible]	(6)(9)	(7)(9)	(8)(9)						N[illegible] J₀	C[illegible] J₂
- 343 923 160	0	118 283·139 984	1	- 343·923 160	0	118 283·139 984						0	$\overline{1}$.074
-342.827 960	2.190 400	117 53[illegible]·010 153	3	- 1 028·483 880	6·571 200	352 593·030 474							
-339.542 360	8.761 600	115 28[illegible]·014 234	3	-1018·627 080	26·284 800	345 867·042 670						.151	$\overline{1}$.060
- 334.066 360	19.713 600	111 600·332 884	1	- 334·066 360	19·713 600	111 600·332 884							
- 298·182 566	69.778 339	95 20[illegible]·463 6[illegible]	2	- 1 490·912 800	348·891 695	476 48[illegible]·318 020						.427	$\overline{.931}$
- 232·343 318	144.000 000	71 97[illegible]·956 7[illegible]	1	- 232·343 318	144·000 000	71 977·956 758							
- 151·590 190	228.361 021	45 959·142 8[illegible]	6	- 909·541 140	1 373·766 126	275 754·857 064						.774	$\overline{.473}$
- 74·728 913	307.451 606	22 337·641 753	1	- 74·728 913	307·451 606	22 337·641 753						.896	
- 19·863 084	362.704 445	5 889·806 844	5	- 99·315 420	1 813·522 225	29 447·034 220							$\overline{.062}$
0	382.593 600	0·	1	0	382·593 600	0						1.00	0
1/8	4·440 [illegible]			- 2 725·100 480	52·569 600	928 343·546 012							S₁ + [illegible]
	23·750 443 / 20			-3,140·907 951	4 359·938 852	987 602·140 606							[illegible]
	TOTAL			- 5 242·328 058	5 242·325 093	1 688 029·937 360							

7/05/0/9
9-9


20
15
Q, $S_1 + r \sin\theta$
10
5
0
0
4
8
12
16
20
24
28
"Z" INCHES
STN MIDWAY BETWEEN
STNS 255TH & 320TH


400
300
OR
200
STATION MIDWAY BETWEEN
STNS 255·0 INS & 320·0 INS
100
0
0
4
8
12
16
20
24
28
s' INCHES

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0510/9

SHEET No. 9 - 11

AIRCRAFT: C105	Centre Fuselage	PREPARED BY C.B.	DATE May 20. 55
		CHECKED BY J.H. Gardner	DATE June 55

Duct Bending - Midway

$$\text{Axial Energy} = \frac{.032 \times 22.000 \times 4 \times 5,242.325\,093}{3 \times 10,600 \times 19.56^2}\left[\sigma_{b_1}^2 + \sigma_{b_1}\sigma_{b_2} + \sigma_{b_1}^2\right]$$

$$= .001\,213\,37\left[\sigma_{b_1}^2 + \sigma_{b_1}\sigma_{b_2} + \sigma_{b_2}^2\right]$$

$$\text{Shear Energy} = \frac{.032 \times 4 \times 1,658\,029.937\,358}{4\,000 \times 19.56^2 \times 22.000}\left[\sigma_{b_1}^2 - 2\sigma_{b_1}\sigma_{b_2} + \sigma_{b_2}^2\right]$$

$$= .006\,417\,56\left[\sigma_{b_1}^2 - 2\sigma_{b_1}\sigma_{b_2} + \sigma_{b_2}^2\right]$$

$$\text{Total: } 2V = \underline{.007\,630\,92\left[\sigma_{b_1}^2 + \sigma_{b_2}^2\right] - .005\,810\,88 \times 2\,\sigma_{b_1}\sigma_{b_2}}$$

$$\text{Frame Load: } P = \frac{\sigma_b \times .032 \times 4 \times 5,242.328\,058}{19.56 \times 22.000}$$

$$= \underline{1.559\,346\,51\;\sigma_b}$$ Total - Down on Duct For Fwd & Aft Frames.

$$\underline{3.118\,693\,02\;\sigma_b}$$ Total - Up on Duct For Centre Frame

FORM 1319A

Duct Bending

Circular Portion

$S_1 = 0.00$ $S_2 = 28.274337 = S$ $r = 18.000$

	1	2	3	4	5	6	7
s	Θ	$\sin\Theta$	$\cos\Theta$	$r\sin\Theta$	$-r^2\cos\Theta$	③ ⑤	$④^2$
0	.000 000	.000 000	1.000 000	.000 000	-324.000 000	-324.000 000	0
4.712 370	.261 799	.258 819	.965 926	4.658 742	-312.960 024	-302.296 224	21.703 877
9.424 779	.523 599	.500 000	.866 025	9.000 000	-280.592 100	-242.999 773	81.000 000
14.137 168	.785 398	.707 107	.707 107	12.727 920	-229.102 668	-162.000 100	162.000 100
18.849 558	1.047 198	.866 025	.500 000	15.588 450	-162.000 000	-81.000 000	242.999 775
23.561 948	1.308 997	.965 926	.258 819	17.386 668	-83.857 356	-21.703 877	302.296 224
28.274 337	1.570 796	1.000 000	.000 000	18.000 000	0	0	324.000 000
						(1.413 716 85)	

FORM 1543

r = 15 000

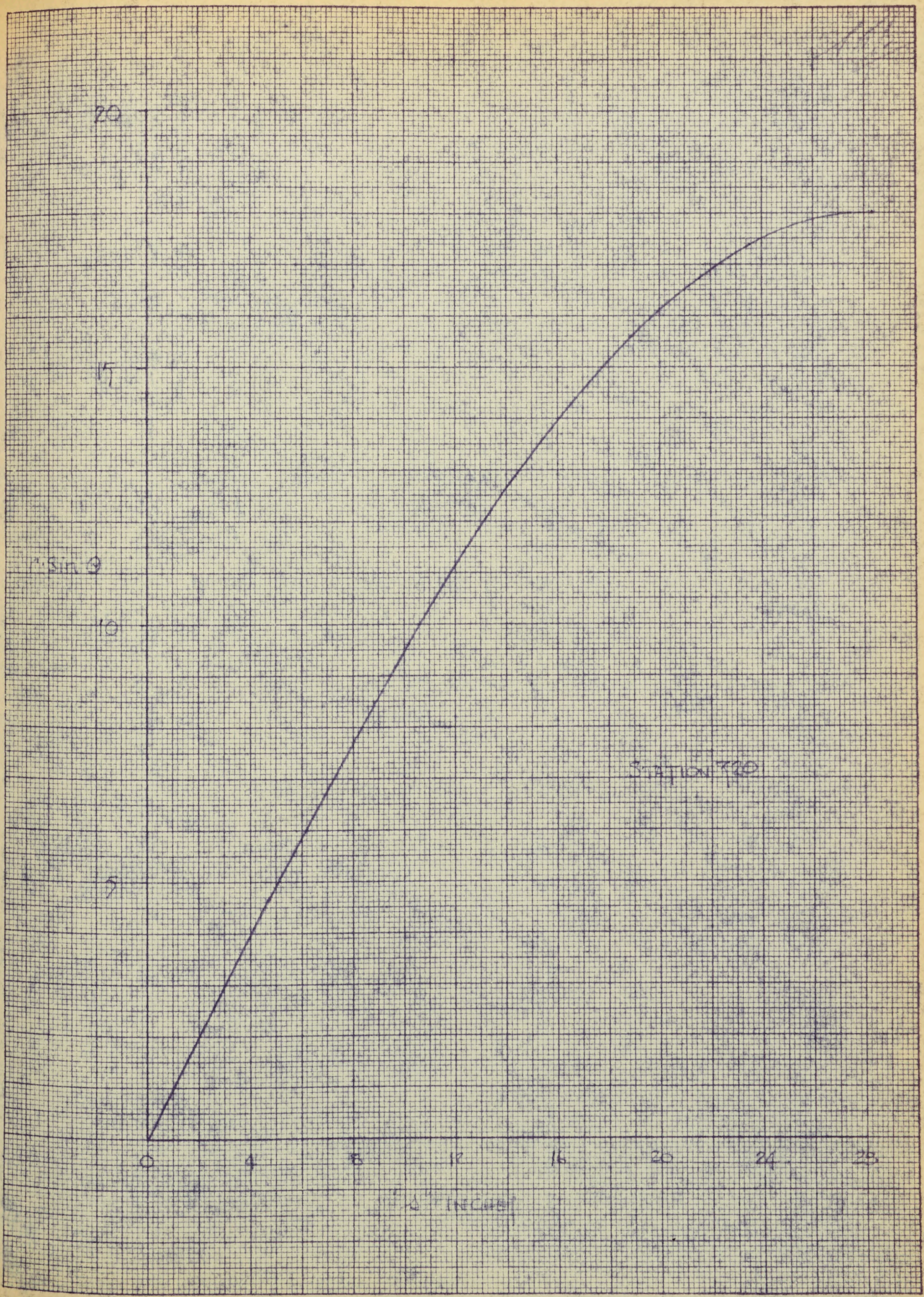
A. V. ROE CANADA LIMITED
MALTON, ONTARIO
TECHNICAL DEPT. (AIRFRAME)

AIRCRAFT - -
WEIGHT - -
C. G. POSITION -

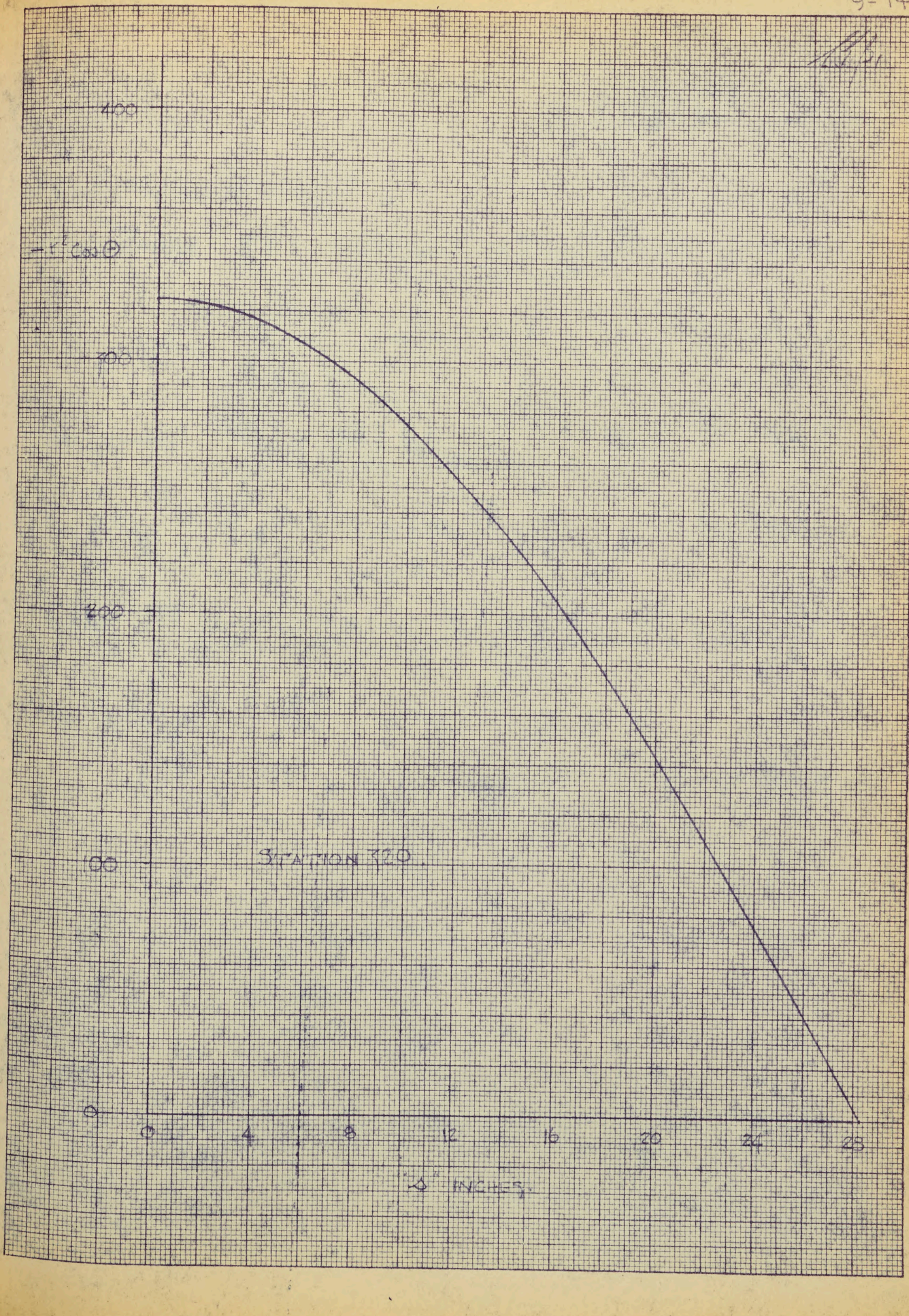
REPORT NO. - 7/[illegible]
SHEET - - 3-12
DATE - - [illegible]
PREPARED BY - [illegible]

	7	8	9	10	11	12	13	14	15	16	17	18	19
(5)	4^2	$(5)^2$	C of F Weights	⑥ ⑨	⑦ ⑨	⑧ ⑨						J_0	T
00 000	0	104 976 000 000	1	− 324 000 000	0	104 976 000 000						0	$\bar{.}218$
224	21 703 877	97 947 976 622	5	− 1 511 481 120	108 519 385	499 719 883 110							
99 773	81 000 000	78 731 920 583	1	− 242 999 773	81 000 000	78 731 926 583						.502	$\bar{.}709$
100	162 000 100	52 48 2 48	6	− 972 000 000	972 000 000	314 919 194 910							
000	242.999 7[illegible]	26 244 000 000	1	− 81 000 000	242 999 773	26 244 000 000						.966	$\bar{.}409$
877	302.296 224	7 042 026 155	5	− 108 519 385	1 511 481 120	35 160 280 775							
	324.000 000	0	1	0	324.000 000	0						1.000	0
3 716 85)		$\frac{28.274\ 337}{20}$		− 3 240 000 278	3 240 000 278	1 043 760 285 377							
				− 4580 442 987	4580 442 987	1 484 063 803 898						$(S_1 + r)L$	= 396

7/0510/9
9-13



7/0510/9


400
$-r^2\cos\Theta$
300
200
100
0
STATION 320
0
4
8
12
16
20
24
28
"α" INCHES.

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0510/9		SHEET No. 9 - 15	
AIRCRAFT: C105	Centre Fuselage	PREPARED BY C.B.	DATE May 20 · 55
		CHECKED BY J. Gardener	DATE June '55

Duct Bending – Station 320

Axial Energy $$2V = \frac{.032 \times 22.000 \times 4 \times 4\,580.442\,987}{3 \times 10{,}600 \times 18.000^2}\left[\sigma_{b_1}^2 + \sigma_{b_1}\sigma_{b_2} + \sigma_{b_2}^2\right]$$

$$= .001\,251\,90 \left[\sigma_{b_1}^2 + \sigma_{b_1}\sigma_{b_2} + \sigma_{b_2}^2\right]$$

Shear Energy $$2V = \frac{.032 \times 4 \times 1\,484\,063.803\,898}{4{,}000.0 \times 18.000^2 \times 22.000}\left[\sigma_{b_1}^2 - 2\sigma_{b_1}\sigma_{b_2} + \sigma_{b_1}^2\right]$$

$$.006\,662\,46 \left[\sigma_{b_1}^2 - 2\sigma_{b_1}\sigma_{b_2} + \sigma_{b_2}^2\right]$$

Total $$2V = .007\,914\,36\left[\sigma_{b_1}^2 + \sigma_{b_1}^2\right] - .006\,036\,51 \times 2\left[\sigma_{b_1}\sigma_{b_2}\right]$$

Frame Load

$$P = -\frac{\sigma_b C}{rL}\int = \frac{4 \times \sqrt{L} \times .032 \times 4{,}580\,442\,987}{18.000 \times 22.000} = 1.480\,547\,24\,\sigma_b$$

Down on Duct for Fwd & Aft Frame

At Centre 2.961 094 48 σ_b up on Duct for Centre Frame

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

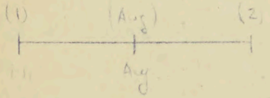
AIRCRAFT: C105	CENTRE FUSELAGE	REPORT No. 7/0510/9 SHEET No. 9-15-1
		PREPARED BY: C.B — DATE: May 24. 55
		CHECKED BY: — DATE: June '55

DUCT

STATION	$\sigma_{b_1}^2+\sigma_{b_2}^2$	$2\cdot\sigma_{b_1}\sigma_{b_2}$	CENTRE FRAME LOAD				
255	.007 26168	−.005 51947	3.223 421 *3.159 477				
MID WAY	.007 63092	−.005 81088	3.118 693 *3.072 035				
320 Apt @ 22"	.007 91436	−.006 03651	2.961 094				
320 Apt @ 16.25	.009 94434	−.008 55752					

HOWEVER, THE DUCT IS NOT UNIFORM ALONG ITS LENGTH.

APPROXIMATELY – $2V = \frac{C_1}{3}\sigma_1^2 + \frac{C_{Avg}}{3}\sigma_1\sigma_2 + \frac{C_2}{3}\sigma_2^2 + D_{Avg}(\sigma_1-\sigma_2)^2$



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

& D is the coefficient of shear energy.

+ FROM NEXT SECTION

$$\begin{array}{c|cc} & \sigma_1 & \sigma_2 \\ \hline \sigma_1 & \frac{C_1}{3}+D_{Avg} & \frac{C_{Avg}}{6}-D_{Avg} \\ \sigma_2 & \frac{C_{Avg}}{6}-D_{Avg} & \frac{C_2}{3}+D_{Avg} \end{array}$$

+ FROM NEXT SECTION

* COMPUTE INTERMEDIATE VALUES BY $y = \alpha x^2 + \beta x + \gamma = 0$.

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO 7/0510/9
SHEET NO 9 · 12 · 2

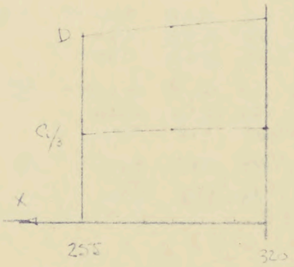
AIRCRAFT C105
CENTRE FUSELAGE

PREPARED BY [illegible] DATE MAY 31 58
CHECKED BY β DATE [illegible]

DUCT BENDING

VALUES OF C_i & D_i

STN	$C_i/3$	D
255	.001 161 48	.006 100 20
MIDWAY	.001 213 37	.006 417 56
320	.001 251 90	.006 662 46



$\frac{C_i}{3} = \alpha x^2 + \beta x + \gamma$

$.001\,251\,90 = 0 + 0 + \gamma$
$.001\,213\,37 = 9\alpha L^2 + 3\beta L + \gamma$
$.001\,161\,48 = 36\alpha L^2 + 6\beta L + \gamma$

$-.000\,038\,53 = 9\alpha L^2 + 3\beta L$
$-.000\,090\,42 = 36\alpha L^2 + 6\beta L$
$\therefore 6\beta L = -.000\,063\,70$
$\beta L = -.000\,010\,6200$

$18\alpha L^2 = -.000\,013\,36$
$\alpha L^2 = -.000\,000\,742\,2$

$D = \alpha x^2 + \beta x + \gamma$

$.006\,662\,46 = 0 + 0 + \gamma$
$.006\,417\,56 = 9\alpha L^2 + 3\beta L + \gamma$
$.006\,100\,20 = 36\alpha L^2 + 6\beta L + \gamma$

$-.000\,244\,90 = 9\alpha L^2 + 3\beta L$
$-.000\,562\,26 = 36\alpha L^2 + 6\beta L$
$\therefore 6\beta L = -.000\,417\,34$
$\beta L = -.000\,069\,560$

$18\alpha L^2 = -.000\,072\,46$
$\alpha L^2 = -.000\,004\,026$

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO 7/0511/19

SHEET NO 9-16

AIRCRAFT: C 105	Centre Fuselage	PREPARED BY C.B. & W.S.	DATE May 31/55
		CHECKED BY B	DATE June 55

Duct Bending

nL	Ci/3	D	Coefficient		STN
0	.001 251 90	—	.007 840 77		302.5
L	.001 240 54	.006 588 87	−.005 968 60		Cross
2L	.001 227 69	—	.007 816 56 .007 645 25	.015 461 81	298.5
3L	.001 213 36	.006 417 55	−.005 810 87		Cross
4L	.001 197 55		.007 615 10 .007 411 56	.015 026 66	276.5
5L	.001 180 25	.006 214 01	−.005 623 89		Cross
6L	.001 161 47		.007 375 48		254.5

Duct-Torsion: See P. 9-19

x	$2Y = x'Tdl^2$	K
0	.019 904 92	.019 904 92
L		.019 885 29
3L	.019 846 00	.019 846 00
5L		.019 806 67
6L	.019 787 00	.019 787 00

$K = \alpha x^2 + \beta x + \gamma$

$.019\ 904\ 92 = 0 + 0 + \gamma$

$.019\ 846\ 00 = 9\alpha L^2 + 3\beta L + \gamma$

$.019\ 787\ 00 = 36\alpha L^2 + 6\beta L + \gamma$

$\therefore -.000\ 058\ 92 = 9\alpha L^2 + 3\beta L$

$-.000\ 117\ 92 = 36\alpha L^2 + 6\beta L$

$\therefore -.000\ 117\ 76 = 6\beta L$

$\beta L = -.000\ 019\ 627$

$36\alpha L^2 = -.000\ 117\ 92 + .000\ 117\ 76$

$36\alpha L^2 = -.000\ 000\ 16$

$\therefore \alpha L^2 = -.000\ 000\ 004\ 444$

FORM 1319A

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No 7/0510/9
SHEET No 9 17

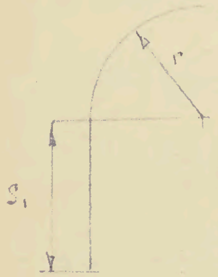
AIRCRAFT: C105

CENTRE FUSELAGE

PREPARED BY	DATE
C.B.	May 17·55
CHECKED BY	DATE
O Lizardenas	June 55

DUCT - TORSION

$$\frac{\partial \sigma_x}{\partial x} + \frac{\partial \tau}{\partial s} = 0 .$$



$$\frac{\partial \tau}{\partial x} + \frac{\partial \sigma_y}{\partial s} = 0 , \qquad \frac{\partial \tau}{\partial x} = 0$$

$$\sigma_y = r p = 0 .$$

$$\tau = f(s)$$

$$\frac{\partial \sigma_x}{\partial x} = -f'(s) ; \qquad \sigma_x = -f'(s)\, x + g(s)$$

BOUNDARY CONDITIONS: $x = 0 \quad \sigma_x = 0 \qquad \therefore g(s) = 0$
$x = L \quad \sigma_x = 0 \qquad \therefore f'(s) = 0$

τ = constant.

DUCT ENERGY - STATION 253 IN

$$2V = \frac{1}{G}\iint \tau_{dk}^2 \, dA \, dx = \frac{t\tau_{dk}^2}{G}\int_0^{\Delta S}\int_0^{L} ds \, dx = \frac{t\tau_{dk}^2 4S \cdot L}{G}$$

$$= \frac{.032 \times 28\cdot 106\,55 \times 22{,}000 \times 4\, \tau'^2_{dk}}{4{,}000}$$

$$= \underline{.019\,787\,00\, \tau'^2_{dk}}$$

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C105	CENTRE FUSELAGE	REPORT NO. 7/0510/9	
		SHEET NO. 9-18	
		PREPARED BY CB	DATE May 24 '55
		CHECKED BY J. Gardener	DATE June 55

DUCT TORSION

STATION MIDWAY BETWEEN STN 255 & START OF CIRCULAR DUCT

'S₁' & 'r' VARYING LINEARLY.

$$S_1 = \frac{8.880\,00 + 0}{2} = 4.440\,00^{IN}$$

$$r = \frac{12.240\,00 + 18.000\,00}{2} = 15.120\,00^{IN}$$

$$S = 4.440\,00 + \frac{3.141\,593 \times 15.120\,00}{2} = 28.190\,443^{IN}$$

$$2V = \frac{.032 \times 4 \times 28.190\,443 \times 22.000}{4\,000.}\,\tau^2 = \underline{.019\,846\,00\,\tau^2}$$

STATION 320^IN

$$S_1 = 0.0^{IN};\ r = 18.000^{IN};\ S = \frac{3.141\,593 \times 18.000}{2} = 28.274\,337^{IN}$$

$$2V = \frac{.032 \times 4 \times 28.274\,337 \times 22.000}{4\,000.}\,\tau^2 = \underline{.019\,905\,13\,\tau^2}$$

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0510/9
SHEET No. 9-19

AIRCRAFT:		PREPARED BY	DATE
C105	CENTRE FUSELAGE.	C.B.	May 19 '55
		CHECKED BY	DATE
			June 55.

DUCT - TORSION. - STATION MIDWAY BTWN STN 255 & START OF CIRCULAR DUCT

"S_1" & "r" VARYING LINEARLY.

$$S_1 = \frac{8,880\ 00 + 0.00}{2} = \underline{4.440\ 00}^{IN}$$

$$r = \frac{12.240\ 00 + 18.000\ 00}{2} = \underline{15.120\ 00}^{IN}.$$

$$S = 4.440\ 00 + \frac{3.141\ 593 \times 15.120\ 00}{2} = 28.190\ 443$$

$$2V = \frac{.032 \times 28.190\ 443 \times 4 \times 22,000\ \tau^2}{4,000} = \underline{.019\ 846\ 00\ \tau^2}$$

RELATIVE τ

FOR CONSTANT TORQUE $\tau_{255} \times A_{255} = \tau_M A_M = \tau_{320} A_{320}$

$$A_{255} = 4 S_1 r + \pi r^2 = 4 \times 8.880 \times 12.240 + 3.141\ 593 \times 12.240^2$$
$$= 434.764\ 800 + 470.665\ 923 = \underline{905.430\ 723}\ IN^2$$

$$A_{MID} = 4 \times 4.440 \times 15.120 + 3.141\ 593 \times 15.120^2$$
$$= 268.531\ 200 + 718.213\ 399 = \underline{986.744\ 599}\ IN^2$$

$$A_{320} = 3.141\ 593 \times 18.000^2 = \underline{1017.876\ 132}\ IN^2$$

	AF	2AF	SQUARE
× .032 =	28.973 783	57.947 566 60.039 114	839.480 101
	31.575 827	63.151 654 64.836 783	997.032 851
	32.572 036	65.144 072	1060.937 529

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT No 7/0510/9

SHEET No 9-19-1

AIRCRAFT: C105	CENTRE FUSELAGE	PREPARED BY C.B.	DATE JUNE 10. 55
		CHECKED BY	DATE

DUCT TORSION   MIDWAY STATION

"S₁" & r VARYING LINEARLY.

$S_1 = \frac{8.880 + 0}{2} = 4.440^{\text{IN}}$; $r = \frac{12.240 + 18.000}{2} = 15.120^{\text{IN}}$

$S = 4.440 + \frac{3.141\,593 \times 15.120}{2} = 28.190\,443$

$2V = \frac{.032 \times 28.190\,443 \times 4 \times 22.000}{4}\tau^2 = .019\,846\,00\,\tau^2$

See p 19

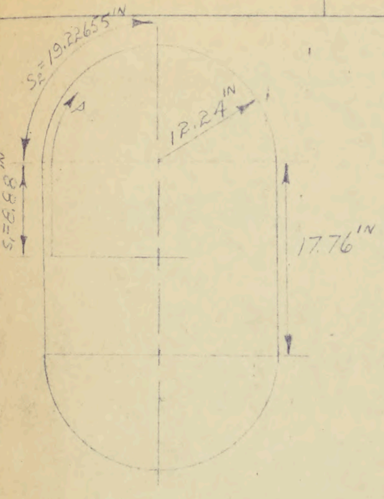
VARIOUS STATIONS   SEE P. 9-16.

STN	$2V/\tau^2$	INTERMEDIATE
255	.019 787 00	
		.019 807
MW	.019 846 00	
		.019 885
320	.019 904 92	

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT		REPORT No 7/0510/9	
		SHEET No 9-20	
C 105	centre Fuselage	PREPARED BY	DATE
		W. GROVER	May 17, 1955
		CHECKED BY	DATE
		O Hyardon	June '55



WARPING IN DUCT

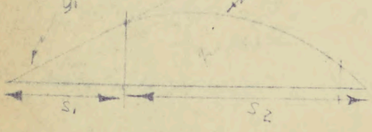
St² 255 Ref. S Young

$S_1 = 8.88000$

$S_2 = \frac{12.24\pi}{2} = 19.22655$

$S = S_1 + S_2 = 28.10655$

Pattern of Warping Stresses



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

at $\Delta = S_1$ $\quad y_1 = y_2 \quad y_1' = y_2'$

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

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

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

$\therefore \; 1 = -\frac{S_1\cos\frac{K\pi}{S}S_2}{\sin\frac{K\pi}{S}S_2}\,\frac{K\pi}{S}$

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

Thus K may be uniquely determined graphically see sheet

$K = 108020.$

Now $\quad \sigma = \frac{\sigma_W}{y_{max}}\frac{x}{L}\left[y_1,\; y_2\right]$ (i.e. y_1 or y_2)

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

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT C 105	Centre Fuselage	REPORT No 7/0510/9 SHEET No 9-21	
		PREPARED BY	DATE
		W. G. POVER	May 13 1955
		CHECKED BY	DATE
		D F Gardner	June 55

and

$$\frac{\partial \sigma}{\partial x} + \frac{\partial \tau}{\partial s} = 0 \qquad \frac{\partial \tau}{\partial x} = c$$

Therefore $\tau = -\frac{\sigma_w}{y_{max} L}\left[\frac{s^2}{2} + C_1 s + \frac{S_1 S}{K\pi}\left(\frac{\cos\frac{K\pi}{S}(S-s)}{\sin\frac{K\pi}{S}S_2} + C_2\right)\right]$

at $s = S_1$

$$\frac{S_1 S}{K\pi}\left(\cot\frac{K\pi}{S}S_2 + C_2\right) = 0$$

$$\therefore \tau = -\frac{\sigma_w}{y_{max} L}\left[C_1 + \frac{s^2}{2}, + \frac{S_1 S}{K\pi}\left\{\frac{\cos\frac{K\pi}{S}(S-s)}{\sin\frac{K\pi}{S}S_2} + \frac{S}{K\pi S_1}\right\}\right]$$

Up to S_1 this torque may be evaluated by simple integration. $-\frac{\sigma_w}{y_{max} L}$ has been neglected throughout & will be reinserted later

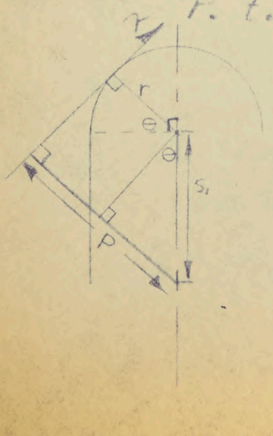
∴ first find

$$rt.\left[\frac{S_1^3}{6} + C_1 S_1\right] = 45.71082 + 3.47812 C_1$$

where $t = .032$ (skin thickness)

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

$\tau .\ t .$ 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 $-\frac{\sigma_w}{y_{max} L}$

$$\tau = \left[\frac{S_1^2}{2} + C_1 + \frac{S_1 S}{K\pi}\left(\frac{\cos\frac{K\pi}{S}(S-s)}{\sin\frac{K\pi}{S}S_2} + \frac{S}{K\pi S_1}\right)\right]$$

Duct Warping:

$$\therefore \text{Torque} = \int \left[\frac{s_1^2}{2} + C_1 + \frac{s_1 s}{k\pi} \left(\frac{\cos \frac{k\pi}{s}(s-\Delta)}{\sin . \frac{k\pi}{s} s_2} + \frac{s}{k\pi s_1} \right) \right] t\left(r + s_1 \sin\left(\frac{\Delta - s_1}{r}\right)\right) d\Delta$$

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

Torque = 1620.33368 + 11.00880 C_1

Thus the total torque is:

1675.04450 + 14.46692 C_1

Equating this to zero we find

$$C_1 = \overline{1}15.62461$$

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

Shear Stresses (neglecting $- \sigma_w / y_{max} t$) $0 \leq \Delta \leq s_1$

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

a) $\Delta = 0$ $\tau = -115.62461$

b) $\Delta = \frac{s_1}{3}$ $\tau = -111.24381$

c) $\Delta = \frac{2s_1}{3}$ $\tau = -98.10141$

d) $\Delta = s_1$ $\tau = -76.19741$

τ for circular portion shown on page 9-24

Computing $\frac{s_1^2}{2} + \frac{s_1 S}{K\pi}\left(\frac{\cos\frac{K\pi}{S}(S-\omega)}{\sin\frac{K\pi}{S}s_2} + \frac{S}{K\pi s_1}\right)$ at 6 positio

where $s_1 = 8.88000$ $K = 1.05020$ $s_2 = 19.226547$ $S = 28.1065$

	1	2	3	4	5	6	7	8	9	10	
n	$s_1 + \frac{n}{6}s_2$	$\frac{s_1^2}{2}$	$\frac{K\pi}{S}$	$\frac{1}{③}$	④s_1	$\frac{④}{s_1}$	(S − ①)	⑦ × ③	③ × s_2	cos ⑧	si
1	12.08442	39.42720	.1207387	8.28235	73.54728	.93270	16.02213	1.93449	2.32139	−.355732	.73
2	15.28885						12.81770	1.54759		.023204	
3	18.49327						9.61327	1.16069		.398707	
4	21.69770						6.40885	.77380		.71526	
5	24.90212						3.20442	.38690		.92612	
6	28.10655						0	0		1	
0	8.88000						19.22655	2.32141		−.682075	

FORM 1344

$$\frac{s_1^2}{2} + \frac{s_1 S}{K\pi}\left(\frac{\cos\frac{K\pi}{S}(S-\omega)}{\sin\frac{K\pi}{S}\, s_2} - \frac{S}{K\pi s_1}\right)$$

6 positions $\oint\left(s_1 \sin\left(\frac{\omega - s_1}{r}\right) + r\right)t$

s_1 = 8.88000 K = 1.05920 s_2 = 19.226547 S = 28.10655 t = ·030 r = 12.24

2	3	4	5	6	7	8	9	10	11	12	13	14	15	16
$\frac{s_1^2}{2}$	$\frac{K\pi}{S}$	$\frac{1}{③}$	④ s_1	$\frac{④}{s_1}$	(S − ①)	⑦ × ③	③ × s_2	cos ⑧	sin ⑨	$\frac{⑩}{⑪}$	⑫ + ⑥	⑬ × ⑤	⑭ + ②	
.42720	·1207337	8.38235	75.54728	·93270	16.02213	1.93440	2.32129	$\overline{\cdot 351732}$	·731262	$\overline{\cdot 48645}$	-·44625	38.88047	72.24767	
					12.81770	1.54759		·023804		·03173	·96443	70.93119	110.35538	
					9.61327	1.16069		·398707		·54522	1.47792	108.69698	148.12418	
					6.40885	·77380		·71526		·97809	1.91079	140.53339	179.96059	
					3.20442	·38690		·92612		1.26643	2.19913	161.74001	201.16721	
					0	0		1		1.36746	2.30016	169.17049	208.59769	
					19.22655	2.32141		$\overline{\cdot 682075}$		$\overline{\cdot 93271}$	·00000	0	39.42720	
													↑ referred	to a

VRO AIRCRAFT LIMITED

MALTON, ONTARIO
ENGINEERING DIVISION

AIRCRAFT - - C 105
WEIGHT - -
C. G. POSITION -

REPORT NO. - 7/0510/3
SHEET - - D - 23
DATE - - May 16
PREPARED BY - W. Grove

16	17	18	19	20	21	22	23	24	25	26	27	28	29
	① − S_1	$\frac{⑰}{r}$	SIN ⑱	⑲ S_1	⑳ + r	㉑ t							
	3.20442	·26180	·25882	2.29832	14.53832	·46523							
	6.40885	·52360	·50000	4.44000	16.68000	·53376							
	9.61327	·78540	·70711	6.27914	18.51914	·59261							
	12.81770	1.04720	·86603	7.69032	19.93032	·63777							
	16.02212	1.30900	·96593	8.57746	20.81746	·66616							
	19.22655	1.57080	1.00000	8.88000	21.12000	·67584							
	0	0	0	0	12.24000	·39168							

as Ψ on following sheet

↑referred to as Ω on following sheet

	1	2	3	4	5	6	7	8	9	10	11
Station	$\Psi + C_1$	Ω	①×②	Δ_1	Δ_2	Δ_3	Δ_4	Δ_5	Δ_6	Weddle Coeff's	⑩×③
S_1	39.42720	.39168	15.44245							1	15.44245
	$+C_1$		+.39168C_1	18.16933							+.39168C_1
$S_1 + \frac{1}{6} S_2$	72.24767	.46523	33.61178	+.07355C_1	7.12378					5	168.05890
	$+C_1$		+.46523C_1	25.29311	−.00502C_1	−3.54191					+2.32615C_1
$S_1 + \frac{1}{3} S_2$	110.35832	.53376	58.90489	+.06853C_1	3.58187	−.00466C_1	−1.92134			1	58.90489
	$+C_1$		+.53376C_1	28.87498	−.00968C_1	−5.46325	+.00065C_1	+1.50845			+.53376C_1
$S_1 + \frac{1}{2} S_2$	148.12418	.59261	87.77987	+.05885C_1	−1.88138	−.00401C_1	−.41299	+.00128	+.23113	6	526.67922
	$+C_1$		+.59261C_1	26.93369	−.01369C_1	−5.87614	+.00093C_1	+1.71959	+.00007C_1		3.55566C_1
$S_1 + \frac{2}{3} S_2$	179.96059	.63777	114.77347	+.04516C_1	−7.75752	−.00308C_1	+1.36669	+.00021C_1		1	114.77347
	$+C_1$		+.63777C_1	19.35608	−.01677C_1	−4.50945	+.00114C_1				+.63777C_1
$S_1 + \frac{5}{6} S_2$	201.16721	.66616	134.00955	+.02839C_1	−12.76697	−.00194C_1				5	670.04775
	$+C_1$		+.66616C_1	6.96911	−.01871C_1						+3.33080C_1
S	206.59769	.67594	140.97866	+.00968C_1						1	140.97866
	$+C_1$		+.67594C_1								+.67594C_1
										Σ 16	94.85584+11.

FORM 1344

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	Weddle Coeff's	⑩ × ③		Six Strip Coeff's	⑬ × ③			
						1	15.44245		41	633.14045			
18.16933							+ .39168C_1			+ 16.05888 C_1			
+ .01355C_1	7.12378					5	168.05890		216	7260.14478			
25.29311	− .00502C_1	− 3.54191					+ 2.32615C_1			+ 100.48068C_1			
+ .06853C_1	3.58187	− .00466C_1	− 1.92134			1	58.90489		27	1590.47203			
28.67498	− .00968C_1	− 5.46325	+ .00065C_1	+ 1.50845			+ .53376C_1			+ 14.41152 C_1			
+ .05885C_1	− 1.88138	− .00401C_1	− .41289	+ .00128	+ .27113	6	526.67922		272	23876.12464			
26.99360	− .01369C_1	− 5.87614	+ .00093C_1	+ 1.71959	+ .00007C_1		3.55566C_1			+ 161.16992C_1			
+ .04516C_1	− 7.75755	− .00308C_1	+ 1.36669	+ .00021C_1		1	114.77347		27	3098.88369			
19.35608	− .01677C_1	− 4.50945	+ .00114C_1				+ .63777C_1			17.21979			
+ .02839C_1	− 12.76691	− .00194C_1				5	670.04775		216	28946.06280			
6.96911	− .01871C_1						+ 3.33080C_1			+ 143.89056C_1			
+ .00966C_1						1	140.97866		41	5780.12506			
							+ .67584C_1			+ 27.70944C_1			
						Σ 16	94.88534 + 11.45166C_1		Σ	71164.91315 + 480.96979C_1			

AVRO AIRCRAFT LIMITED
MALTON, ONTARIO
ENGINEERING DIVISION

AIRCRAFT - - C 105
WEIGHT - -
C. G. POSITION -

REPORT NO. - 7/0510/9
SHEET - - 9,-4
DATE - - May 17, 1955
PREPARED BY - W. Prover

17	18	19	20	21	22	23	24	25	26	27	28	29

① with C_1

$\bar{7}4.19741$

Weddle

$$\frac{19.22655}{20}\left[1634.88534 + 11.45166\,C_1\right] - \frac{19.22655}{6\times140}\left[.27113 + .00087\,C_1\right]$$

$\bar{4}3.37694$

$$= 1629.33368 + 11.00880\,C_1$$

$\bar{5}.26672$

Six Strip

32.49957

$$\frac{19.22655}{840}\left[71184.91315 + 480.96979\,C_1\right]$$

64.33598

$$= 1629.33368 + 11.00880\,C_1$$

35.54260

92.97308

5

69 79 C_1

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C 105	Center Fuselage	REPORT No 7/0510/9 SHEET No 9-25	
		PREPARED BY W. Grever	DATE May 17/55
		CHECKED BY Abarden.s	DATE June 55

2) Calculation of σ's (neglecting $+ {}^{cw}/_{L}$)

a) $\Delta = 0$ $\qquad \sigma = 0$

b) $\Delta = S_1$ $\qquad \sigma = S_1 = 8.88000$

c) $\Delta = S_1 + \frac{1}{6} S_2$

$$\sigma = S_1 \frac{\sin \frac{K\pi}{S}(S-\Delta)}{\sin \frac{K\pi}{S} S_2}$$

$$= \frac{8.88000}{.731262} \sin .1207387(S-\Delta)$$

$\sigma = 12.14306 \sin 1.93449$

$= 11.34878$

d) $\Delta = S_1 + \frac{1}{3} S_2$

$\sigma = 12.14306 \sin 1.54759$

$\sigma = 12.13978$

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

$\sigma = 12.14306 \sin 1.16069$

$\sigma = 11.13616$

f) $\Delta = S_1 + \frac{2}{3} S_2$

$\sigma = 12.14306 \sin .77390$

$\sigma = 8.48630$

g) $\Delta = S_1 + \frac{5}{6} S_2$

$\sigma = 12.14306 \text{ SIN } .38690$

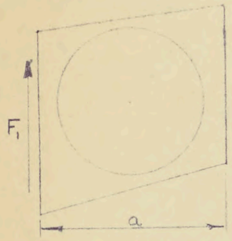
$\sigma = 4.58182$

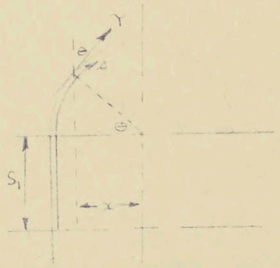
h) $\Delta = S$ $\qquad \sigma = 0$

for graphs see pages

AIRCRAFT: C 105	Centre Fuselage	PREPARED BY W. Grover	DATE June 7, 1955
		CHECKED BY J Hastings	DATE June '55.

Duct WARPING





Vertical component of force $= \tau t\, ds \cos\theta$

let Shear Strain of force be γ

$\therefore$ work done by vertical component of force $= \frac{\tau t\, ds}{2} \cos\theta \times \gamma$

$$= \frac{r t \tau}{2} \gamma \cos^2\theta\, ds \qquad (x = r\cos\theta)$$

$$= \frac{r t \tau}{2} \gamma \cos^2\left(\frac{S-s}{r}\right) ds$$

$$\text{Total work done} = \frac{r t \gamma}{2} \int \tau \cos^2\left(\frac{S-s}{r}\right) ds$$

$$= \frac{F_1 a \gamma}{2}$$

$$\therefore F_1\, a = t r \int \tau \cos^2\left(\frac{S-s}{r}\right) ds$$

Neglecting $^{\sigma w}/_{y_{max}} L$ τ & $\cos\left(\frac{S-s}{r}\right)$ have been evaluated at six intervals around the circular portion of the duct. Thus the above integral may be approximated using Weddle's Formula

Over the straight portion of the duct direct integration may again be used.

$$r t \int_0^{S_1} \left(\frac{s^2}{2} + C_1\right) ds = r t \left(\frac{S_1^3}{6} + S_1 C_1\right)$$

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO 1/0510/9
SHEET NO 9 - 27

PREPARED BY	DATE
W. Grever	June 15/55
CHECKED BY	DATE
	June 22.55

AIRCRAFT: C 105

Centre Fuselage

Duct Warping:

$$tr\left(\frac{s_1^3}{6} + C_1 s_1\right) = 45.71082 + 3.47812 C_1$$

$$= 45.71082 - 402.15627$$

$$= -356.44545$$

Station	Cos θ	Cos² θ	rt Cos² θ	γ	γrt cos² θ	Weddle Coeft.	Product
s_1	1.00000	1.000000	·39168	$\overline{76}.19741$	$\overline{29}.84500$	1	$\overline{29}.84500$
$s_1 + \frac{1}{6}s_2$	·96593	·93302	·36545	$\overline{43}.37694$	$\overline{15}.85216$	5	$\overline{79}.26050$
$s_1 + \frac{1}{3}s_2$	·86602	·74999	·29376	$\overline{5}.26622$	$\overline{1}.54700$	1	-1.54700
$s_1 + \frac{1}{2}s_2$	·70711	·50000	·19584	32.49957	6.36472	6	38.18832
$s_1 + \frac{2}{3}s_2$	·50000	·25000	·09792	64.33596	6.29978	1	6.29978
$s_1 + \frac{5}{6}s_2$	·25882	·06699	·02624	85.54260	2.24464	5	11.22320
S	0	0	0	92.97308	0	1	0

$$\Sigma = \overline{54}.94120$$

$$I = \frac{19.22655}{20}\left(\overline{54}.94120\right)$$

$$= -52.81649$$

$$\text{Total Integral} = -356.44546 - 52.81649$$

$$= -409.26194$$

$$\therefore \text{Frame force } F_1 = \frac{-409.26194}{45} = -9.09471\left(\frac{-\sigma_w}{Y_{max} L}\right)$$

over 1/4 of the section

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0510/9

SHEET No. 9 - 28

PREPARED BY	DATE
W. Grever	June 15/56
CHECKED BY	**DATE**
P	June 18 56

AIRCRAFT: C 105

Centre Fuselage

Duct Warping:

Strain Energy due to Shear:

The shear stress at any point is $\left(\frac{\sigma_{w_1}}{y_{max}L} - \frac{\sigma_{w_2}}{y_{max}L}\right)K_{\tau_s}$ where K_{τ_s} is the coefficient previously determined.

The strain energy $V = \iint \frac{\tau^2}{G}\, dA\, dx$

$$\therefore\ 2V = \frac{(\sigma_{w_1} - \sigma_{w_2})^2}{L G\, y_{max}^2} \int K_{\tau_s}^2\, ds$$

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

Station	K_{τ_s}	$K_{\tau_s}^2$	"3/8" Coeff's	Product
0	−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	5806.04529	1	5806.04529

$\Sigma = 85172.31143$

$I = 1.11\,(85172.31143) = 94541.26569$

Station	K_{τ_s}	$K_{\tau_s}^2$	Weddle Coeff's	Product
S_1	$\bar{7}6.19741$	5806.04529	1	5806.04529
$S_1 + 1/6\ S_2$	$\bar{4}3.37694$	1881.55892	5	9407.79460
$S_1 + 1/3\ S_2$	$\bar{5}.26622$	27.73307	1	27.73307
$S_1 + 1/2\ S_2$	32.49957	1056.22205	6	6337.33230
$S_1 + 2/3\ S_2$	64.33598	4139.11832	1	4139.11832
$S_1 + 5/6\ S_2$	85.54260	7317.53641	5	36587.68205
S	92.97306	8643.99360	1	8643.99360

$\Sigma = 70949.69923$

$I = .9613275\,(70949.69923) = 68205.89689$

$\therefore$ Total $I = 162,747.16268$

$$\therefore\ 2V = \frac{.032 \times 162747.16268}{22 \times 4,000 \times (12.14306)^2}\left[\sigma_{w_1} - \sigma_{w_2}\right]^2 = .000401351\left[\sigma_{w_1} - \sigma_{w_2}\right]^2$$

FORM 1319A

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

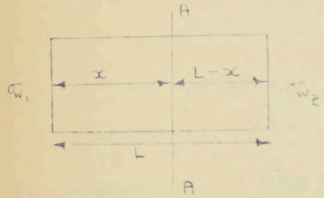
REPORT No 7/0510/9

SHEET No 9-29

AIRCRAFT		PREPARED BY	DATE
C105	Centre Fuselage	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

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

but $\sigma = \frac{1}{y_{max}}\left[\sigma_{w_1}\frac{x}{L} + \sigma_{w_2}\frac{L-x}{L}\right] K_{\sigma_s}$

$$\therefore\ 2V = \frac{1}{E y_{max}^2}\iint\left(\sigma_{w_1}\frac{x}{L} + \sigma_{w_2}\left(1-\frac{x}{L}\right)\right)^2 K_{\sigma_s}^2\, t\, ds\, dx$$

$$= \frac{tL}{3E\, y_{max}^2}\int\left(\sigma_{w_1}^2 + \sigma_{w_1}\sigma_{w_2} + \sigma_{w_2}^2\right) K_{\sigma_s}^2\, ds$$

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

Station	K_{σ_s}	$K^2_{\sigma_s}$	⅜ coeff's	Product
0	0	0	1	0
$S_1/3$	2.96000	8.76160	3	26.28480
$2S_1/3$	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_{σ_s}	$K^2_{\sigma_s}$	Weddle Coeff.	Product
S_1	8.88000	78.85440	1	78.85440
$S_1 + S_2/6$	11.34878	128.79481	5	643.97405
$S_1 + S_2/3$	12.13978	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

$\Sigma = 1791.26971$

$I = .9613275(1791.26971) = 1721.99683$

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO	7/0510/9		
SHEET NO	9-30		
AIRCRAFT: C 105	Centre Fuselage	PREPARED BY: W. Grever	DATE: June 16/55
		CHECKED BY:	DATE: JULY 55

Duct Warping:

Total $I = 1955.40585$

$$\therefore\quad 2V = \frac{\cdot 032 \times 22\,000 \times 1955.40585}{3 \times 16{,}600 \times (12.14306)^2}\left[\sigma_{W_1}^2 + \sigma_{W_1}\sigma_{W_2} + \sigma_{W_2}^2\right]$$

$$= \cdot 00029358\left[\sigma_{W_1}^2 + \sigma_{W_1}\sigma_{W_2} + \sigma_{W_2}^2\right]$$

over 1/4 of the duct

Summary: Station 255

Force on Frame:

$$F_1 = \cdot 136175\ \sigma_{W_1}$$

Strain Energies:

a) Axial

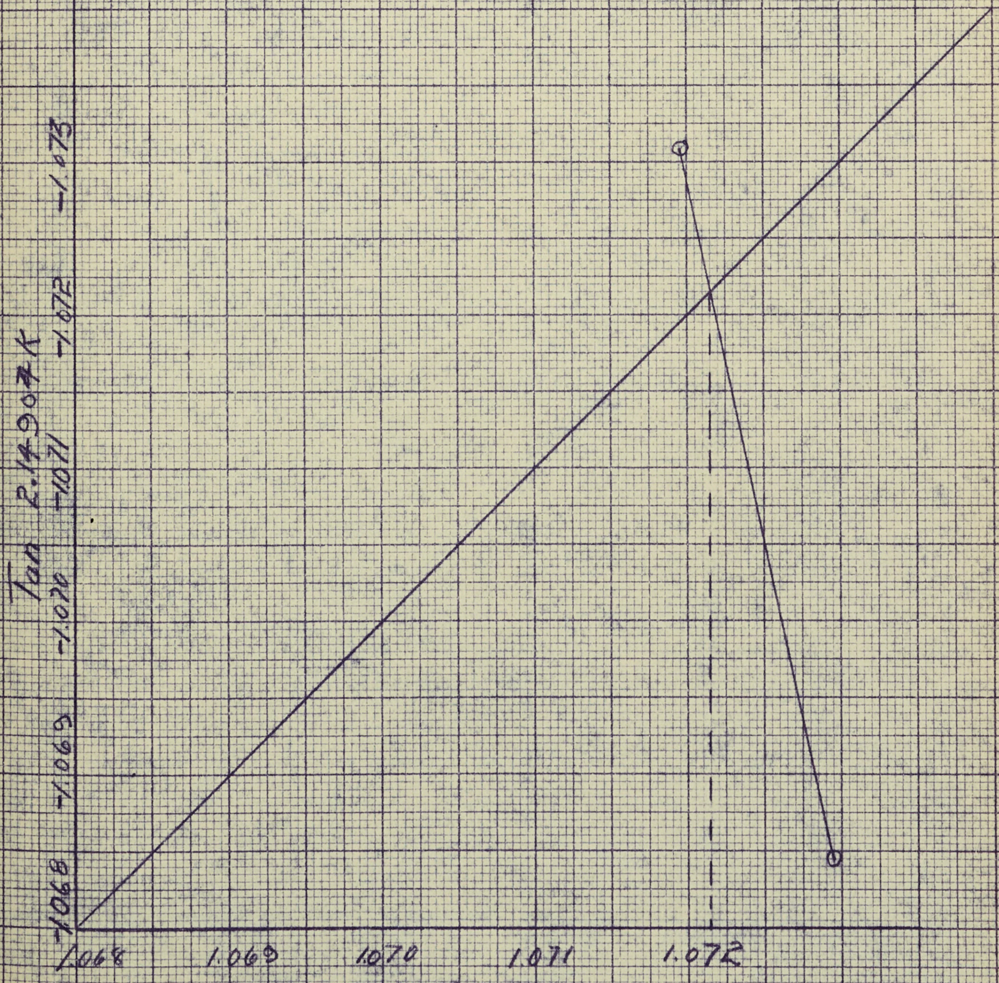
$$2V_\sigma = \cdot 00117432\left[\sigma_{W_1}^2 + \sigma_{W_2}^2\right] + \cdot 00055716\left[2\sigma_{W_1}\sigma_{W_2}\right]$$

b) Shear

$$2V_\tau = \cdot 00160540\left[\sigma_{W_1}^2 + \sigma_{W_2}^2\right] - \cdot 00160540\left[2\sigma_{W_1}\sigma_{W_2}\right]$$

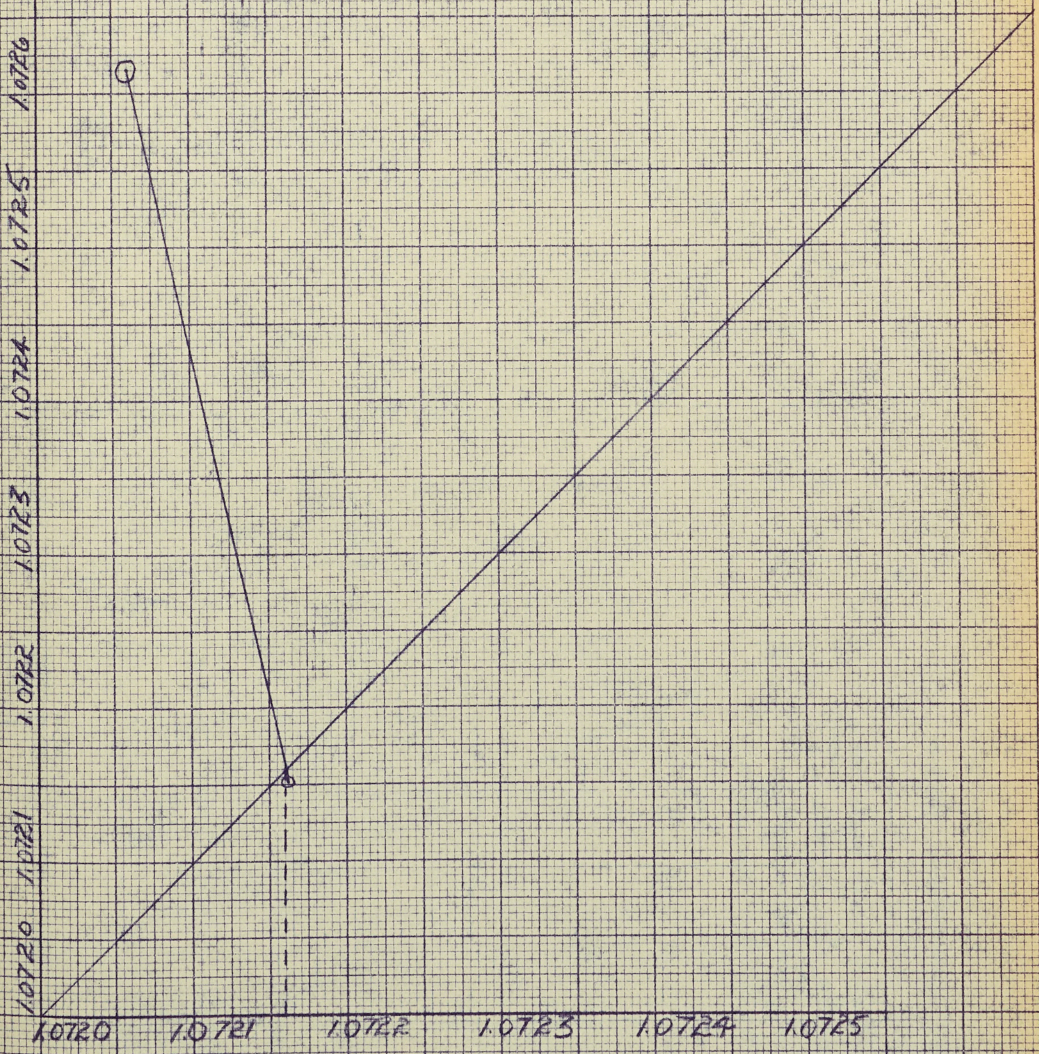
FORM 1319A

DUCT : STN. 255.00"
DETERMINATION OF "K"



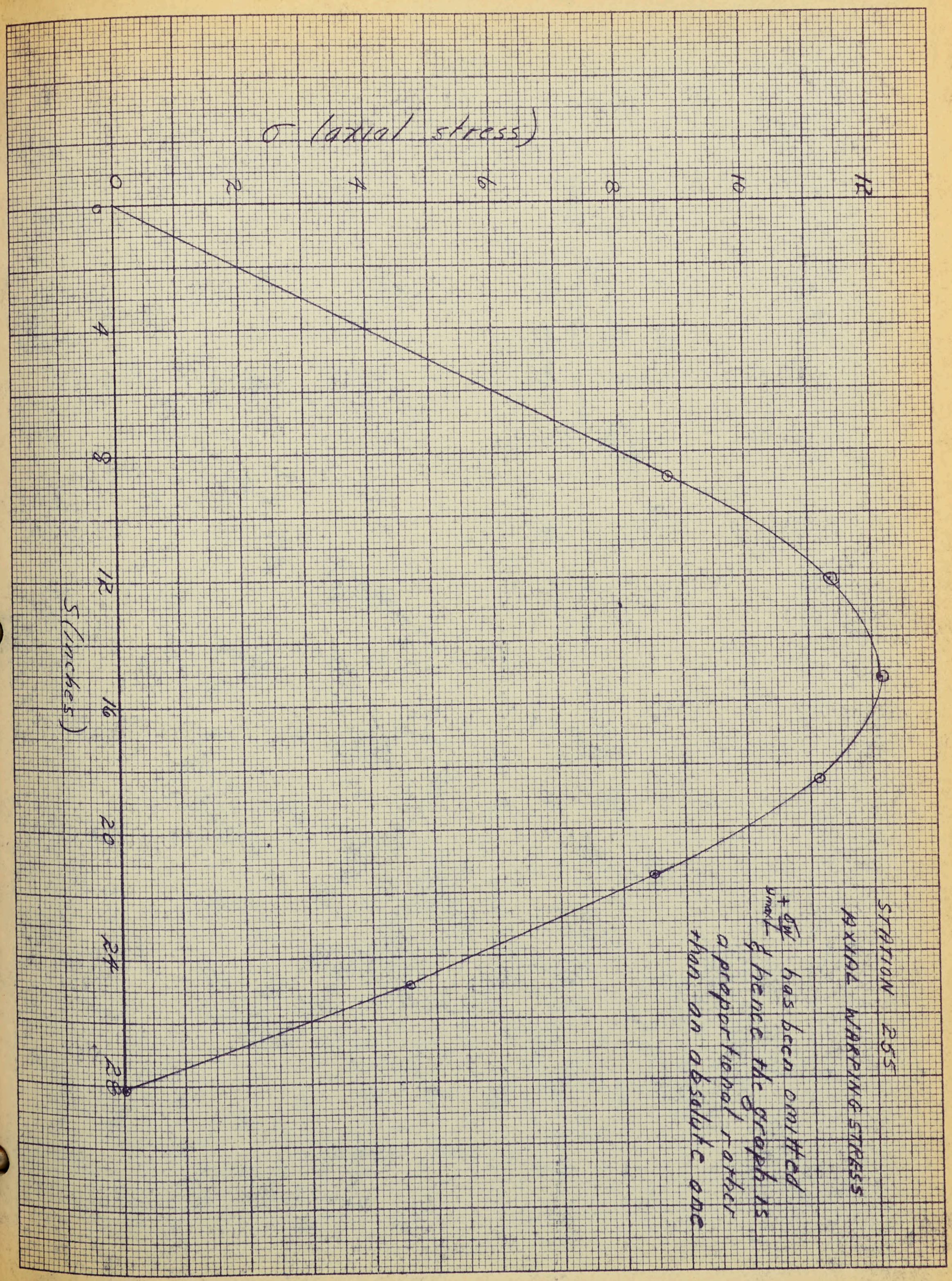
.99256 K = 1.07215
K = 1.08019

DIST - STN 255.00"
DETERMINATION OF "K"



.992557 K = 1.07216
∴ K = 1.08020

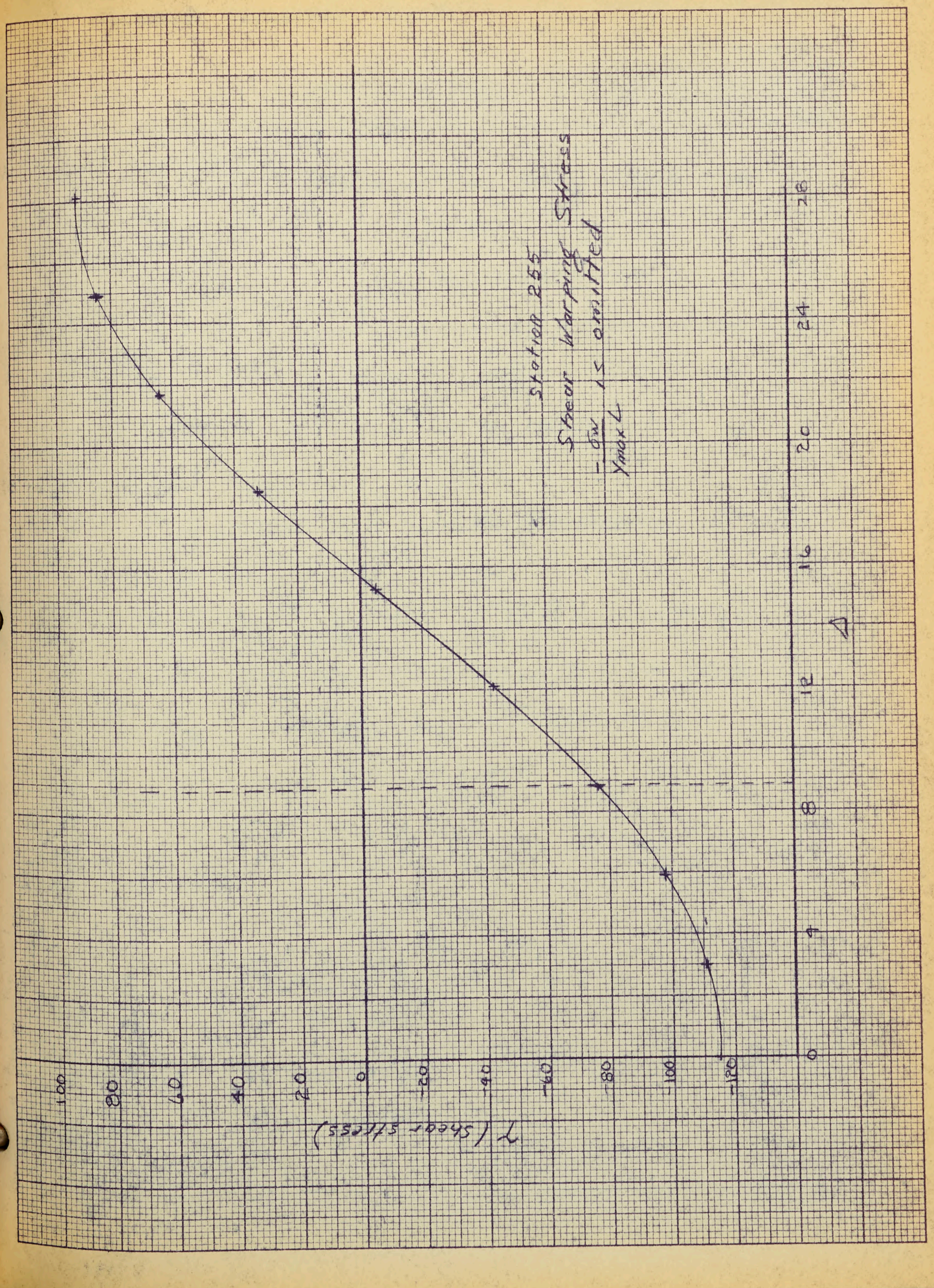
7/05109



STATION 255

AXIAL WARPING STRESS

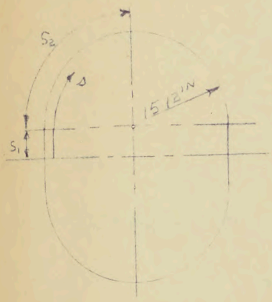
$+\frac{dW}{u_{max}L}$ has been omitted & hence the graph is a proportional rather than an absolute one


Station 255
Shear Warping Stress
$-\frac{\sigma w}{\gamma_{max} L}$ is omitted
τ (Shear stress)
100
80
60
40
20
0
-20
-40
-60
-80
-100
-120
Δ
0
4
8
12
16
20
24
28

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C 105	Centre. Fuselage	REPORT No. 7/0510	
		SHEET No. 9-35	
		PREPARED BY	DATE
		W. Grover	June 16/55
		CHECKED BY	DATE
		B	June 22.55

Duct Warping:



Station midway between 255 & 320

$S_1 = 4.44$

$S_2 = \frac{15.12}{2}\pi = 23.75044$

See sheets 9-20 To 22 for basic information

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

$K = 1.01161$ ← see sheet

Torque: a) for portion $0 \leq s \leq S_1$

$rt\left[S_1^3/6 + C_1 S_1\right] = 7.058288 + 2.148250$

b) for portion $S_1 \leq s \leq S$

$\int\left[\frac{S_1^2}{2} + C_1 + \frac{SS_1}{K\pi}\left(\frac{\cos\frac{K\pi}{S}(S-s)}{\sin\frac{K\pi}{S}S_2} + \frac{S}{K\pi S_1}\right)\right]t\left[r + S_1\sin\left(\frac{s-S_1}{r}\right)\right]ds$

$= 1465.39876 + 13.63969\,C_1$ ← see sheets

adding $1472.45705 + 15.78794\,C_1 = 0$

$\therefore\ C_1 = -93.26467$

(again $\frac{\sigma_x}{y_{max}L}$ has been omitted)

—"—

Determination of γ $0 \leq s \leq S_1$

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

a) $s = 0$ $\gamma = -93.26467$

b) $s = S_1/3$ $\gamma = -92.16947$

c) $s = 2S_1/3$ $\gamma = -88.88387$

d) $s = S_1$ $\gamma = -83.40787$

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No 7/0510/9		SHEET No 9.36	
AIRCRAFT C 105	Centre Fuselage	PREPARED BY w. Grover	DATE June 16 /55
		CHECKED BY	DATE July '55

Duct Warping: Minwel

Calculation of σ's (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) $\Delta = S_1 + 1/6\, S_2 = 8.39841$ $\sigma = S_1 \dfrac{\sin\frac{k\pi}{s}(S-\Delta)}{\sin\frac{k\pi}{s} S_2} = 9.919571 \sin 2.231274$

$\sigma = 7.83348$

d) $\Delta = 12.35681$ $\sigma = 9.919571 \sin 1.785004$
$= 9.69286$

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:
$rt(S_1^3/6 + C_1 S_1) = 7.058288 + 2.14825\, C_1 = \overline{193.29754}$

Station	$\cos\theta$	$\cos^2\theta$	$rt\cos^2\theta$	T	$Trt\cos^2\theta$	Weighting Coeff	Product
S_1	1.000000	1.00000	.48384	$\overline{83.40187}$	$\overline{40.35606}$	1	$\overline{40.35606}$
$S_1 + 1/6\, S_2$	.96593	.93302	.45143	$\overline{58.70480}$	$\overline{26.50111}$	5	$\overline{132.50555}$
$S_1 + 1/3\, S_2$	.86603	.75000	.36288	$\overline{23.42899}$	$\overline{8.50191}$	1	$\overline{8.50191}$
$S_1 + 1/2\, S_2$	.70711	.50000	.24192	15.51036	3.75227	6	22.51362
$S_1 + 2/3\, S_2$	.50000	.25000	.12096	50.48645	6.10684	1	6.10684
$S_1 + 5/6\, S_2$	.25882	.06699	.03241	74.64878	2.41937	5	12.09685
S	0	0	0	83.26569	0	1	0

$\Sigma = \overline{140.64621}$

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

K = 1.01161
S_1 = 4.44
S_2 = 23.75044
S = 28.19044
r = 15.12
t = .032

STATION MIDWAY BETWEEN 255 & 320

Calculation of

$$\frac{S_1^2}{2} + \frac{SS_1}{K\pi}\left(\frac{\cos\frac{K\pi}{S}(S-\Delta)}{\sin\frac{K\pi}{S}S_2} + \frac{S}{K\pi S_1}\right)$$

f' (S

	1	2	3	4	5	6	7	8	9	10	11
n	Δ $S_1 + \frac{n}{6} S_2$	$\frac{S_1^2}{2}$	$\frac{K\pi}{S}$	$\frac{1}{③}$	④ S_1	④/S_1	S − ①	⑦ × ③	③ × S_2	cos ⑧	sin ⑨
0	4.44	9.85680	.112735	8.870359	39.38439	1.99783	23.75044	2.677505	2.677505	$\bar{.}$894241	.4476
1	8.39841						19.79203	2.231255		$\bar{.}$613479	
2	12.35681						15.83363	1.785004		$\bar{.}$212573	
3	16.31522						11.87522	1.338753		.229966	
4	20.27363						7.91681	.892502		.627466	
5	24.23203						3.95841	.446251		.902072	
6	28.19044	↓	↓	↓	↓	↓	0	0	↓	1.00000	↓

FORM 1544

STATION MIDWAY BETWEEN 255 & 320

Calculation of:

$$\frac{S_1^2}{2} + \frac{SS_1}{K\pi}\left(\frac{\cos\frac{K\pi}{S}(S-\ldots)}{\sin\frac{K\pi}{S}S_2} + \frac{S}{K\pi S_1}\right) \qquad \xi'\left(S_1\sin\left(\frac{\Delta - S_1}{r}\right) + r\right)t$$

3	4	5	6	7	8	9	10	11	12	13	14	15	16	17
	$\frac{1}{③}$	④ S_1	④/S_1	S − ①	⑦ × ③	③ × S_2	cos ⑧	sin ⑨	$\frac{⑩}{⑪}$	⑫ + ⑥	⑬ × ⑤	⑭ + ②		
735	8.870359	39.38439	1.99783	23.7[illegible]44	2.677505	2.677505	$\overline{.894241}$	.44760	$\overset{3}{\overline{1.99779}}$	0	0	9.85[illegible]		0
				19.79203	2.231255		$\overline{.613479}$		$\overline{1.37060}$	.62725	24.70507	34.55087		3.356
				15.83363	1.785004		$\overline{.212573}$		$\overline{.47492}$	1.52291	53.97893	63.85568		7.9168
				11.87522	1.338753		.229266		.51378	2.51161	98.91823	108.77553		11.875
				7.91681	.892502		.627466		1.40185	3.39968	133.80452	143.75112		15.833
				3.95841	.446251		.902072		2.01535	4.01318	158.05665	167.91345		19.79
↓	↓	↓	↓	0	0	↓	1.00000	↓	2.22414	4.23197	166.67356	176.53036		23.750

reffered to as φ
on following sheet

AVRO AIRCRAFT LIMITED

MALTON, ONTARIO
ENGINEERING DIVISION

AIRCRAFT - - C105
WEIGHT - -
C. G. POSITION -

REPORT NO. - [illegible]
SHEET - - 9-37
DATE - - May 26/55
PREPARED BY - W. Grover

7	18	19	20	21	22	23	24	25	26	27	28	29
	⑰ / r	sin ⑱	⑲ Si	⑳ + r	㉑ t							
	0	0	0	15.12	·48384							
891	·21180	·258820	1.14916	16.26916	·52061							
81	·52360	·500001	2.22000	17.34000	·55488							
522	·78540	·707108	3.13956	18.25956	·58431							
3365	1.04720	·866027	3.84516	18.96516	·60689							
203	1.30900	·965927	4.28872	19.40872	·62108							
5094	1.57080	1.00000	4.44	19.56000	·62592							

A

referred to 63-52
on following sheet

Centre Fuselage   Duct Warping

$\int(-\Omega)(\psi_1 + C_1)\,ds$ by Weddle & Six Strip Numerical Integratio

	1	2	3	4	5	6	7	8	9	10	11
n	$\psi + C_1$	$-\Omega$	①×②	$-\Delta_1$	$-\Delta_2$	$-\Delta_3$	$-\Delta_4$	$-\Delta_5$	$-\Delta_6$	Weddle Coeff	⑩×③
0	9.85680	·48584	4.76911							1	4.76 911
	$+C_1$		+ ·48384C_1	13.22310							+ ·48 384C_1
1	34.55967	·52061	17.99221	+ ·03677C_1	7.53511					5	8 9.36 105
	$+C_1$		+ ·52061C_1	20.75621	− ·00250	− 3.48540					+ 2.60 305C_1
2	69.83568	·55488	38.75042	+ ·03427C_1	4.04971	+ ·00234C_1	− 1.68945			1	3 8.75 092
	$+C_1$		+ ·55488C_1	24.80792	− ·00484	− 5.17485	− ·00033C_1	− 1.35323			+ ·55 488C_1
3	108.77503	·58431	63.55634	+ ·02943C_1	− 1.12514	+ ·00201C_1	− ·33622	− ·00014C_1	·28989	6	38 1.35 004
	$+C_1$		+ ·58431C_1	22.68278	− ·00665C_1	− 5.51107	− ·00047C_1	+ 1.64312	·00003C_1		+ 3.50 586C_1
4	143.75112	·60669	87.24112	+ ·02258C_1	− 6.63621	+ ·00154C_1	+ 1.30690	− ·00011C_1		1	8 7.24 112
	$+C_1$		+ ·60669C_1	17.04657	− ·02839C_1	− 4.20417	− ·00058C_1				+ ·60 689C_1
5	167.91345	·62108	104.28763	·01419C_1	− 10.84038	+ ·00096C_1				5	52 1.43 845
	$+C_1$		+ ·62108C_1	6.20613	− ·00935C_1						3.10 540C_1
6	176.53036	·62592	110.49388	+ ·00484C_1						1	11 0.49 388
	$+C_1$		+ ·62592C_1								·62 592C_1
									Σ =	234.00407 + 11.465	

FORM 1544

arping

eddle & Six Strip Numerical Integration

6	7	8	9	10	11	12	13	14	15	16	17	18	19
$-\Delta_3$	$-\Delta_4$	$-\Delta_5$	$-\Delta_6$	Weddle Coeff	⑩ × ③			Six Strip Coeffs					
				1	4.76 911			41	12 5.53 351		Weddle		
					+ .48 384C_1				+ 1 9.83 744C_1		$\frac{23.75094}{20}$	$\{1234.00$	$407 + 11.4$
				5	8 9.96 105			216	388 6.31 736			=	1465.30
— 3.48540					+ 2.60 305C_1				+ 11 2.45 176C_1		Six Strip		
+ .00234C_1	— 1.68945			1	3 8.75 042			27	104 6.26 134		$\frac{23.75094}{840}$	$\{5$	1827.881
— 5.17485	— .00033C_1	+ 1.35363			+ .55 488C_1				+ 1 4.98 176C_1				
+ .00201C_1	— .33622	— .00014C_1	.28989	6	38 1.35 004			272	1 728 7.86 848			= 1465.	39878
— 5.51107	— .00047C_1	+ 1.64312	.00003C_1		+ 3.50 586C_1				+ 15 8.93 232C_1				
+ .00154C_1	+ 1.30690	— .00011C_1		1	8 7.24 112			27	235 5.51 024				
— 4.20417	— .00058C_1				+ .60 689C_1				+ 1 6.38 603C_1				
+ .00096C_1				5	52 1.43 845			211	2 252 6.14 104				
					3.10 540C_1				+ 13 4.15 328C_1				
				1	11 0.49 388			41	453 0.24 908				
					.62 592C_1				+ 2 5.66 272C_1				

$\Sigma = 1234.00407 + 11.48584C_1$ $\Sigma = 51827.88105 + 482.40531C_1$

AVRO AIRCRAFT LIMITED
MALTON, ONTARIO
ENGINEERING DIVISION

AIRCRAFT - - C 105
WEIGHT - -
C. G. POSITION -

REPORT NO. - 71/STR/9
SHEET - - 9 - 38
DATE - - May 26/55
PREPARED BY - W. Grover

18	19	20	21	22	23	24	25	26	27	28	29
											① with C_1
											− 53.40787
											− 58.70462
											− 23.42899
											+ 15.51136
											+ 50.48645
											+ 74.64878
											+ 83.2[illegible]

lc

$\text{44}\left[1234.00407 + 11.485849C_1\right] - \frac{23.75044}{840}\left[.28883 + .0006 3C_1\right]$

$= 1465.30878 + 13.63969C_1$

Strip

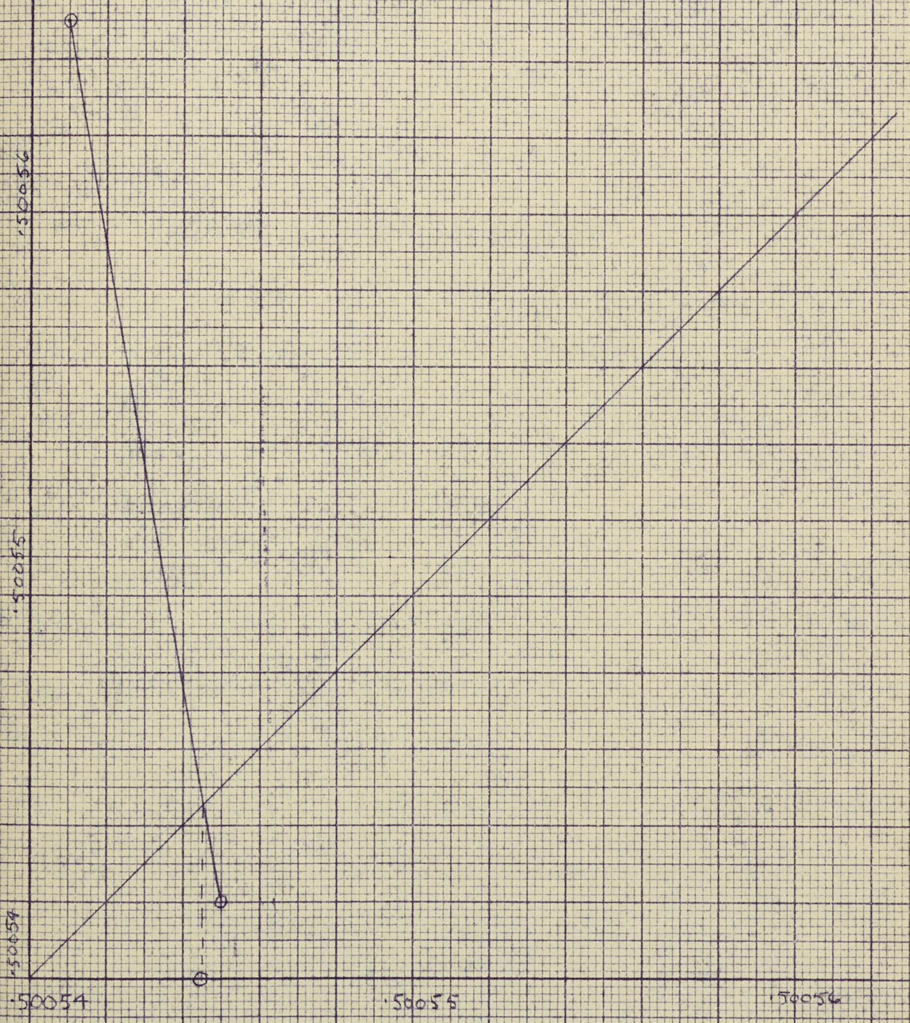
$\frac{\text{5044}}{\text{40}}\left[51527.88105 + 482.40531C_1\right]$

$= 1465.39878 + 13.63969C_1$

$r = -93.26467$

iC_1

DETERMINATION OF K



·494801 K

·494801 K = ·500545

K = 1.011607

REPORT No 7/C510/9	SHEET No 9-40	
AIRCRAFT: C 105	Centre Fuselage	
PREPARED BY	DATE	
W. Grover	May 30/55	
CHECKED BY	DATE	
Vf Gardener	June '55	

$$\therefore F_1 = \frac{\bar{3}6031801}{45} = \bar{8}.007067$$

<u>Strain Energy due to Axial Stresses</u>

Station	$K\sigma_s$	$K^2\sigma_s$	"3/8" Coeff.	Product
0	0	0	1	0
$S_1/3$	1.48000	2.19040	3	6.57120
$2S_1/3$	2.96000	8.76160	3	26.28480
S_1	4.44000	19.71360	1	19.71360

$\Sigma = 52.56960$

$I = .555(52.56960) = 29.17613$

Station	$K\sigma_s$	$K^2\sigma_s$	Weddle Coeff	Product
S_1	4.44	19.71360	1	19.71360
$S_1 + 1/6\,S_2$	7.83348	61.36341	5	306.81705
$S_1 + 1/3\,S_2$	9.69286	93.95153	1	93.95153
$S_1 + 1/2\,S_2$	9.65371	93.19412	6	559.16472
$S_1 + 2/3\,S_2$	7.72381	59.65724	1	59.65724
$S_1 + 5/6\,S_2$	4.28115	18.32825	5	91.64125
S	0	0	1	0

$\Sigma = 1130.94539$

$I = 1.187522(1131.02481) = 1345.02253$

$\therefore$ total $I = 1372.19366$

$$V_c = \frac{.032 \times 22.00000}{2\times3\times10,600\,(9.319571)^2}(1372.19866)\left[\sigma_{W_1}^2 + \sigma_{W_1}\sigma_{W_2} + \sigma_{W_1}^2\right]$$

$$= .000154364\left[\sigma_{W_1}^2 + \sigma_{W_2}^2\right] + .000077182\left[2\sigma_{W_1}\sigma_{W_2}\right]$$

AIRCRAFT: C105	Centre Fuselage	PREPARED BY W. Grover	DATE June 17/55
		CHECKED BY	DATE June 55

Duct Warping:

Strain Energy due to shear:

Station	$K\tau_s$	$K\tau_s^2$	"3/8 Coeff	Product
0	$\overline{9}3.26467$	8698.29867	1	8698.29867
$S_1/3$	$\overline{9}2.16947$	8495.21120	3	25485.63360
$2S_1/3$	$\overline{8}8.88387$	7900.34235	3	23701.02705
S_1	$\overline{8}3.40787$	6956.87278	1	6956.87278

$\Sigma = 64841.83210$

$I = \cdot 555(64841.83210) = 35987.21682$

Station	$K\tau_s$	$K\tau_s^2$	Weddle Coeff's	Product
S_1	$\overline{8}3.40787$	6956.87278	1	6956.87278
$S_1 + \frac{1}{6}S_2$	$\overline{5}8.70480$	3446.25354	5	17231.26770
$S_1 + \frac{1}{3}S_2$	$\overline{2}3.42699$	548.91757	1	548.91757
$S_1 + \frac{1}{2}S_2$	15.51036	240.57127	6	1443.42762
$S_1 + \frac{2}{3}S_2$	50.48645	2548.88163	1	2548.88163
$S_1 + \frac{5}{6}S_2$	74.64878	5572.44036	5	27862.20180
S	83.26569	6933.17513	1	6933.17513

$\Sigma = 63524.74423$

$$I = \frac{23.75044}{20}\left[63524.74423\right]$$

$$= 75437.03132$$

$$\text{total } I = 111424.24814$$

$$2V = \frac{.032\, I}{22 \times 4{,}000 \times (9.919571)^2}\left[\sigma_{W_1} - \sigma_{W_2}\right]^2$$

$$= \cdot 000411776\left[\sigma_{W_1} - \sigma_{W_2}\right]^2$$

over 1/4 of the section

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0510/9

SHEET No. 9-42

AIRCRAFT: C 105	Centre Fuselage	PREPARED BY W. Grover	DATE June 17/55
		CHECKED BY	DATE June 55

Duct Warping:

Summary: Station Midway between 255 & 320

Force on Frame:

$F_1 = \cdot 146763 \ \sigma_{w_1}$

Strain Energies:

a) Axial

$$2V = \cdot 001234912 [\sigma_{w_1}^2 + \sigma_{w_2}^2] + \cdot 000617456 [2\sigma_{w_1} \sigma_{w_2}]$$

b) Shear

$$2V = \cdot 001647104 [\sigma_{w_1}^2 + \sigma_{w_2}^2] - \cdot 001647104 [2\sigma_{w_1} \sigma_{w_2}]$$

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

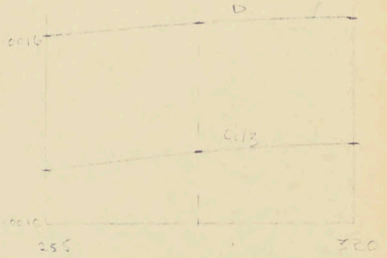
REPORT No. 7/0510/19
SHEET No. 9-43

AIRCRAFT: C105	Centre Fuselage	PREPARED BY: W. Grover	DATE: June 22/55
		CHECKED BY: β.	DATE: June/55

Duct Warping:

Values of Ci's & D's

Station	Ymax	Ci/3	D
255	12.14306	·001 174 32	·001 605 40
Midway	9.919571	·001 234 91	·001 647 10
320		·001 251 88	·001 665 60



$Ci/3 = \alpha x^2 + \beta x + \gamma$

$·001\ 174\ 32 = \gamma$

$·001\ 234\ 91 = 9\alpha L^2 + 3\beta L + \gamma$

$·001\ 251\ 88 = 36\alpha L^2 + 6\beta L + \gamma$

$·000\ 060\ 59 = 9\alpha L^2 + 3\beta L$

$·000\ 077\ 56 = 36\alpha L^2 + 6\beta L$

$·000\ 242\ 36 = 36\alpha L^2 + 12\beta L$

$·000\ 164\ 80 = 6\beta L$

$\beta L = ·000\ 027\ 4\dot{6}$

$36\alpha L^2 = -·000\ 087\ 24$

$\alpha L^2 = -·000\ 002\ 42\dot{3}$

$D = \alpha x^2 + \beta x + \gamma$

$·001\ 605\ 40 = \gamma$

$·001\ 647\ 10 = 9\alpha L^2 + 3\beta L + \gamma$

$·001\ 665\ 60 = 36\alpha L^2 + 6\beta L + \gamma$

$·000\ 041\ 70 = 9\alpha L^2 + 3\beta L$

$·000\ 060\ 20 = 36\alpha L^2 + 6\beta L$

$·000\ 166\ 80 = 36\alpha L^2 + 12\beta L$

$·000\ 106\ 60 = 6\beta L$

$\beta L = ·000\ 017\ 77$

$36\alpha L^2 = -·000\ 046\ 40$

$\alpha L^2 = -·000\ 001\ 29$

nL	Stn	Ci/3	D	Coeff (i)	Cix Coeff's
0	255	·001 174 32		·002 796 20	·002796 20
L		·001 199 37	·001 621 88	$\overline{·}$001 021 20	$\overline{·}$001 022 20
				·002 841 45	·005 708 12
2L		·001 219 57		·002 866 67	
3L		·001 234 91	·001 647 10	$\overline{·}$001 029 64	$-$·001 029 64
				·002 892 52	·005 799 94
4L		·001 245 42		·002 907 42	
5.L		·001 251 07	·001 662 00	$\overline{·}$001 036 46	$\overline{·}$001 036 46
6L	320	·001 251 88		·002 913 88	·002 913 88

NOTE: NORMALLY x measured from STN 320 - CHECK calcs indicate no difference in six decimals when calculated from this stn.

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT No 7/0510/9

SHEET No 9 43 ADD 1

PREPARED BY	DATE
W. Grover	June 26/55
CHECKED BY	DATE
	[illegible]

AIRCRAFT: C 105

Centre Fuselage

"W" Group - Station 255 - 320

x	Station	Frame Load	Frame Load
0	320	·148055	·148055
L			
2L			·148227
3L	mid	·146763	
4L			·144267
5L			
6L	255	·136175	·136175

$$F_L = \alpha x^2 + \beta x + \delta$$

$$\cdot 148055 = \delta$$

$$\cdot 146763 = 9\alpha L^2 + 3\beta L + \delta$$

$$\cdot 136175 = 36\alpha L^2 + 6\beta L + \delta$$

$$-\cdot 001292 = 9\alpha L^2 + 3\beta L$$

$$-\cdot 011880 = 36\alpha L^2 + 6\beta L$$

$$-\cdot 002584 = 18\alpha L^2 + 6\beta L$$

$$-\cdot 009296 = 18\alpha L^2$$

$$\alpha L^2 = -\cdot 00051644$$

$$6\beta L = \cdot 006712$$

$$\beta L = \cdot 00111867$$

Now $P = \tau_m t f h t$

$$\therefore \tau_m = \frac{P}{t f h t} = \frac{P}{\cdot 262475096}$$

Station	Forward & Aft	Centre Frame
254.5	A ·518811	—
276.5	A/ ·549641	1.099282
298.5	F/ ·564728 A/ 1·129456	1.129456
320.5	F/ ·564073	1.128146

Note: Stress Point "112" - i.e. former 254·5 is considered rigid.

Note: To the above value at 320·5 for centre frame is added the value 1.128143 to obtain a balanced coefficient for the warping group 109

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

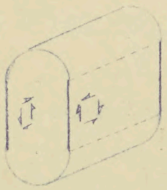
REPORT No 7/0510/9		SHEET No 9 - 44	
AIRCRAFT: C105	CENTRE FUSELAGE.	PREPARED BY C.B.	DATE May 19 '55
		CHECKED BY J. L. Gardener	DATE June '55.

DUCT - CROSS TERMS.

IT CAN BE SHOWN BY A STUDY OF THE STRESSES THAT COUPLING ONLY EXISTS FOR $\sigma_{b_1}\,\sigma_{b_2}$, $\sigma_{w_1}\,\sigma_{w_2}$, $\tau\sigma_{w_1}$ & $\tau\sigma_{w_2}$.

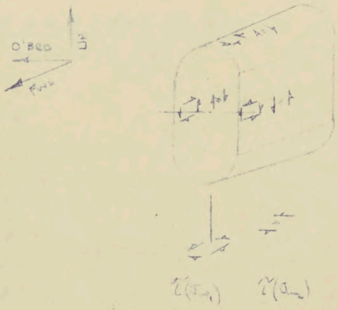
$\sigma_{b_1}\,\sigma_{b_2}$ & $\sigma_{w_1}\,\sigma_{w_2}$ HAVE PREVIOUSLY BEEN COMPUTED.

TORSION & WARPING - SHEAR TERMS ONLY AFFECTED COUPLING



$\tilde{\tau}_T$ = CONSTANT

TORSION



WARPING

TOTAL SHEAR STRESS = $\tilde{\tau}_T + \tilde{\tau}_{(\sigma w_1)} - \tilde{\tau}_{(\sigma w_2)} = \tilde{\tau}$

SHEAR ENERGY $2V = \frac{1}{G}\iint \tilde{\tau}^2 t\, ds\, dx$

$$= \underbrace{\frac{1}{G}\iint\left(\tilde{\tau}_T^{\,2} + \tilde{\tau}_{\sigma w_1}^{\,2} + \tilde{\tau}_{\sigma w_2}^{\,2} - 2\tilde{\tau}_{\sigma w_1}\tilde{\tau}_{\sigma w_2}\right) t\, ds \cdot dx}_{\text{ALREADY CALCULATED}} + \frac{2}{G}\iint \tilde{\tau}_T\left(\tilde{\tau}_{\sigma w_1} - \tilde{\tau}_{\sigma w_2}\right) t \cdot ds \cdot dx$$

$$\frac{2t \cdot \tilde{\tau}_T}{G}\int -\left(\frac{\sigma_{w_1} - \sigma_{w_2}}{y(s)_{max}\, L}\right) L\, \bar{y}(s)\, ds$$

where $\bar{y}(s)$ is the coefficient previously determined for warping shear and & $y(s)_{max}$ is the max coefficient previously determined for warping axial stress

REPORT NO 7/0510/9
SHEET NO 9-4

AIRCRAFT	PREPARED BY	DATE
C 105	W. Grover	June 20/55
Centre Fuselage	CHECKED BY	DATE
	B	June [illegible] 55

Duct Cross Terms: Station 255

S	Coeff	$K r_s$	② × ③	$\int K r_s\, ds$
0	1	$\bar{1}15.62461$	$\bar{1}15.62461$	
2.960000	3	$\bar{1}11.24381$	$\bar{3}33.73143$	1.11 × − 819.85768 = −910.0420248
5.920000	3	$\bar{9}8.10141$	$\bar{2}94.30423$	
8.88000	1	$\bar{7}6.19741$	$\bar{7}6.19741$	
12.084424	5	$\bar{4}3.37694$	$\bar{2}16.88470$	
15.288848	1	$\bar{5}.26622$	$\bar{5}.26622$	
18.493272	6	32.49957	194.99742	.9613274 × 481.67115 = 463.0436728
21.697696	1	64.33598	64.33598	
24.902120	5	85.54260	427.71300	I = −446 99835
28.106550	1	92.97308	92.97308	

$$2V = 2\gamma_r(\sigma_{w_1}-\sigma_{w_2}) \times \frac{.032 \times 4 \times -446\,99835}{4{,}000 \times 12.14306} = .001\,177\,95 \times 2\gamma_r(\sigma_{w_1}-\sigma_{w_2})$$

Station Midway between 255 & 320

S	coeff	$K r_s$	② × ③	
0	1	$\bar{9}3.26467$	$\bar{9}3.26467$	
$S_1/3$	3	$\bar{9}2.16947$	$\bar{2}76.50841$	.555 × − 719.83256 = − 399 50707
$2S_1/3$	3	$\bar{8}8.88387$	$\bar{2}66.65161$	
4.44000	1	$\bar{8}3.40787$	$\bar{8}3.40787$	
8.39841	5	$\bar{5}8.70480$	$\bar{2}93.52400$	
12.35681	1	$\bar{2}3.42899$	$\bar{2}3.42899$	
16.31522	6	15.51036	93.06216	1.187522 × 199.69734 = 237.14498
20.27363	1	50.48645	50.48645	
24.23203	5	74.64878	373.24390	I = $\bar{1}62.36209$
28.19044	1	83.26569	83.26569	

$$2V = \frac{2.4\,T_{dl}(\sigma_{w_1}-\sigma_{w_2}) \times .032 \times 162.36209}{4{,}000 \times 9.919571} = .000\,583\,77 \times 2\,T_{dl}(\sigma_{w_1}-\sigma_{w_2})$$

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0510/9

SHEET No. 9-46

AIRCRAFT		PREPARED BY	DATE
C105	Centre Fuselage	W. Grover	June 20/55
		CHECKED BY	DATE
		B	July 55

Duct Cross Terms:

Station		K
0	·001 177 95	
L		·000 945 40
2L		
3L	·000 523 77	·000 523 77
4L		
5L		·000 159 02
6L	0	

320

$$K = \alpha x^2 + \beta x + \gamma$$

$$\cdot 001\ 177\ 95 = \gamma$$

$$\cdot 000\ 523\ 77 = 9\alpha L^2 + 3\beta L + \gamma$$

$$0 = 36\alpha L^2 + 6\beta L + \gamma$$

$$-\cdot 000\ 654\ 18 = 9\alpha L^2 + 3\beta L$$

$$-\cdot 001\ 177\ 95 = 36\alpha L^2 + 6\beta L$$

$$-\cdot 002\ 616\ 72 = 36\alpha L^2 + 12\beta L$$

$$6\beta L = -\cdot 001\ 438\ 77$$

$$\beta L = -\cdot 000\ 239\ 795$$

$$36\alpha L^2 = \cdot 000\ 260\ 82$$

$$\alpha L^2 = \cdot 000\ 007\ 245$$

Note:

x is normally calculated from station 320 but no difference in six decimals is apparent when the calculations are reversed.

FORM 1310A

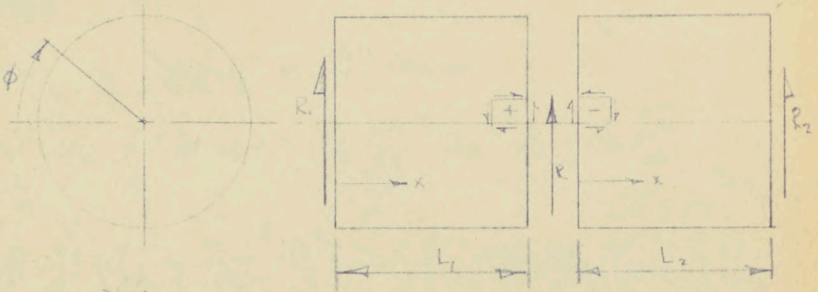
AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0510/9	SHEET No. 9-47	
PREPARED BY CB	DATE May 9. 55	
CHECKED BY J F Gardener	DATE June '55	
AIRCRAFT: C105	CENTRE FUSELAGE	

CIRCULAR DUCT

"B" GROUP



$$\frac{\partial\sigma}{\partial x} + \frac{\partial\tau}{r\,\partial\phi} = 0 \; ; \quad \frac{\partial\tau}{\partial x} = 0$$

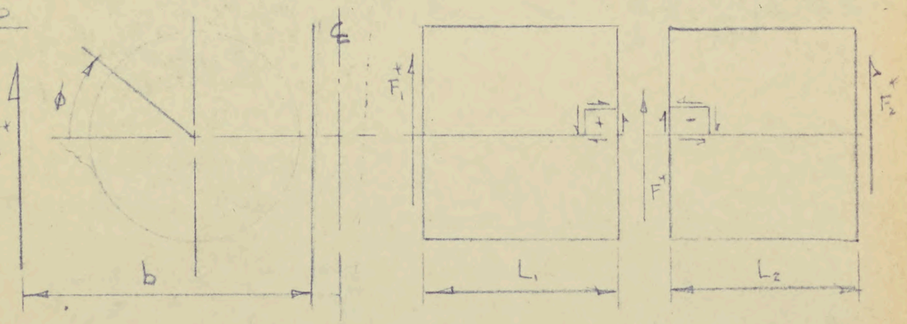
$$\sigma = \sigma_b \sin\phi \frac{x}{L_1} \qquad \sigma = \sigma_b \sin\phi \left(1 - \frac{x}{L_2}\right)$$

$$\tau = \sigma_b \cos\phi \frac{r}{L_1} \qquad \tau = -\sigma_b \cos\phi \frac{r}{L_2}$$

$$R_1 = -\oint \tau t \cos\phi \, r\, d\phi = -\sigma_b \pi \frac{r^2 t}{L_1} \text{(dn)} \qquad R_2 = -\oint \tau t \cos\phi\, r\, d\phi = -\sigma_b \pi \frac{r^2 t}{L_1} \text{(dn)}$$

$$R = \sigma_b \pi r^2 t \left(\frac{1}{L_1} + \frac{1}{L_2}\right) \text{(up) (Acting on the duct)}.$$

"W" GROUP



$$\sigma = \sigma_w \sin 2\phi \frac{x}{L_1} \qquad \sigma = \sigma_w \sin 2\phi \left(1 - \frac{x}{L_2}\right)$$

$$\tau = \sigma_w \frac{1}{2} \cos 2\phi \frac{r}{L_1} \qquad \tau = -\sigma_w \frac{1}{2} \cos 2\phi \frac{r}{L_2}$$

$$F_1^* b = -\oint \tau t r\, d\phi \cdot \cos\phi \cdot r \cos\phi = -\sigma_w \frac{r^3 t}{2L_1} \oint \cos^2\phi \cos 2\phi \, d\phi$$

$$F_1^* = -\sigma_w \frac{\pi}{4} \frac{r^3 t_d}{b L_1} \text{(down)}. \quad F_2^* = -\sigma_w \frac{\pi}{4} \frac{r^3 t_d}{b L_2} \qquad F^* = \sigma_w \frac{\pi}{4} \frac{r^3 t_d}{b}\left(\frac{1}{L_1} + \frac{1}{L_2}\right)$$

ACTING ON THE DUCT.

AIRCRAFT: C105	CENTRE FUSELAGE	PREPARED BY CB.	DATE May 7-55
		CHECKED BY O Hardenas	DATE June '55

DUCT ENERGY.

CIRCULAR DUCT

DUE TO BENDING & SHEAR – "B" GROUP

$$V = \frac{t}{2E} \oint \int_0^{L_1} \left[\sigma_{b_2} \frac{x}{L_1} + \sigma_{b_1}\left(1 - \frac{x}{L_1}\right)\right]^2 \sin^2\phi \, r \, dx \, d\phi$$

$$+ \frac{t}{2G_1} \oint \int_0^{L_1} \left(\sigma_{b_2} - \sigma_{b_1}\right)^2 \frac{r^2}{L_1^2} \cos^2\phi \, r \, dx \, d\phi$$

$$= \frac{\pi r t}{2E} \frac{L_1}{3} \left(\sigma_{b_2}^2 + \sigma_{b_2}\sigma_{b_1} + \sigma_{b_1}^2\right) + \frac{\pi r^3 t}{2G L_1}\left(\sigma_{b_2}^2 - 2\sigma_{b_2}\cdot\sigma_{b_1} + \sigma_{b_1}^2\right)$$

$$= \frac{1}{2}\left[\frac{\pi r^3 t}{G L_1}\left[\sigma_{b_1}^2 - 2\sigma_{b_2}\sigma_{b_1} + \sigma_{b_2}^2 + \frac{1}{3}\frac{G}{E}\cdot\frac{L_1^2}{r^2}\left(\sigma_{b_1}^2 + \sigma_{b_1}\sigma_{b_2} + \sigma_{b_2}^2\right)\right]\right]$$

DUE TO WARPING & SHEAR – "W" GROUP

$$V = \frac{1}{2}\left[\frac{\pi r^3 t}{4G L_1}\left[\sigma_{w_1}^2 - 2\sigma_{w_1}\sigma_{w_2} + \sigma_{w_2}^2 + \frac{4}{3}\frac{G}{E}\frac{L_1^2}{r^2}\left(\sigma_{w_1}^2 + \sigma_{w_1}\sigma_{w_2} + \sigma_{w_2}^2\right)\right]\right]$$

BOTH OF FORM

$$\begin{matrix} 1 + \frac{\varepsilon}{3} & -1 + \frac{\varepsilon}{6} \\ -1 + \frac{\varepsilon}{3} & 1 + \frac{\varepsilon}{3}. \end{matrix}$$

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C105	CENTRE FUSELAGE	REPORT No 7/0510/9	
		SHEET No 9-49	
		PREPARED BY	DATE
		C.B.	May 11 '55
		CHECKED BY	DATE
		O Hardene	June '55.

CIRCULAR DUCT

DUCT IN TORSION:

$$2V = \frac{2\pi \cdot r \cdot L\, t_d}{G}\,\tau_{dl}^2$$

τ_{dl}, E } Kips/Sq. In.

Stress Point 407

$$= \frac{2\pi \times 18.000 \times 16.250 \times .032}{4,000}\,\tau_{dl}^2$$

$$= \underline{0.014\,702\,50}\quad \tau_{dl}^2$$

Stress Points 207, 219, 231, 307, 319, 331

$$= \frac{2\cdot\pi \times 18.000 \times 22.000 \times .032}{4,000}\,\tau_{dl}^2$$

$$= \underline{0.019\,904\,92}\quad \tau_{dl}^2$$

DUCT IN BENDING

$$2V = \frac{\pi r^3 t}{GL}\left[\sigma_{b_1}^2 - 2\sigma_{b_1}\sigma_{b_2} + \sigma_{b_2}^2 + \frac{1}{3}\,\frac{G}{E}\times\frac{L^2}{r^2}\left(\sigma_{b_1}^2 + \sigma_{b_1}\sigma_{b_2} + \sigma_{b_2}^2\right)\right]$$

Stress Point 406

$$= \frac{\pi \times 18.000^3 \times .032}{4,000 \times 16.25}\left[\sigma_{b_1}^2 - 2\sigma_{b_1}\sigma_{b_2} + \sigma_{b_2}^2 + \frac{1}{3}\times\frac{4,000}{10,600}\times\frac{16.250^2}{18.000^2}\left(\sigma_{b_1}^2 + \sigma_{b_1}\sigma_{b_2} + \sigma_{b_2}^2\right)\right]$$

$$= 0.009\,019\,85\left[\sigma_{b_1}^2 - 2\sigma_{b_1}\sigma_{b_2} + \sigma_{b_2}^2 + 0.102\,516\,69\left(\sigma_{b_1}^2 + \sigma_{b_2}\sigma_{b_1} + \sigma_{b_2}^2\right)\right]$$

$$= \underline{0.009\,944\,54\left(\sigma_{b_1}^2 + \sigma_{b_2}^2\right) - 0.008\,557\,56 \times 2\,\sigma_{b_1}\sigma_{b_2}}$$

Stress Points 206, 218, 230, 306, 312, 331

$$= 0.006\,662\,39\left[\sigma_{b_1}^2 - 2\sigma_{b_1}\sigma_{b_2} + \sigma_{b_2}^2 + 0.187\,902\,78\left(\sigma_{b_1}^2 + \sigma_{b_2}\sigma_{b_1} + \sigma_{b_2}^2\right)\right]$$

$$= \underline{0.007\,914\,27\left(\sigma_{b_1}^2 + \sigma_{b_2}^2\right) - 0.006\,036\,45 \times 2\,\sigma_{b_1}\sigma_{b_2}}$$

AVRO AIRCRAFT LIMITED TECHNICAL DEPARTMENT (Aircraft)		REPORT No. 7/0510/9 SHEET No. 9-50	
AIRCRAFT:		PREPARED BY	DATE
C105	CENTRE FUSELAGE	C.B.	May 11. 55
		CHECKED BY	DATE
		J Gardener	June 55

CIRCULAR DUCT.

DUCT IN WARPING.

$$2V = \frac{\pi r^3 t_D}{4GL}\left[\sigma_{w_1}^2 - 2\sigma_{w_1}\sigma_{w_2} + \sigma_{w_2}^2 + \frac{4}{3}\cdot\frac{G}{E}\cdot\frac{L^2}{r^2}\left(\sigma_{w_1}^2 + \sigma_{w_1}\sigma_{w_2} + \sigma_{w_2}^2\right)\right]$$

STRESS POINT 405

$$= \frac{\pi \times 18.000^3 \times .032}{4.000 \times 4{,}000 \times 16.25}\left[\sigma_{w_1}^2 - 2\sigma_{w_1}\sigma_{w_2} + \sigma_{w_2}^2 + \frac{4}{3} \times \frac{4{,}000}{10{,}600} \times \frac{16.250^2}{18.000^2}\left(\sigma_{w_1}^2 + \sigma_{w_1}\sigma_{w_2} + \sigma_{w_2}^2\right)\right]$$

$$= 0.002\,254\,98\left[\sigma_{w_1}^2 - 2\sigma_{w_1}\sigma_{w_2} + \sigma_{w_2}^2 + .410\,066\,76\left(\sigma_{w_1}^2 + \sigma_{w_1}\sigma_{w_2} + \sigma_{w_2}^2\right)\right]$$

$$= \underline{\underline{0.003\,179\,67\left(\sigma_{w_1}^2 + \sigma_{w_2}^2\right) - 0.001\,792\,64 \times 2\,\sigma_{w_1}\sigma_{w_2}}}$$

STRESS POINTS 205, 217, 229 305, 317, 329

$$= \frac{\pi \times 18.000^3 \times .032}{4.000 \times 4{,}000 \times 22.00}\left[\sigma_{w_1}^2 - 2\sigma_{w_1}\sigma_{w_2} + \sigma_{w_2}^2 + \frac{4}{3} \times \frac{4{,}000}{10{,}600}\;\frac{22.000^2}{18.000^2}\left(\sigma_{w_1}^2 + \sigma_{w_1}\sigma_{w_2} + \sigma_{w_2}^2\right)\right]$$

$$= 0.001\,665\,60\left[\sigma_{w_1}^2 - 2\sigma_{w_1}\sigma_{w_2} + \sigma_{w_2}^2 + 0.751\,611\,12\left(\sigma_{w_1}^2 + \sigma_{w_1}\sigma_{w_2} + \sigma_{w_2}^2\right)\right]$$

$$= \underline{\underline{0.002\,917\,48\left(\sigma_{w_1}^2 + \sigma_{w_2}^2\right) - 0.001\,039\,66 \times 2\,\sigma_{w_1}\sigma_{w_2}}}$$

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO	7/0510/9		
SHEET NO	9 - 51	STARTS @ 9-0	
AIRCRAFT: C105	CENTRE FUSELAGE	PREPARED BY C.B.	DATE May 25-'55
		CHECKED BY J.F. Gardener	DATE June '55

DUCT - REACTION ON FRAME DUE TO CIRCULAR DUCT

"B" GROUP.

$$R_1 = R_2 = -\sigma_D \frac{\pi r^2 t}{L} = -\sigma_D \left(\frac{3.141\,593 \times 18.000^2 \times .032}{L} \right)$$

DUE TO STRESS POINTS 206, 218, 231, 306, 318 & 331

$$R_1 = R_2 = + \underline{\sigma_D \times 1.480\,547\,10}$$

DUE TO STRESS POINT 406

$$R_1 = R_2 = - \underline{\sigma_D \times 2.004\,453\,00}$$

"W" GROUP.

$$F_1^* = F_2^* = -\sigma_w \frac{\pi}{4} \frac{r^3 t}{b L} = -\sigma_w \cdot \frac{3.141\,593}{4} \times \frac{18.000^3 \times .032}{45{,}000 \quad L}$$

DUE TO STRESS POINTS 205, 217, 229, 305, 317, 329

$$= - \underline{\sigma_w \times 0.148\,054\,7}$$

DUE TO STRESS POINT 405

$$= - \underline{\sigma_w \times 0.200\,4433}$$

R's at centre of opposite sign of sum of R_1 & R_2.

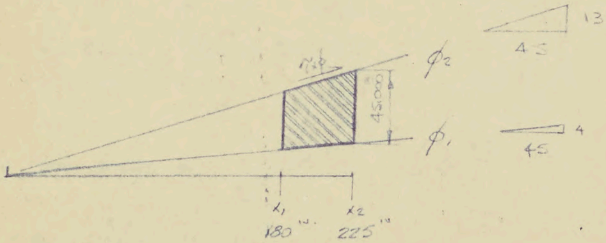
FORM 1519A

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No	7/05 10/9		
SHEET No	10-1		
AIRCRAFT: C105	CENTRE FUSELAGE	PREPARED BY: CB	DATE: May 4 · 55
		CHECKED BY:	DATE:

IDEALIZED FRAME:

FOLLOWING GEN/1090/330 Pg 6 & subs.



$$2V = \frac{At}{G}\,\tilde{\tau}_m^2\left(1 + \frac{4G}{3E}\left(\tan^2\phi_2 + \tan\phi_2\tan\phi_1 + \tan^2\phi_1\right)\right)$$

$$= \frac{(225^2 - 180^2)\,t\,\tilde{\tau}_m^2}{10\,G}\left(1 + \frac{4G\,(13^2 \times 4 \times 13 \times 4^2)}{3E \times 45^2}\right)$$

$$= \frac{1{,}822.500\,000\;t\,\tilde{\tau}_m^2}{G} + \frac{284.399\,849\;t\,\tilde{\tau}_m^2}{E}$$

G = 4,090 K.s.i

E_t = 10,500 k.s.i } Avg 10,600 k.s.i

E_c = 10,700 k.s.i }

Ref AN.C.5 P45 24ST UNCLAD SHEET.

$$\therefore\; 2V = \underline{472.455\,174 \times 10^{-3}\; t\,\tilde{\tau}_m^2}$$ PER SIDE

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

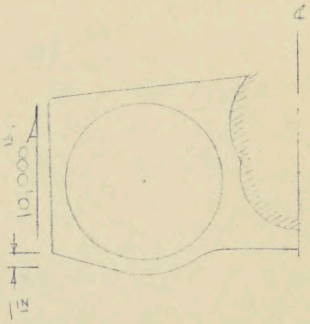
REPORT No. 7/0510/9
SHEET No. 10-2

AIRCRAFT		PREPARED BY	DATE
C105	CENTRE FUSELAGE	C.B.	May 4 55
		CHECKED BY	DATE

FRAMES:

FORMERS 268 TO 469 INC.

STIFFNESS UNDER FRAME LOADING
AS SHOWN = 10,000 lb/IN.
= 10 KIPS/IN.
REF.: R. WADE APRIL 21. 55

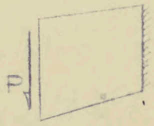


ENERGY PER SIDE $= \frac{1}{2}\delta P = \frac{1}{2} K P^2$

$K = \delta / P = 10^{-1}$

EQUATING ENERGY.

FOLLOWING GEN/1090/330 Pg 6 & subs.



$P = \tau_{12\phi}\, t\, x_2 (\tan\phi_2 - \tan\phi_1)$

$= \tau_m x_1 t (\tan\phi_2 - \tan\phi_1) = 36\tau_m t.$

$\therefore \quad 36^2 \times 10^{-1} \tau_m^2 t^2 = 472.455\,174 \times 10^{-3} \tau_m^2 t$

$t = \frac{472.455\,174}{360^2} = .003\,645\,487\,45$

$\therefore 2V = 1.722\,329\,408 \times 10^{-3} \tau_m^2.$ PER HALF FRAME.

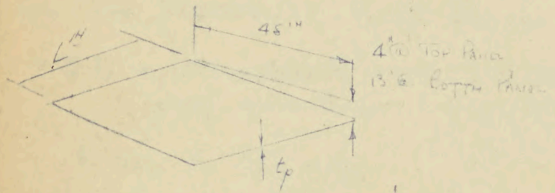
$\therefore \tau_m = \frac{P}{36 \times .003\,645\,487\,45} = 7.6\,197\,705\,P.$ For one web

$= 3.8\,098\,852\,P$ FOR TWO webs.

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No	7/0510/9		
SHEET No	11 - 1		
AIRCRAFT		PREPARED BY	DATE
C105	CENTRE FUSELAGE	CB.	MAY 23.55
		CHECKED BY	DATE
		E.Ce.	

PANELS –



$$2V = \frac{A t \tau_p^2}{G}$$

$$A_{p_1} = \sqrt{45^2 + 4^2} \times 16.25 = 734.133\ 205$$

or

$$\sqrt{45^2 + 4^2} \times 22.00 = 993.303\ 416$$

$$A_{p_2} = \sqrt{45^2 + 13^2} \times 16.25 = 761.152\ 499$$

or

$$\sqrt{45^2 + 13^2} \times 22.00 = 1030.483\ 384$$

$$t = .040 \times \frac{6}{10} = .024''$$

G = 4,000 to .625 × 4,000 Take at .80 × 4,000 = 3,200.

	L =	22"	16.25"
2V =	Upper	.007 454 28 τ_{p}^2	.005 506 00 τ_{p}^2
	Lower	.007 728 62 τ_{p}^2	.005 708 64 τ_{p}^2

FORM 1319A

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO 7/0510/9

SHEET NO 12-1

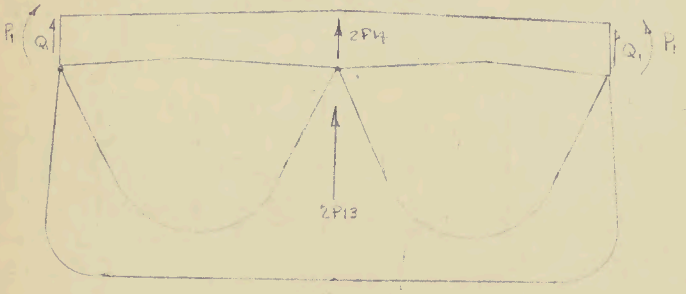
AIRCRAFT. C105	CENTRE FUSELAGE	PREPARED BY C.B.	DATE July '55
		CHECKED BY E.A.	DATE

BULKHEAD 485

FROM RPT 7/0556/27 Pg 112 REVISED

$$2 \cdot E \cdot U = 109 \cdot 6956\, P_0^2 + 62.5775\, Q_1^2 + 1382\, P_1^2 + 47.9414\, P_0 Q_1 - 3.9826\, P_0 P_1 + 2.0016\, Q_1 P_1$$

Pg. 107 $P_0 = P_{13} + P_{14}$.



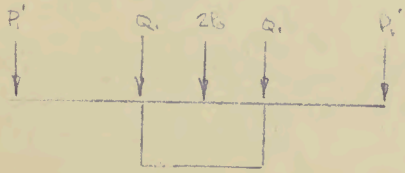
Ref Pg 1

REVERSING DIRECTION OF ALL FORCES TO AGREE WITH SIGN CONVENTION OF THE WING & FUSELAGE ANALYSES & DIVIDING BY E = 10,500 KSI

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

$$2V = \begin{array}{c|ccc} & P_1' & Q_1 & P_0 \\ \hline P_1' & 131.619 & .009531 & \overline{.018965} \\ Q_1 & .009531 & .005960 & .002283 \\ P_0 & \overline{.018965} & .002283 & .010447 \end{array}$$

NOTE: P_1' here has a value $P_1' = P_1/100$.

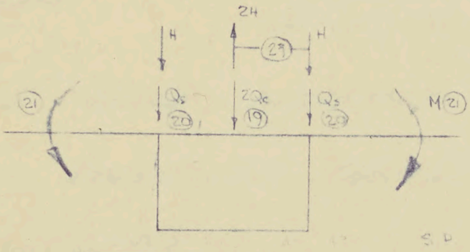


FORM 1319A

REPORT No	7/0510/9		
SHEET No	12-2		
AIRCRAFT: C105	CENTRE FUSELAGE	PREPARED BY: C.B.	DATE: July '55
		CHECKED BY: E.A.	DATE: Aug '55

BULKHEAD 485.

TRANSPOSE TO THIS SET OF LOADS.



$$\begin{vmatrix} P_1' \\ Q_1 \\ P_0 \end{vmatrix} = \begin{array}{cccc} 23 & 21 & 20 & 19 \\ \hline \end{array} \begin{vmatrix} 0 & .1 & 0 & 0 \\ +1 & 0 & 1 & 0 \\ -1 & 0 & 0 & 1 \end{vmatrix}$$

$$Z'' = \begin{vmatrix} \cdot C_{uv} \end{vmatrix} \begin{vmatrix} 0 & 1 & 0 & 0 \\ +1 & 0 & 1 & 0 \\ -1 & 0 & 0 & 1 \end{vmatrix}$$

$$\begin{vmatrix} 0 & +1 & -1 \\ 1 & 0 & 0 \\ 0 & 1 & 0 \\ 0 & 0 & 1 \end{vmatrix} = \begin{vmatrix} +.011841 & +.028496 & +.003677 & -.008160 \\ +.028496 & .131619 & .007531 & -.018965 \\ +.003677 & .007531 & .005960 & .002283 \\ -.008160 & -.018965 & .002283 & .010447 \end{vmatrix}$$

THESE VALUES CAN BE PLACED IN CIK 4 & VALUES, 1.000 000, PLACED IN THE TIa FOR STRESS POINTS. 19, 20, 21, 23.

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TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/9
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PREPARED BY	DATE
E. AUGUSTINE	11/OCT./55.
CHECKED BY	DATE

AIRCRAFT: C-105 | CENTRE SECTION

<u>METHOD.</u> — (THE FOLLOWING METHOD WAS DEVELOPED BY A. GRZEDZIELSKI).

FIG. 5 BELOW INDICATES THE BREAKDOWN OF THE STRUCTURE CONSIDERED INTO FREE BODY ELEMENTS.

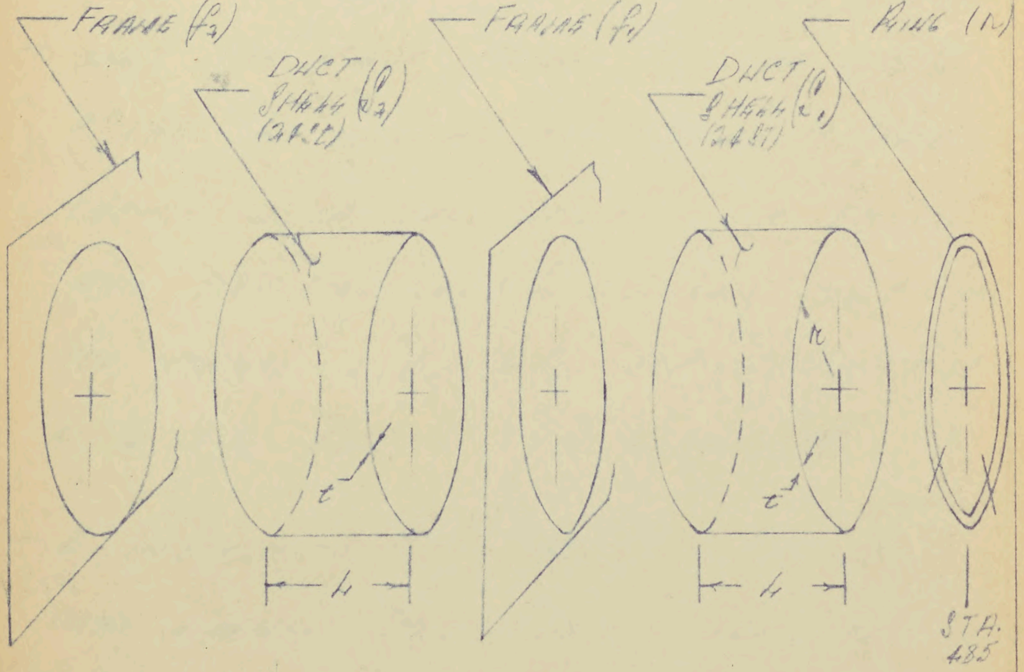

FIG. 5.

$t = .032$
$L = 16.25$
$R = 18.00$

SUBSCRIPT
- R RING
- S_1, S_2 DUCT SEGMENTS.

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AIRCRAFT: C-105	CENTRE SECTION	PREPARED BY E. AUGUSTINE	DATE 11/OCT./55
		CHECKED BY	DATE

ASSUMPTIONS —

- THE SECOND FRAME FWD. OF 485 (f_2) IS RIGID UNDER STRESSES INDUCED BY WING DISTORTION.
- THE DISTURBANCE DOES NOT PASS FWD. OF f_2.
- ONLY THE FIRST FRAME FWD. OF 485 (f_1) & THE TWO 16.25 IN. DUCT SEGMENTS (PROBLEM #4) DEFORM UNDER THE INFLUENCE OF THE WING DEFORMATION. (f_1 DEFORMS ONLY FOR THE TORSION TYPE WING DEFORMATION — IS RIGID FOR BENDING TYPE - REF. PG. 7).
- A CONSEQUENCE OF ASSUMPTION #2 IS THAT NORMAL STRESS AROUND PERIPHERY OF D_2 AT f_2 VANISHES.

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REPORT NO. 7/0510/9
SHEET NO. 13-1

PREPARED BY	DATE
G. AUGUSTINE	11/OCT-55
CHECKED BY	**DATE**

AIRCRAFT: C-105
CENTRE SECTION

SOLUTION OF LOCAL PROBLEM ASSOCIATED WITH DEFORMATION OF DUCT RING AT FUSELAGE STATION 485.

FORM 1319 A

REPORT NO. 7/0510/9
SHEET NO. 13-2

AIRCRAFT:		PREPARED BY	DATE
C-105	CENTRE SECTION	R. AUGUSTINE	11/OCT./55
		CHECKED BY	DATE

<u>INTRODUCTION</u> — IN THE BODY OF THE CENTRE FUSELAGE PROBLEM, FRAMES IN GENERAL ARE ASSUMED RIGID UNDER DUCT LOADING ARISING FROM THE TORSION & BENDING TYPES OF REDUNDANT GROUPS.

DUE TO THE FLEXIBLE NATURE OF ATTACHING THE DUCT TO FRAMES 485, THE ABOVE ASSUMPTION IS INVALID & THE IMPORTANCE OF INVESTIGATING THE INFLUENCE OF THE DISTORTION OF THE CONNECTING RING IS INDICATED.

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TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0510/9
SHEET No. 13-3

PREPARED BY	DATE
A. AUGUSTINE	11/OCT./56
CHECKED BY	DATE

AIRCRAFT:
C-105
CENTRE SECTION

DEFINITION OF PROBLEM.

1. GEOMETRY — THE ACTUAL & IDEALISED STRUCTURES ARE REPRODUCED IN FIG'S. 1 & 2 BELOW.

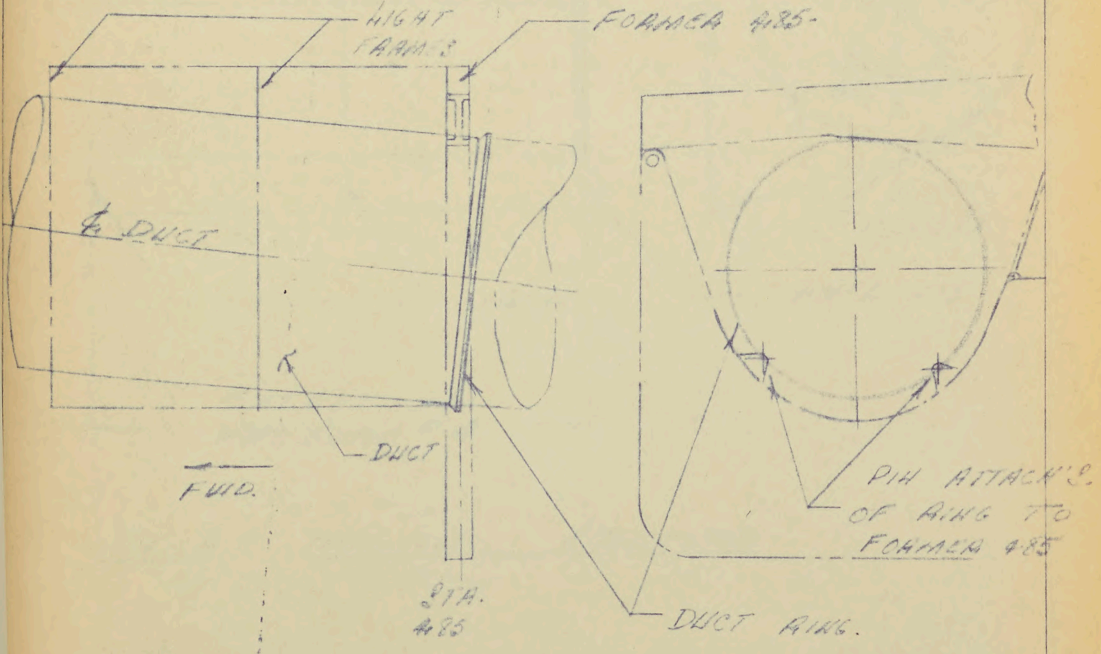

FIG. 1. ACTUAL STRUCTURE.

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TECHNICAL DEPARTMENT (Aircraft)

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E. AUGUSTINE	11/OCT/55
CHECKED BY	DATE

AIRCRAFT: C-105
CENTRE SECTION

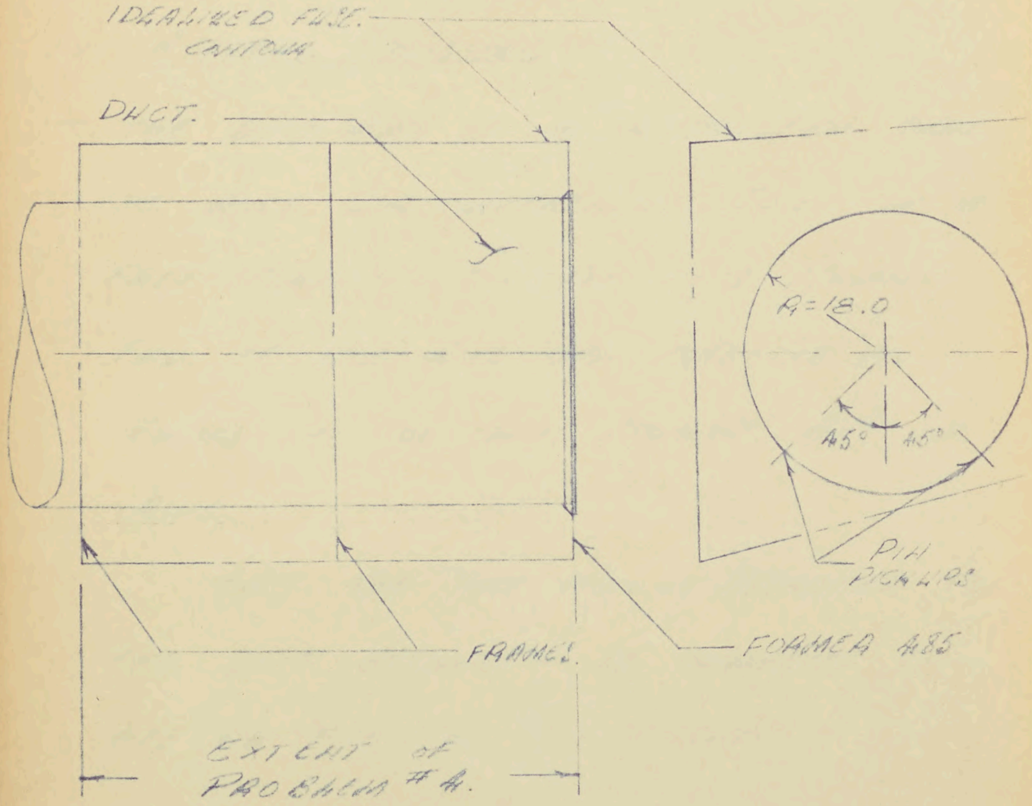

FIG. 2 – IDEALIZED STRUCTURE

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 485 VIA THE TWO PIN PICKUPS.

1.2. IDEALISED STRUCTURE —

— THE DISPLACEMENT OF THE DUCT ℄ FROM THE HORIZONTAL IS IGNORED.

— THE TWO PIN ATTACHMENTS ARE ASSUMED TO BE AT 45° EITHER SIDE OF DUCT ℄.

— THE TWO FRAMES ARE EACH THE SINGLE THICKNESS ($t = .0036$) TRAPEZOIDAL PANELS THAT

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REPORT No. 7/0510/9
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AIRCRAFT: C-105	CENTRE SECTION	PREPARED BY K. AUGUSTINE	DATE 11/OCT./55
		CHECKED BY	DATE

REPRESENT ONE ACTUAL FRAME (REF.).

a. FORCES ON RING —

CONSIDER THE RING AS A FREE BODY.

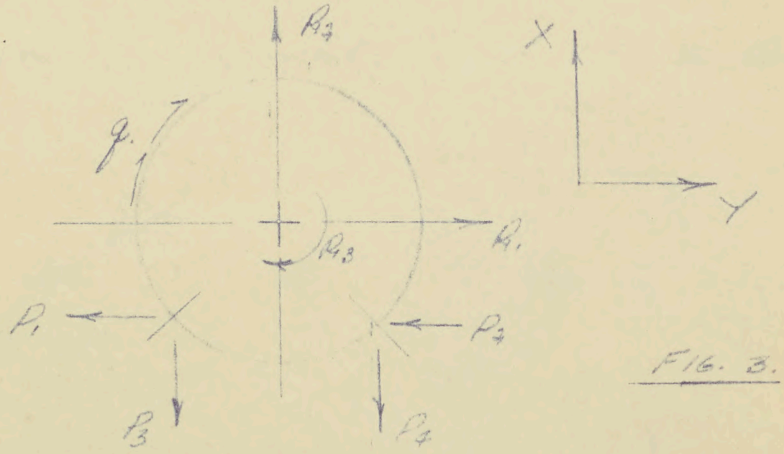

FIG. 3.

REPLACE THE SHEAR FLOW q ON FWD. FACE OF RING BY TWO FORCES R_1, R_2, & A TORQUE, R_3, APPLIED AT RING CENTRE. THE PIN ATTACHMENTS CAN SUSTAIN FORCES P_1 --- P_4 AS SHOWN.

THEN FOR EQUILIBRIUM —

$$\begin{array}{ccc} P_2 & P_3 & P_4 \\ \left|\begin{array}{ccc} 1 & 0 & 0 \\ 0 & 1 & 1 \\ 0 & \overline{1} & 1 \end{array}\right| \end{array} = \begin{array}{ccc} R_1 & R_2 & R_3 \\ \left|\begin{array}{ccc} 1 & 0 & 0 \\ 0 & 1 & 0 \\ \overline{h/d} & 0 & \overline{1/d} \end{array}\right| \end{array} + \begin{array}{c} P_1 \\ \left|\begin{array}{c} \overline{1} \\ 0 \\ 0 \end{array}\right| \end{array}$$

AIRCRAFT: C-105	CENTRE SECTION	REPORT NO 7/0510/9	
		SHEET NO 13-7	
		PREPARED BY	DATE
		Z. AUGUSTINE	11/OCT./52
		CHECKED BY	DATE

FROM WHICH, ASSUMING $P_1 = R_1 = 0$

$P_2 = 0$; $P_3 = \frac{1}{2} R_2 + \frac{1}{2} R_3/d$; $P_4 = \frac{1}{2} R_2 - \frac{1}{2} R_3/d$

i.e. THE REACTIONS, AT PIN ATTACHMENTS, TO THE DUCT SHEAR FLOW APPLIED TO THE RING MAY BE INDICATED AS IN FIG. 4 BELOW.

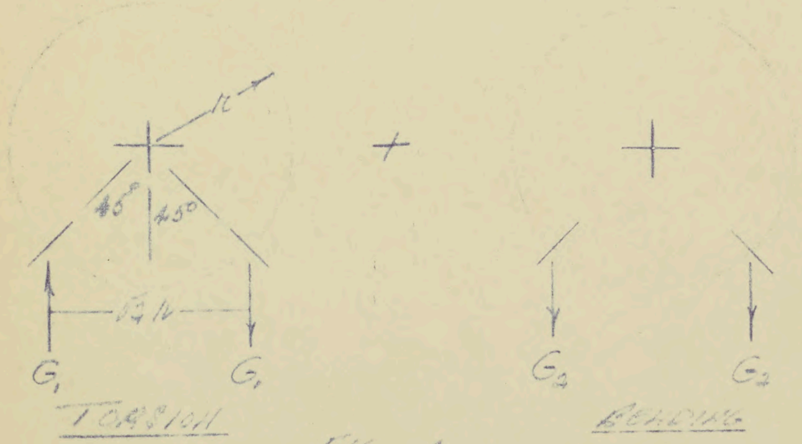

FIG. 4.

VIEW ON L.H.S. OF A/C LKG. FWD.

THESE TWO MODES OF RESISTANCE ARE CONSISTENT WITH THE TWO GROUPS - TORSION & BENDING - INTRODUCED IN THE BODY OF THE PROBLEM. THE HORIZONTAL REACTIONS AT THE PIN ARE IGNORED ($R_1 \equiv 0$). [SEE NEXT SECT. FOR SHIMS]

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REPORT NO. 7/0510/9
SHEET NO. 13·8

		PREPARED BY	DATE
AIRCRAFT: C-105	CENTRE SECTION	E. AUGUSTINE	11/OCT/58.
		CHECKED BY	DATE

THE PROBLEM THEN, IS THE CALCULATION OF THE WING DISTORTION EFFECT UNDER THE TORSION & BENDING TYPE REACTIONS ON THE LOCAL STRESS & ENERGY DISTRIBUTION IN THE DUCT.

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REPORT No 7/0510/9
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PREPARED BY F. AUGUSTINE
DATE 11/OCT./55.
CHECKED BY
DATE

AIRCRAFT: C-105
CENTRE SECTION

ESSENTIALLY, THE METHOD CONSISTS OF —

- REPLACING THE CONCENTRATED LOAD TYPE OF REACTIONS, G_1, G_3 (REF. PG. 7), BY APPROPRIATE 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 UNKNOWNS IN THE TRIG. SERIES BY THEOREM OF LEAST WORK.
- INCLUSION OF RESULTS IN THE BODY OF THE PROBLEM IN A MANNER CONSISTENT WITH EXISTING REDUNDANT GROUP DEFINITIONS.

ORM 1319A

A. V ROE CANADA LIMITED
MALTON - ONTARIO
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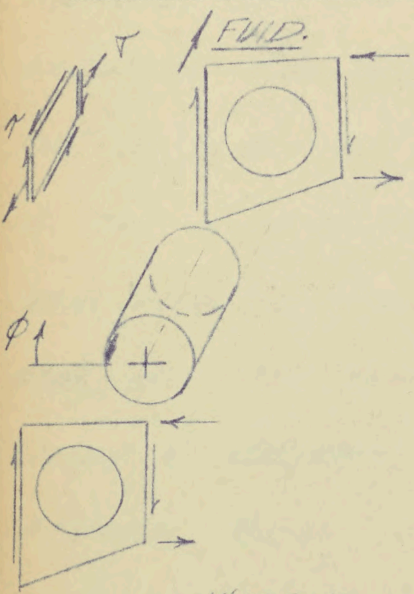
REPORT No. 7/0510/9
SHEET No. 13-12

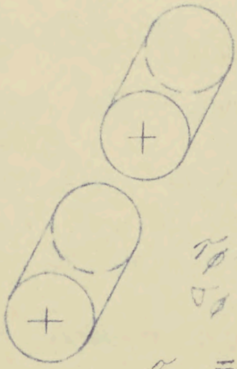
PREPARED BY	DATE
E. Ahlusthe	11/Oct./55
CHECKED BY	DATE

AIRCRAFT: C-105 | Centre Section

Sign Conventions & Relevant Stress Distributions - Non-inclusive of Ring Influence - are reproduced below for easy reference (Ref. pg. 5-2, 9-44)



TORSION



$\tau_\phi = \text{Const.} = \tau_{407}$

$\sigma_\phi \equiv 0$

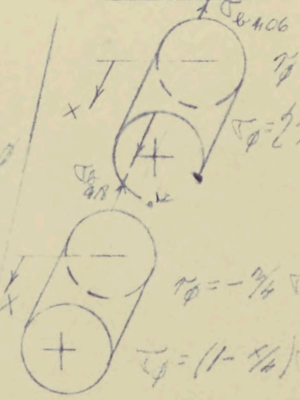
$\tau_\phi = \text{Const.} = \tau_{419}$

$\sigma_\phi \equiv 0$

$\tau_{407} = 1.990493\ T_{(402)}$

$\tau_{419} = 1.990493\ T_{(408)}$

BENDING



$\tau_\phi = x/L(\sigma_{b418} - \sigma_{b406}) \cos\phi$

$\sigma_\phi = \{x/L(\sigma_{b418} - \sigma_{b406}) + \sigma_{b406}\} \sin\phi$

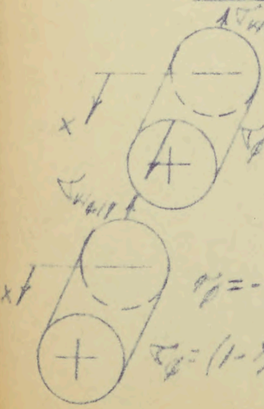
$\tau_\phi = -x/L\ \sigma_{b418} \cos\phi$

$\sigma_\phi = (1 - x/L)\sigma_{b418} \sin\phi$

$\sigma_{b418} = B(409)$

$\sigma_{b406} = B(111)$

WARPING



$\tau_\phi = x/2L(\sigma_{w419} - \sigma_{w405}) \cos 2\phi$

$\sigma_\phi = \{x/L(\sigma_{w419} - \sigma_{w405}) + \sigma_{w405}\} \sin 2\phi$

$\tau_\phi = -x/2L\ \sigma_{w419} \cos 2\phi$

$\sigma_\phi = (1 - x/L)\sigma_{w419} \sin 2\phi$

$\sigma_{w419} = W(406)$

$\sigma_{w405} = W(112)$

FORM 1319A

REPORT No. 7/0510/9
SHEET No 13.-13

AIRCRAFT:		PREPARED BY	DATE
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THE TORSION & BENDING STRESS DISTRIBUTIONS (DEFINED BY STRESS POINTS 410 & 418 ALSO) INDICATED ON THE PREVIOUS PAGE MUST BE ADJUSTED FOR THE EFFECTS OF WING DEFORMATION. THE DISTRIBUTIONS SHOWN ARE WHAT WOULD EXIST IF THE WING WERE RIGID — THEY WOULD THEN CORRESPOND TO SIMILAR DISTRIBUTIONS IN THE BODY OF THE PROBLEM WHERE UNDER THE TORSION & BENDING GROUPS THE FRAMES ARE ASSUMED RIGID.

REPLACE THE CONCENTRATED LOADS - GROUPS G_1, G_2 (REF. PG. 13) BY FOURIER SERIES SHEAR FLOWS

$$2r \cdot G_1 = 2\pi r^2 (\tau_{A19} \cdot t)$$

$$G_1 = 2\pi r (\tau_{A19} \cdot t)$$

$$G_2 = \frac{\pi r^2}{24} (\tau_{b_{A8}} \cdot t) \quad \text{— (REF. 9-47)}$$

SUBSCRIPTS -

r - RING
T - TORSION
b - BENDING
n - TERM OF SERIES.

REPLACING G_1 (TORSION) —

$$(\tau t)_{rT} = 2\left\{\tfrac{1}{2} + \sum_{n=2}^{\infty}\left(\cos\frac{n\pi}{4} - n \cdot \sin\frac{n\pi}{4}\right)\cos n\theta\right\}(\tau_{A19} \cdot t) \quad -(1)$$

REPLACING G_2 (BENDING) —

$$(\tau t)_{rb} = -\frac{2G_2}{\pi r}\left\{\sum_{n=1}^{\infty}\left(\sin\frac{n\pi}{4}\right) \cdot \sin n\theta\right\} \cdot G_2 \quad -(2)$$

AIRCRAFT: C-105	CENTRE SECTION	REPORT No. 7/0510/9	
		SHEET No. 13-15	
		PREPARED BY E. AUGUSTINE	DATE 11/OCT./55
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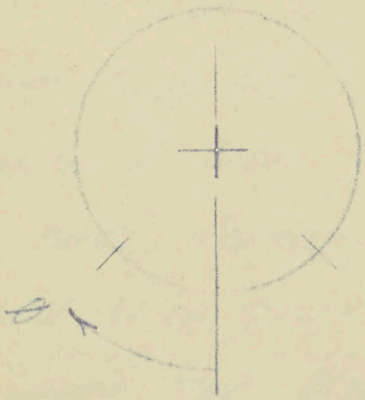
θ IN ABOVE EXPRESSIONS IS MEASURED FROM THE ORIGIN DEFINED IN FIG. 6. BELOW.



$\theta = (\pi/2 + \phi)$ (REF. PG. 12).

VIEW ON L.H.S. OF A/C. LKG. FWD.

WRITE —

$$G_3 = \pi r^2/32\,(\tau_{3.415}\cdot t)$$

$$2\,4/n\left(\cos\frac{n\pi}{4} - n\sin\frac{n\pi}{8}\right) = b_n$$

$$\sqrt{2}\left(\sin\frac{n\pi}{8}\right) = a_n$$

THEN (1), (2) PREVIOUS PAGE BECOME —

$$(r\tau)_{NT} = r/4\left\{(4/\pi) + \sum_{n=2}^{\infty} b_n\left(\cos\frac{n\pi}{2}\cdot\cos n\phi - \sin\frac{n\pi}{2}\cdot\sin n\phi\right)\right\}(\tau_{3.415}\cdot t) \quad —(3)$$

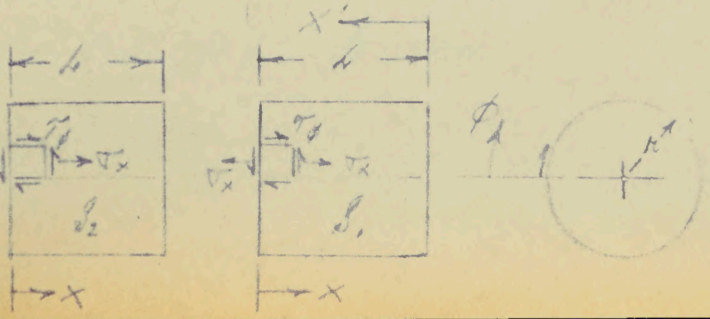
$$(r\tau)_{NG} = -r/4\left\{\sum_{n=1}^{\infty} a_n\left(\sin\frac{n\pi}{2}\cdot\cos n\phi + \cos\frac{n\pi}{2}\cdot\sin n\phi\right)\right\}(\tau_{3.418}\cdot t) \quad —(4)$$

A. V. ROE CANADA LIMITED MALTON - ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT No. 7/0510/9 SHEET No. 13-16	
AIRCRAFT:		PREPARED BY	DATE
		F. BAUGHETTINE	11/OCT/55
C-105	CENTRE SECTION	CHECKED BY	DATE

STRESS DISTRIBUTION IN DUCTS S_1 & S_2.

DUE TO THE RING DEFORMATION UNDER THE SHEAR FLOWS $(TW)_{RT}$ & $(TW)_{RL}$, & HENCE DUCT DEFORMATION, A STRESS DISTRIBUTION IS INDUCED IN THE DUCT SEGMENTS S_1 & S_2. (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 SHELL PLUS THE BOUNDARY CONDITIONS.



VIEW ON
L.H.S. A/C
LKG. FWD.

FORM 1319 A

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

REPORT NO. 7/0519/1
SHEET NO. 13-17
PREPARED BY: E. AUGUSTINE DATE: 11/OCT./55
CHECKED BY: DATE:

AIRCRAFT: C-105 CENTRE SECTION

DIFFERENTIAL EQUATIONS:—

$$\frac{\partial \sigma_x}{\partial x} + \frac{1}{r}\frac{\partial \tau}{\partial \phi} = 0$$

$$\frac{\partial \tau}{\partial x} + \frac{\partial \sigma_\phi}{r \partial \phi} = 0$$

WHERE σ_ϕ IS ASSUMED ZERO.

BOUNDARY CONDITIONS.—

1). $\sigma_{x_{s_1}} = 0$ AT $x = l$

2). $\sigma_{x_{s_2}} = 0$ AT $x = 0$

3). $\underset{(x=0)}{\sigma_{x_{s_1}}} \equiv \underset{(x=l)}{\sigma_{x_{s_2}}}$

4). EQUILIBRIUM CONDITIONS ON RING
— SEE NEXT SECTION.

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AIRCRAFT: C-105	CENTRE SECTION	REPORT NO. 7/0510/9	
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		PREPARED BY	DATE
		E. AUGUSTINE	18/OCT./55
		CHECKED BY	DATE

SOLUTION —

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

$$\tau = f(\phi)$$

$$\frac{\partial \sigma_x}{\partial x} = -\frac{1}{r} f'(\phi)$$

$$\sigma_x = -\frac{x}{r} f'(\phi) + g(\phi)$$

S_1 —

$$\sigma_{x S_1 (x=h)} = 0 = -h/r\, f'(\phi) + g(\phi)$$

$$g(\phi) = h/r\, f'(\phi)$$

$$\therefore \quad \tau_{S_1} = f_1(\phi)$$

$$\sigma_{x S_1} = \frac{(h-x)}{r} f_1'(\phi)$$

S_2 —

$$\sigma_{x S_2 (x=0)} = 0 = g(\phi)$$

$$\tau_{S_2} = f_2(\phi)$$

$$\sigma_{x S_2} = -x/r\, f_2'(\phi)$$

$$\sigma_{x S_1 (x=0)} \equiv \sigma_{x S_2 (x=h)}$$

$$h/r\, f_1'(\phi) = -h/r\, f_2'(\phi)$$

$$\therefore \quad f_1'(\phi) = -f_2'(\phi)$$

$$\int \quad f_1(\phi) = -f_2(\phi) + K'$$

$$\tau_{S_1} = -\tau_{S_2} + K'$$

$$\sigma_{x S_1} = (h/x - 1)\, \sigma_{x S_2}$$

FORM 1319A

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

		REPORT NO. 7/0510/9	
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AIRCRAFT:		PREPARED BY	DATE
		F. AHLHUETING	11/Oct/1955
		CHECKED BY	DATE
C-105	CENTRE SECTION		

ASSUME A GENERAL EXPRESSION IN SERIES FORM FOR $f_1(\phi)$ —

$$f_1(\phi) = \sum_{n=1}^{\infty}\left(\frac{\alpha_n}{n}\cdot\sin n\phi + \frac{\beta_n}{n}\cdot\cos n\phi\right)$$

$$f_1'(\phi) = \sum_{n=1}^{\infty}\left(\alpha_n\cdot\cos n\phi - \beta_n\cdot\sin n\phi\right)$$

WRITE: $\alpha_n = r/4\,\tau_{\alpha n}$; $\beta_n = r/4\cdot\tau_{\beta n}$; $K' = r/4\cdot K$.

<u>THEN</u> —

$$q_{s_1} = r/4\left\{\sum_{n=1}^{\infty}\left(\frac{\tau_{\alpha n}}{n}\cdot\sin n\phi + \frac{\tau_{\beta n}}{n}\cdot\cos n\phi\right) + K\right\} \text{— (5)}$$

$$\tau_{xs_1} = (1 - x/4)\sum_{n=1}^{\infty}\left(\tau_{\alpha n}\cdot\cos n\phi - \tau_{\beta n}\cdot\sin n\phi\right) \text{— (6)}$$

$$q_{s_2} = -r/4\left\{\sum_{n=1}^{\infty}\left(\frac{\tau_{\alpha n}}{n}\cdot\sin n\phi + \frac{\tau_{\beta n}}{n}\cdot\cos n\phi\right)\right\} \text{— (7)}$$

$$\tau_{xs_2} = x/4\sum_{n=1}^{\infty}\left(\tau_{\alpha n}\cdot\cos n\phi - \tau_{\beta n}\cdot\sin n\phi\right) \text{— (8)}$$

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AIRCRAFT: C-105	CENTRE SECTION	REPORT NO. 7/0510/9	SHEET NO. S 13-20
		PREPARED BY: E. BHUGSTINE	DATE: 11/OCT/55
		CHECKED BY:	DATE:

TOTAL SHEAR FLOW, $(\tau t)_R$, ACTING ON RING:—

$$(\tau t)_R = (\tau t)_{RT} + (\tau t)_{RB} - (\tau t)_{R1} \quad \text{(Ref. Pg. 15 \& 18).}$$

$$= R/4\left\{(4/R) + \sum_{n=1}^{\infty} b_n\cdot\cos(n\pi/2 + n\phi)\right\}(\tau_{4R}\cdot t)$$

$$-(R/4)\left\{\sum_{n=1}^{\infty} a_n\cdot\sin(n\pi/2 + n\phi)\right\}(\tau_{4R}\cdot t)$$

$$-R/4\left\{\sum_{n=1}^{\infty}\left(\frac{\tau_{A_n}\cdot t}{n}\cdot\sin n\phi + \frac{\tau_{B_n}\cdot t}{n}\cdot\cos n\phi\right) + Kt\right\}$$

COMBINING COS & SIN TERMS:—

$$(\tau t)_R = R/4\left[\sum\left\{-\frac{\tau_{A_n}\cdot t}{n} - a_n\cos\frac{n\pi}{2}\cdot(\tau_{4R}\cdot t) - b_n\cdot\sin\frac{n\pi}{2}(\tau_{4R}\cdot t)\right\}\sin n\phi\right.$$

$$+\left\{-\frac{\tau_{B_n}\cdot t}{n} - a_n\cdot\sin\frac{n\pi}{2}(\tau_{4R}\cdot t) + b_n\cdot\cos\frac{n\pi}{2}(\tau_{4R}\cdot t)\right\}\cos n\phi\Big\}$$

$$\left.+(4/R)(\tau_{4R}\cdot t) - (Kt)\right] \text{———} (9)$$

FORM 1319A

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TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0570/9		
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PREPARED BY	DATE	
E. AUGUSTINE	11/OCT./55	
CHECKED BY	DATE	

AIRCRAFT:
C-105

CENTRE SECTION

TOTAL SHEAR FLOW $(\tau t)_{f_1}$ ACTING ON f_1 —

$$(\tau t)_{f_1} = \frac{1}{4}\left\{ 2\sum_{n=1}^{\infty}\left(\frac{T_{\alpha n}}{n}\cdot \sin n\phi + \frac{T_{\beta n}}{n}\cdot \cos n\phi\right) + k_1 \right\} - (4)$$

FORM 1319A

EQUILIBRIUM CONDITIONS ON RING —

$$\sum \text{VERT. FORCES} \parallel \; r_0 \int_0^{2\pi} (r\tau)_n \cdot \sin\phi \cdot d\phi = 0$$

GIVING —

$$\sigma_{Z_1} = (a_1 \cdot \cos \pi/2)\,\sigma_{b_{418}} = 0$$

$$\sum \text{HORIZ. FORCES} \parallel \; -r_0 \int_0^{2\pi} (r\tau)_n \cdot \cos\phi \cdot d\phi = 0$$

GIVING —

$$\sigma_{R_1} = -(a_1 \cdot \sin \pi/2)\,\sigma_{b_{44}} = -\sigma_{b_{418}}$$

$$\sum \text{MOMENT} \parallel \; r_0^2 \int_0^{2\pi} (r\tau)_n \cdot d\phi = 0$$

GIVING —

$$M_1 = (t/r)\,\tau_{419}$$

THE ABOVE QUANTITIES — σ_{R_1} & M_1 DEFINE THE BENDING & TORSIONAL GROUPS IN THE BODY OF THE PROBLEM. IN THIS LOCAL PROBLEM, THEY ARE TO BE CONSTRUED AS APPLIED LOADS GIVING RISE TO HIGHER HARMONIC REDUNDANTS — WHILE IN

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THE BODY OF THE PROGRAM. THEY ARE AS QUADRANTS ACTIVATED BY THEIR APPLIED LOADS THEN DEFINED.

FORM 1319 A

A. V. ROE CANADA LIMITED
MALTON - ONTARIO
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REPORT NO. 7/0570/9
SHEET NO. 13-24

AIRCRAFT:		PREPARED BY	DATE
C-105	CENTRE SECTION	E. AUGUSTINE	12/OCT/55
		CHECKED BY	DATE

TOTAL STRAIN ENERGY (V) & APPLICATION OF LEAST WORK FOR SOLUTION OF nTH TERM SERIES COEFFICIENTS: τ_{A_n}, τ_{B_n}.

$V = V_R + V_{S_1} + V_{S_2} + V_{F_1}$

AN EXPRESSION FOR EACH OF THE ABOVE (V_i) IS FOUND & THE FUNCTIONS $\frac{\partial V_i}{\partial \tau_{A_n}}$ & $\frac{\partial V_i}{\partial \tau_{B_n}}$ FORMED. τ_{A_n} & τ_{B_n} ARE FOUND FROM THE SOLUTION OF THE TWO EQUATIONS:

$$\frac{\partial V}{\partial \tau_{A_n}} = 0$$

$$\frac{\partial V}{\partial \tau_{B_n}} = 0$$

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REPORT NO. 7/0570/9
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AIRCRAFT:		PREPARED BY	DATE
C-105	CENTRE SECTION	E. AUGUSTINE	13/OCT/55
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V_R - STRAIN ENERGY OF RING -

$$V_R = \frac{\pi R^5}{2EI_R} \sum \frac{\alpha_n^2+\beta_n^2}{n^2(n^2-1)^2} \qquad \text{(REF. APPENDIX).}$$

WHERE -

α_n - NTH HARMONIC OF SIN SERIES } SHEAR FLOW ON RING.
β_n - " " " COS " }

(H HAS NO EFFECT - PASSING THRU TO DUCT.)

& REF. PG. 20. -

$$\alpha_n = R/4 \left\{ -\frac{\tau_{\theta_n} t}{n} - a_n \cdot \cos\frac{n\pi}{2}(\tau_{\theta_{4,8}} t) - b_n \cdot \sin\frac{n\pi}{2}(\tau_{4,8} \cdot t) \right\}$$

$$\beta_n = R/4 \left\{ -\frac{\tau_{\theta_n} t}{n} - a_n \cdot \sin\frac{n\pi}{2}(\tau_{\theta_{4,8}} t) + b_n \cdot \cos\frac{n\pi}{2}(\tau_{4,8} \cdot t) \right\}$$

FORM $\dfrac{\partial V_R}{\partial \tau_{\alpha_n} t}$ & $\dfrac{\partial V_R}{\partial(\tau_{\beta_n} \cdot t)}$ —

$$\frac{\partial V_R}{\partial(\tau_{\alpha_n} t)} = \frac{\pi R^7}{4^2 EI_R} \frac{\left\{ \frac{\tau_{\alpha_n} t}{n} + a_n \cos\frac{n\pi}{2}(\tau_{\theta_{4,8}} t) + b_n \sin\frac{n\pi}{2}(\tau_{4,8} \cdot t) \right\}}{n^3(n^2-1)^2}$$

$$\frac{\partial V_R}{\partial(\tau_{\beta_n} t)} = \frac{\pi R^7}{4^2 EI_R} \frac{\left\{ \frac{\tau_{\beta_n} t}{n} + a_n \sin\frac{n\pi}{2}(\tau_{\theta_{4,8}} t) - b_n \cos\frac{n\pi}{2}(\tau_{4,8} \cdot t) \right\}}{n^3(n^2-1)^2}$$

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REPORT NO. 7/0510/9
SHEET NO. 1326

AIRCRAFT:		PREPARED BY	DATE
C-105	CENTRE SECTION	[illegible]	[illegible]
		CHECKED BY	DATE

V_{S_1}, V_{S_2} – STRAIN ENERGY OF DUCTS –

GENERAL –

$$V_S = \frac{Rt}{2G}\int_0^{2\pi}\int_0^{L}\left\{(G/E)\sigma_x^2 + \tau^2\right\}d\phi\, dx$$

WRITE $X = L - X'$ (REF. PG. 16)

THEN $(1 - X/L) = \frac{L - X}{L} = \frac{X'}{L}$

SUBSTITUTING $(1 - X/L)$ IN σ_{xS} (PG. 19), THE EXPRESSIONS FOR τ_S & σ_{xS} ARE SIMILAR FOR BOTH S_1 & S_2.

$\therefore$

$$V_{S_1} + V_{S_2} = \left(\frac{Rt}{G}\right)\int_0^{2\pi}\int_0^{L}\left\{(G/E)\,\sigma_{xS_2}^2 + \tau_{S_2}^2\right\}d\phi\cdot dx + \left(\frac{R^3t}{2GL^2}\right)\gamma_1^2\int_0^{2\pi}\int_0^{L} d\phi\, dx$$

($\because$ OF ORTHOGONALITY γ_1 DOES NOT COUPLE WITH HARMONICS).

$\therefore$

$$V_{S_1} + V_{S_2} = \left(\frac{\pi R^3}{GLt}\right)\left\{\sum_{n=1}^{\infty}\left(\frac{GL^2}{3ER^2} + \frac{1}{n^2}\right)\left((\sigma_{a_n} t)^2 + (\sigma_{b_n} t)^2\right) + (\gamma_1 t)^2\right\}$$

FORM $\frac{\partial V_{(S_1+S_2)}}{\partial(\sigma_{a_n} t)}$ & $\frac{\partial V_{(S_1+S_2)}}{\partial(\sigma_{b_n} t)}$ –

$$\frac{\partial V_{(S_1+S_2)}}{\partial(\sigma_{a_n} t)} = \left(\frac{2\pi R^3}{GLt}\right)\left(\frac{GL^2}{3ER^2} + \frac{1}{n^2}\right)(\sigma_{a_n} t)$$

$$\frac{\partial V_{(S_1+S_2)}}{\partial(\sigma_{b_n} t)} = \left(\frac{2\pi R^3}{GLt}\right)\left(\frac{GL^2}{3ER^2} + \frac{1}{n^2}\right)(\sigma_{b_n} t)$$

FORM 1319A

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

REPORT No. 7/0510/9
SHEET No. 13·27

AIRCRAFT:		PREPARED BY	DATE
C-105	CENTRE SECTION	B. AUGUSTINE	12/OCT/[illegible]
		CHECKED BY	DATE

V_{f_1} - STRAIN ENERGY OF FRAME AS A RING (REF. APPENDIX).

$$V_{f_1} = \frac{\pi R^5}{2EI_{f_1}} \sum_{n=1}^{\infty} \frac{(\alpha_n^2 + \beta_n^2)}{n^4(n^2-1)^2}$$ (REF. APPENDIX E, PG. 25).

WHERE —

$$\alpha_n = 2R/t \; \frac{\tau_{\alpha n}}{n}$$

$$\beta_n = 2R/t \; \frac{\tau_{\beta n}}{n}$$

$$\therefore \quad V_{f_1} = \left(\frac{2\pi R^7}{t^2 E I_{f_1}}\right) \sum_{n=1}^{\infty} \left\{ \frac{(\tau_{\alpha n} t)^2 + (\tau_{\beta n} t)^2}{n^4(n^2-1)^2} \right\}$$

FORM $\frac{\partial V_{f_1}}{\partial(\tau_{\alpha n} t)}$ & $\frac{\partial V_{f_1}}{\partial(\tau_{\beta n} t)}$ —

$$\frac{\partial V_{f_1}}{\partial(\tau_{\alpha n} t)} = \left(\frac{4\pi R^7}{t^2 E I_{f_1}}\right)\left(\frac{1}{n^4(n^2-1)^2}\right)(\tau_{\alpha n} t).$$

$$\frac{\partial V_{f_1}}{\partial(\tau_{\beta n} t)} = \left(\frac{4\pi R^7}{t^2 E I_{f_1}}\right)\left(\frac{1}{n^4(n^2-1)^2}\right)(\tau_{\beta n} t).$$

FORM 1318A

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TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0570/9
SHEET No. 13·28.

AIRCRAFT:		PREPARED BY	DATE
C-105	CENTRE SECTION	E. ANGHETINE	12/OCT./55
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FORM $\frac{\partial V}{\partial(\bar\tau_{a_n}\cdot t)} = 0$ & $\frac{\partial V}{\partial(\bar\tau_{b_n}\cdot t)} = 0$ (REF. PGS. 25, 26, 27).

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

$$\frac{R^4}{4I_n}\left(\frac{\frac{\bar\tau_{a_n}\cdot t}{n} + a_n \cos\frac{n\pi}{2}(\bar\tau_{b_{418}}\cdot t) + b_n \sin\frac{n\pi}{2}(\bar\tau_{a_{418}}\cdot t)}{n^3(n^2-1)^2}\right)$$

$$+\ 2\left(\frac{E}{Gt}\right)\left(\frac{L^2}{3R^2}\left(\frac{G}{E}\right) + \frac{1}{n^2}\right)(\bar\tau_{a_n}\cdot t)$$

$$+\ \frac{4R^2}{4I_{f.}}\left(\frac{1}{n^4(n^2-1)^2}\right)(\bar\tau_{a_n}\cdot t) = 0$$

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

$$\frac{R^4}{4I_n}\left(\frac{\frac{\bar\tau_{b_n}\cdot t}{n} + a_n\cdot \sin\frac{n\pi}{2}(\bar\tau_{b_{418}}\cdot t) - b_n\cdot \cos\frac{n\pi}{2}(\bar\tau_{a_{418}}\cdot t)}{n^3(n^2-1)^2}\right)$$

$$+\ 2\left(\frac{E}{Gt}\right)\left(\frac{L^2}{3R^2}\left(\frac{G}{E}\right) + \frac{1}{n^2}\right)(\bar\tau_{b_n}\cdot t)$$

$$+\ \frac{4R^4}{4I_{f.}}\left(\frac{1}{n^4(n^2-1)^2}\right)(\bar\tau_{b_n}\cdot t) = 0$$

FORM 1319 A

AIRCRAFT:		PREPARED BY	DATE
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COLLECTING TERMS—

$$\left\{\frac{\left(\frac{R^4}{4}\right)\left(\frac{1}{I_n}+\frac{4}{I_f}\right)}{n^4(n^2-1)^2}+2(E/Gt)\left(\frac{d^2}{3R^2}(G/E)+\frac{1}{n^2}\right)\right\}(\tau_{A_n}\cdot t)$$

$$=\frac{\left(\frac{R^4}{4I_n}\right)}{n^3(n^2-1)^2}\left\{-a_n\cdot\cos\frac{n\pi}{4}(\tau_{a_{4,18}}\cdot t)-b_n\cdot\sin\frac{n\pi}{2}(\tau_{4,19}\cdot t)\right\}$$

$$\left\{\frac{\left(\frac{R^4}{4}\right)\left(\frac{1}{I_n}+\frac{4}{I_f}\right)}{n^4(n^2-1)^2}+2(E/Gt)\left(\frac{d^2}{3R^2}(G/E)+\frac{1}{n^2}\right)\right\}(\tau_{B_n}\cdot t)$$

$$=\frac{\left(\frac{R^4}{4I_n}\right)}{n^3(n^2-1)^2}\left\{-a_n\cdot\sin\frac{n\pi}{4}(\tau_{a_{4,18}}\cdot t)+b_n\cdot\cos\frac{n\pi}{2}(\tau_{4,19}\cdot t)\right\}$$

DEFORMATION OF f_1 UNDER BENDING $(\tau_{a_{4,18}})$ IS ASSUMED NON-EXISTENT—

∴ THE SOLUTION FOR $(\tau_{A_n}\cdot t)$ & $(\tau_{B_n}\cdot t)$ MAY BE WRITTEN—

$$(\tau_{A_n}\cdot t)=-A_{1n}\cdot a_n\cdot\cos\frac{n\pi}{4}(\tau_{a_{4,18}}\cdot t)-A_{2n}\cdot b_n\cdot\sin\frac{n\pi}{2}(\tau_{4,19}\cdot t)$$

$$(\tau_{B_n}\cdot t)=-A_{1n}\cdot a_n\cdot\sin\frac{n\pi}{2}(\tau_{a_{4,18}}\cdot t)+A_{2n}\cdot b_n\cdot\cos\frac{n\pi}{2}(\tau_{4,19}\cdot t)$$

WHERE FOR THE CALC. OF A_{1n} ; $I_{f_1}=\infty$.

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TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C-105	CENTRE SECTION	REPORT No. 7/0510/9	SHEET No. 13-30
		PREPARED BY F. AUGUSTINE	DATE 12/OCT./55
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CALCULATION of τ_{s_1} & σ_{xs_1} —

SUBSTITUTE σ_{a_n} & σ_{b_n} IN τ_{s_1} & σ_{xs_1} — PG. 17.

$$\tau_{s_1} = r/L \sum_{\substack{a_n : n=1 \\ b_n : n=2}}^{\infty} \left\{ \frac{A_{1n}\cdot a_n}{n} \sigma_{b_{4,18}} \cdot \sin n(\phi+\pi/2) + \frac{A_{2n}\cdot b_n}{n} \tau_{4,19} \cdot \cos n(\phi+\pi/2) \right\} + \tau_{4,19}$$

$$\sigma_{xs_1} = (1-x/L) \sum_{\substack{a_n : n=1 \\ b_n : n=2}}^{\infty} \left\{ -A_{1n}\cdot a_n \cdot \sigma_{b_{4,18}} \cos n(\phi+\pi/2) - A_{2n}\cdot b_n \cdot \tau_{4,19} \cdot \sin n(\phi+\pi/2) \right\}$$

WRITE —

$A_{1n}\cdot a_n = \sigma_{n_b}$ (b - BENDING)

$A_{2n}\cdot b_n = \tau_{n_T}$ (T - TORSION)

THEN —

$$\frac{\tau_{s_1}}{\sigma_{b_{4,18}}} = (r/L) \sum_{n=1}^{\infty} \frac{\sigma_{n_b}}{n} \cdot \sin n(\phi+\pi/2) \; ; \quad \frac{\tau_{s_1}}{\tau_{4,19}} = (r/L) \sum_{n=2}^{\infty} \frac{\tau_{n_T}}{n} \cdot \cos n(\phi+\pi/2) + 1$$

$$\frac{\sigma_{xs_1}}{\sigma_{b_{4,18}}} = -(1-x/L) \sum_{n=1}^{\infty} \sigma_{n_b} \cdot \cos n(\phi+\pi/2) \; ; \quad \frac{\sigma_{xs_1}}{\tau_{4,19}} = -(1-x/L) \sum_{n=2}^{\infty} \tau_{n_T} \sin n(\phi+\pi/2)$$

FORM 1319A

A. V. ROE CANADA LIMITED MALTON - ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7/0510/9 SHEET NO. B-31	
AIRCRAFT: C-105	CENTRE SECTION	PREPARED BY E. AUGUSTINE	DATE 13/OCT/55
		CHECKED BY	DATE

CALCULATION OF ENERGY COEFFICIENTS FOR INCLUSION IN C.I. MATRIX.

CALCULATION IS DONE IN TWO STAGES –

1. CONSIDERING TERMS IN σ_{R418}^2 & τ_{419}^2 ONLY
2. CONSIDERING TERMS IN σ_{R418} & τ_{419} THAT COUPLE WITH EXISTING STRESSES (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 σ_{An} & σ_{Bn} IN V_R, V_{S1}, V_{S2}, V_G & EXPRESS IN TERMS OF $\sigma_{nL} = k_{1n} a_n$ & $\sigma_{nT} = k_{2n} b_n$ (REF. PG. 3)

V_R – STRAIN ENERGY OF RING – (REF. PG. 25, 29).

$$\frac{2V_R}{\sigma_{R418}^2} = \frac{\pi R^7 L^2}{4^2 A I_R} \sum_{n=1}^{\infty} \frac{\left(\frac{\sigma_{nL}}{n} - a_n\right)^2}{n^2(n^2-1)^2}$$

$$\frac{2V_R}{(\tau_{419} \cdot L)^2} = \frac{\pi R^7}{4^2 A I_R} \sum_{n=2}^{\infty} \frac{\left(\frac{\sigma_{nT}}{n} - b_n\right)^2}{n^2(n^2-1)^2}$$

FORM 1710A

REPORT NO. 7/0570/9
SHEET NO. 13-32

AIRCRAFT:		PREPARED BY	DATE
C-105	CENTRE SECTION	E. AUGUSTINE	12/OCT./55
		CHECKED BY	DATE

$V_{s_1}+V_{s_2}$ – STRAIN ENERGY OF DUCT – (REF. PGS. 26, 29)

✓ ✓

$$2\frac{(V_{s_1}+V_{s_2})}{\sigma_{\theta 418}^{2}} = \frac{2\pi R^3 \cdot t}{Gh}\sum_{n=1}^{\infty}\left(\frac{Gh^3 n^2}{3ER^2}+1\right)\left(\frac{\sigma_{n\theta}}{n}\right)^2$$

$$\frac{2(V_{s_1}+V_{s_2})}{(\tau_{418}t)^2} = \frac{2\pi R^3}{Ght}\sum_{n=2}^{\infty}\left(\frac{Gh^2}{3ER^2}+\frac{1}{n^2}\right)\sigma_{n\tau}^2 + 2\frac{\pi R h}{Gt}$$

V_f – STRAIN ENERGY OF FRAME – (REF. PGS. 27, 29)

$$\frac{2V_f}{(\tau_{418}t)^2} = \frac{4\pi R^7}{h^2 E I_f}\sum_{n=2}^{\infty}\frac{\sigma_{n\tau}^2}{n^2(n^2-1)^2}$$

FORM 1319A

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

AIRCRAFT: C-105	CENTRE SECTION	REPORT NO. 7/0510/9 SHEET NO. 13·38
		PREPARED BY: G. AUGUSTINE — DATE: 12/OCT/55
		CHECKED BY: — DATE:

SUBSTITUTE NUMERICAL VALUES IN THE EXPRESSIONS FOR V_n, V_{S_1, S_2}, & V_{F_1} —

$b = 18.00$ in.

$h = 16.25$ in.

$E = (10.6)10^3$ kips / in.^{2}

$t = .032$

$G = 4 \times 10^3$ kips/in^2

$I_n = .2018$ in.^{4} — REF.

$I_{F_1} = 1.22$ in.^{4} — APPENDIX.

FORM 1312A

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

REPORT No. 7/0510/9
SHEET No. 13·32

AIRCRAFT:		PREPARED BY	DATE
C-105	CENTRE SECTION	CHECKED BY	DATE

NUMERICAL CALCULATIONS - CONT'D. —

(i) BENDING —

$$A_{1n} = \frac{n(P/4)}{.00058606631\,n^4(n^2-1)^2 + .00571679\,n^2(n^2-1)^2 + 1.10769231}$$

$$\sigma_{nb} = A_{1n} \cdot a_n$$

$$\frac{\sigma_{nb}}{n} = \frac{\sigma_a(P/4)\sin\left(\frac{n\pi}{4}\right)}{.00058606639\,n^4(n^2-1)^2 + .00571674\,n^2(n^2-1)^2 + 1.10769231}$$

THEN —

$$\frac{2V_2}{\sigma_{a_{4\cdot18}}^{2}} = 3495.441270\left\{\sum_{n=1}^{\infty}\frac{\left(\frac{\sigma_{nb}}{n} - \sigma_a \cdot \sin\frac{n\pi}{4}\right)^2}{n^2(n^2-1)^2}\right\}(10)^{-3}$$

$$\frac{2(V_{s_1}+V_{s_2})}{\sigma_{a_{4\cdot18}}^{2}} = \left\{\sum_{n=1}^{\infty}(1.849394690\,n^2 + 18.0399378)\left(\frac{\sigma_{nb}}{n}\right)^2\right\}(10)^{-3}$$

FORM 1319A

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

REPORT NO. 7/0510/9
SHEET NO. 13·35

AIRCRAFT: C-105
CENTRE SECTION

PREPARED BY: E. AUGUSTINE
DATE:
CHECKED BY:
DATE:

①	②	③	④	⑤	⑥	⑦	⑧	⑨	⑩
n	$n^2(n^2-1)^2$	$n^4(n^2-1)^2$	$\frac{f_{1,n}}{n(n/4)}$	$\sin\frac{n\pi}{4}$	$\frac{\sqrt{2}/4 \times ⑤/④}{\xi_{n}/n}$	$(1.84939469n^2 + 18.03993718)(10)^{-3}$	⑦×⑥²	$(\sqrt{2}/4 ⑤ - ⑥)^2$	8.495443 × ⑨/2
1	0	0	1.10761231	$1/\sqrt{2}$	1.0	.01988933	.01988933	0	0
2	36	144	1.39782031	1	.12061615	.02543752	.03194319	.08619168	.00836881
3	576	5184	7.43879952	$1/\sqrt{2}$	.14890876	.03468441	.00076909	.72435830	.00439573
4	3600	57600	—	0	0	—	0	0	0
5	14400	360,000.	294.41336877	$\overline{1/\sqrt{2}}$	$\overline{.00376237}$	.06427480	.00000091	.99248242	.00024092
6	44100	1,587,600	1183.65213207	$\overline{1}$	$\overline{.00132344}$	.08461819	.00000015	.99622017	.00015822
7	112896	5,531,904.	3888.57342326	$\overline{1}/\sqrt{2}$	$\overline{.00028486}$	.10866027	.00000001	.99943035	.00003094
8	254016	16,257,024	—	0	0	—	0	0	0
9	518400	41,990,400.	27593.25377097	$1/\sqrt{2}$	.00004017	.16784090	0	.99991966	.00000674
10	980100	98,010,000.	63044.50045591	1	.00002485	.20397140	0	.99989135	.00000714
11	1,742,400	210,830,400.	133,522.65601857	$1/\sqrt{2}$	.00000830	.24181669	0	.99998340	.00000201
12	2,944,656	424,030,464	—	0	0	—	0	0	0
13	4,769,856	806,105,664	491,900.30923358	$1/\sqrt{2}$	$\overline{.00000222}$	.33058763	0	.99999536	.00000093

ORM 1319A

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

REPORT NO. 7/0510/9
SHEET NO. 13 36

AIRCRAFT:		PREPARED BY	DATE
C-105	CENTRE SECTION	E. AUGUSTINE	13. OCT/55
		CHECKED BY	DATE

NUMERICAL CALCULATIONS — CONT'D. —

(ii) TORSION

$$k_{2n} = \frac{1{,}706.211920\,(r/4)}{\left(9.754480 + n^2 + \frac{3137.329058}{n^2(n^2-1)^2}\right) n(n^2-1)^2} = \frac{1{,}706.211920\,(r/4)}{A_n}$$

$$\sigma_{nT} = k_{2n} \cdot b_n$$

$$\sigma_{nT} = \frac{1{,}706.21192 \cdot (r/4) \cdot b_n}{\left(9.754480 + n^2 + \frac{3137.329058}{n^2(n^2-1)^2}\right) n(n^2-1)^2}$$

$$(r/4)\,b_n = 2\left(\cos\frac{n\pi}{4} - n\sin\frac{n\pi}{2}\right)$$

THEN —

$$\frac{2V_n}{(\tau_{4,9}\cdot t)^2} = 2{,}782{,}043.00 \left\{ \sum_{n=2}^{\infty} \frac{\left\{\frac{R}{4}\left(\frac{\sigma_{nT}}{n}\right) - \left(\frac{r}{4}b_n\right)\right\}^2}{n^2(n^2-1)^2} \right\} (10)^{-3}$$

$$\frac{2(V_c + V_{s_n})}{(\tau_{4,9}\cdot t)^2} = 17{,}617.89538 \left\{ \sum_{n=2}^{\infty} \left(.102517 + \frac{1}{n^2}\right)(\sigma_{n_T})^2 \right\} (10)^{-3}$$

$$\frac{2V_f}{(\tau_{4,9}\cdot t)^2} = 2{,}252{,}921.0 \left\{ \sum_{n=2}^{\infty} \frac{\sigma_{nT}^2}{n^4(n^2-1)^2} \right\} (10)^{-3}$$

FORM 1319A

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

REPORT NO. 7/0510/9
SHEET NO. 13·37.

PREPARED BY	DATE
E. AUGUSTINE	13/OCT/55
CHECKED BY	DATE

AIRCRAFT: C-105

CENTRE SECTION

①	②	③	④	⑤	⑥	⑦	⑧	⑨
				(REF. PG. 36)	$\frac{1,926.3113 \times 10^5}{⑤}$	$\frac{\pi}{4} \cdot b_n$	$= ⑥ \times ⑦$	$= ⑧ / ①$
n	n^2	$n(n^2-1)^2$	$n^2(n^2-1)^2$	$A_n/10^6$	$\frac{b_{2n}}{\pi/4}$	$2(\cos\frac{n\pi}{2} - n\sin\frac{n\pi}{4})$	∇_{nT}	$\frac{\nabla_{nT}}{n}$
2	4	18	36	.00181625	939,415.	$\bar{4}$	$\bar{3}$ .757660	$\bar{1}$ .878830
3	9	192	576	.00464664	367,193.	$\bar{3}$ .6368	$\bar{2}$ .077137	— .692372
4	16	900	3600	.02396336	71,200.9	$\bar{2}$	— .142402	— .03560
5	25	2880	14400	.10072037	16,990.1	5 .6568	.095327	.019166
6	36	7350	44100	.33681832	5,065.67	12	.060788	.010132
7	49	16128	112896	.94804094	1,799.72	11 .8136	.020361	.002909
8	64	31752	254016	2.34224441	728.452	2	.001457	.000182
9	81	57600	518400	5.22780664	326.392	$\bar{11}$ .3136	— .003692	— .000411
10	100	98010	980100	10.75735031	158.609	$\bar{20}$	— .003172	— .0003172
11	121	158400	1,742,400	20.71179484	82.3788	$\bar{16}$ .9704	— .001398	— .000127
12	144	245388	2,944,656	37.72896578	45.2219	$\bar{2}$	— .000090	— .0000075

FORM 1319A

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

REPORT No. 7/0510/9
SHEET No. 13-88

PREPARED BY	DATE
E. AUGUSTINE	13/OCT./55.
CHECKED BY	DATE

AIRCRAFT. C-105

CENTRE SECTION

⑩ $= 1.107692 \times ⑨$ $\left(\frac{r}{H}\right)\left(\frac{J_{nI}}{n}\right)$	⑪ $= (⑩ - ⑦)^2$ $\left\{\left(\frac{r}{H}\right)\left(\frac{J_{nI}}{n}\right) - \left(\frac{r}{H}\right)A_n\right\}^2$	⑫ $\frac{2{,}782{,}043}{④}$ $\frac{2{,}782{,}043}{n^2(n^2-1)^2}$	⑬ $= ⑪ \times ⑫$ $\cdot \frac{2V_n}{(\tau_{4.19}\cdot t)^2}$	⑭ $17.61781538 \times (.1025174 + 1/n^2)$	⑮ $= ⑧^2 \times ⑭$ $\frac{2V_{s1+2}}{(\tau_{4.10}\cdot t)^2}$	⑯ $\frac{4{,}252{,}921}{③ \times ③}$ $\frac{4{,}252{,}921}{n^3(n^2-1)^2}$	$= ⑧^2 \times ⑯$ $\frac{2V_{e1}}{(\tau_{4.19}\cdot t)^2}$
$\bar{2}$ .081165	3 .681928	77.279	289.536	6 .21060763	87.6938	15 .6453	220.912
— .766735	23 .910780	4.82994	115.488	3 .76367575	16.2384	.434587	1.87503
— .039435	3 .843813	.772790	2.970	2 .90725249	.059	.0439099	.00090
.021230	31 .759655	.193197	6.136	2 .51084960	.023	.0062584	.00006
.011223	143 .730774	.0630849	9.067	2 .29552368	.00848	.0004091	.000005
.003222	127 .974634	.0246425	3.152	2 .16567779	.0009	.00040738	—
.000202	3 .999190	.0109523	.044	2 .08141336	.000004	.00013862	—
— .000455	127 .987261	.0053666	.687	2 .02366194	.00003	.00005363	—
— .000351	399 .985960	.00383853	1.135	1 .18231273	—	.00002439	—
— .000141	289 .989699	.00159667	.460	1 .95177806	—	.00001686	—
— .0000083	3 .979965	.000949777	.004	1 .92847246	—	.00000583	—
		$2\sum_{n=3}^{14}$	$139.143(\tau_{4.19}\cdot t)^2$		$16.330(\tau_{4.10}\cdot t)^2$		$1.876(\tau_{4.19}\cdot t)^2$

FORM 1319A

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

REPORT NO. 7/0510/7
SHEET NO. 134

AIRCRAFT:		PREPARED BY	DATE
C-105	CENTRE SECTION	F. AUGUSTINE	13/OCT./55
		CHECKED BY	DATE

7. STRAIN ENERGY COEFFICIENTS OF COUPLING TERMS & OF EXISTING STRESSES.

EXISTING STRESSES — (REF. PG. 12).

$\sigma_{b_{406}}$ - BENDING

σ_{W417}, σ_{W405} — WARPING

- COUPLING COEFFICIENTS OF σ_{418} & σ_{419} WITH THE ABOVE (σ_{b406}, σ_{W417}, σ_{W405}) &
- COEFFICIENTS OF σ_{b406}^2, σ_{W417}^2, σ_{W405}^2 WILL BE CALCULATED.

$\underline{V_n}$ - $\underline{\text{RING}}$. -

CONSIDER SHEAR FROM TERMS -

$$\sigma_{\tau_n} = (n/4)\left\{\left(\frac{f_{22}}{4}-f\right) b_2 \cdot (\sigma_{417}\, t)\cos 2\phi + \left(\frac{\sigma_{W417}\, t}{4}\right)\cos 2\phi\right.$$

$\underline{\text{THEN}}$ - WRITING $f_{22} = \dfrac{\sigma_{2T}}{f_2}$ &

& SUBSTITUTING IN FORMULA - PG. 25

$$2V_n = \frac{\pi R^7 t^2}{36 L^2 E I_n}\left\{\left(\frac{\sigma_{2T}}{4} - b_2\right)(\sigma_{b406})(\sigma_{W417}) + \frac{1}{4}(\sigma_{W417})^2\right\}$$

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

REPORT NO 7/0510/9	SHEET NO 13A0	
AIRCRAFT: C-105	CENTRE SECTION	
PREPARED BY E. AUGUSTINE	DATE 14/OCT./55	
CHECKED BY	DATE	

COUPLING - CONT'D.

$V_{S_1} + V_{S_2}$ - DUCTS. -

CONSIDER TERMS —

$$(\tau t)_{S_1} = -\frac{r}{2A}\left\{ b_2 \cdot k_{22}(\sigma_{419} \cdot t) + (\sigma_{W_{417}} \cdot t)\right\} \cos 2\phi$$

$$\sigma_{x_{S_1}} = (1 - x/A)\left\{ b_2 \cdot k_{22}(\sigma_{419} \cdot t) + \sigma_{W_{417}}\right\} \cdot \sin 2\phi$$

THEN -

$$2V_{S_1} = \left(\frac{\pi r^3}{GAt}\right)\left(\frac{GA^2}{3EI^2} + \frac{1}{4}\right)\left\{2b_2 \cdot k_{22}(\sigma_{419} t)(\sigma_{W_{417}} t) + (\sigma_{W_{417}} t)\right\}$$

& —

$$(\tau t)_{S_2} = \frac{r}{2A}\left\{ b_2 \cdot k_{22}(\sigma_{419} t) + (\sigma_{W_{417}} t - \sigma_{W_{405}} t)\right\} \cos 2\phi$$
$$+ \frac{r}{A}\left\{ -(\sigma_{b_{406}} t) + a_1 k_{11}(\sigma_{b_{418}} t)\right\} \cos \phi$$

$$\sigma_{x_{S_2}} = \left[\frac{x}{A}\left\{\sigma_{W_{417}} - \sigma_{W_{405}} + b_2 \cdot k_{22}\, \sigma_{419}\right\} + \sigma_{W_{405}}\right] \sin 2\phi$$
$$+ \left[\frac{x}{A}\left\{-\sigma_{b_{406}} + a_1 k_{11} \sigma_{b_{418}}\right\} + \sigma_{b_{406}}\right] \sin \phi$$

THEN - WRITING $a_1 \cdot k_{11} = \sigma_{1b}$

$b_2 \cdot k_{22} = \sigma_{2T}$

& SUBSTITUTING IN FORMULAE - PGE. 26 —

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

REPORT NO. 7/0510/9
SHEET NO. 13-41

AIRCRAFT:		PREPARED BY	DATE
C-105	CENTRE SECTION	E. AUGUSTINE	14/OCT/55
		CHECKED BY	DATE

COUPLING - CONT'D. -

$$2V_{3_1} = \left(\frac{\pi r^3 \cdot t}{Gh}\right)\left(\frac{Gh^2}{3Er^2} + \frac{1}{4}\right)\left\{\left(\sigma_{W_{417}}\right)^2 + 2\sigma_{2T}\left(\sigma_{W_{417}}\right)\left(\tau_{419}\right)\right\}$$

$$2V_{3_2} = \left(\frac{\pi r^3 t}{Gh}\right)\Bigg[\left(\frac{Gh^2}{3Er^2} + \frac{1}{4}\right)\left(\sigma_{W_{417}}^2 + \sigma_{W_{405}}^2\right)$$

$$+ \left(\frac{Gh^2}{3Er^2} + 1\right)\left(\sigma_{2_{406}}^2\right)$$

$$+ \left(\frac{Gh^2}{3Er^2} - \frac{1}{2}\right)\left(\sigma_{W_{417}}\sigma_{W_{405}}\right)$$

$$+ \left(\frac{2Gh^2}{3Er^2} + \frac{1}{2}\right)\sigma_{2T}\left(\sigma_{W_{417}} \cdot \tau_{419}\right)$$

$$+ \left(\frac{Gh^2}{3Er^2} - \frac{1}{2}\right)\sigma_{2T}\left(\sigma_{W_{405}} \cdot \tau_{419}\right)$$

$$+ \left(\frac{Gh^2}{3Er^2} - 2\right)\sigma_{1b}\left(\sigma_{B_{418}} \cdot \sigma_{B_{406}}\right)$$

FORM 1318A

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

REPORT NO. 7/0570/9
SHEET NO 13.412

AIRCRAFT: C-105	CENTRE SECTION	PREPARED BY E. AUGUSTINE	DATE 14/OCT./55
		CHECKED BY	DATE

COUPLING - CONT'D -

V_{f_1} - FRAME f_1

CONSIDER SHEAR FLOW TERMS --

$$\tau_{f_1} = -\frac{D}{4}\left(\tau_{H_{417}} - \frac{1}{4}\tau_{H_{405}} + b_2 \cdot k_{22}\,\tau_{417}\right)$$

THEN - WRITING $b_2 \cdot k_{22} = \tau_{2T}$

& SUBSTITUTING IN FORMULA - PG. 27 -

$$2V_{f_1} = \frac{\pi D^2 t^2}{364\, h^2 A I_{f_1}}\Big\{(\tau_{H_{417}})^2 + \frac{1}{4}(\tau_{H_{405}})^2 - (\tau_{H_{417}} \cdot \tau_{H_{405}})$$

$$+ 2\tau_{2T} \cdot (\tau_{H_{417}} \cdot \tau_{417})$$

$$- \tau_{2T}(\tau_{H_{405}} \cdot \tau_{417})\Big\}$$

FORM 1318A

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

REPORT No. 7/0510/9
SHEET No. R43

AIRCRAFT: C-105
CENTRE SECTION

PREPARED BY	DATE
E. AUGUSTINE	19/OCT
CHECKED BY	**DATE**

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

(REF. PG. 33)

(i) BENDING — 1ST HARMONIC ($n=1$) ONLY.

$$2V_{q_2} = -2 \times .00855762\, T_{a_{418}} \cdot T_{a_{406}} + .00894467\, T_{a_{406}}^{2}$$

(ii) TORSION —

$$2V_{2} = 2 \times .08409830\, T_{411} \cdot T_{w_{417}} + .02427386\, T_{w_{417}}^{2}$$

$$2V_{s_1} = -2 \times .01194876\, T_{419} \cdot T_{w_{417}} + .00317183\, T_{w_{417}}^{2}$$

$$2V_{s_2} = -2 \times .01194876\, T_{419} \cdot T_{w_{417}} + 2 \times .00673646\, T_{419} \cdot T_{w_{405}}$$
$$+ .00317783\, T_{w_{417}}^{2} + .00317783\, T_{w_{405}}^{2}$$
$$- 2 \times .00179272\, T_{w_{417}} \cdot T_{w_{405}}$$

$$2V_{f_1} = -2 \times .06020079\, T_{411} \cdot T_{w_{417}} + 2 \times .03010039\, T_{411} \cdot T_{w_{405}}$$
$$+ .01602077\, T_{w_{417}}^{2} + .00400517\, T_{w_{405}}^{2}$$
$$- 2 \times .00801038\, T_{w_{417}} \cdot T_{w}$$

ORM 1319A

REPORT NO.	7/0510/9
SHEET NO.	13.44
PREPARED BY	E. AUGUSTINE
DATE	17/OCT/55
CHECKED BY	
DATE	
AIRCRAFT:	C-105
	CENTRE SECTION

PREPARATION OF THE FOLLOWING RESULTS FOR INTRODUCTION INTO PROBLEM #4.

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

(i). BENDING (A18) - REF. PGS. 35, 43.

	$n \geq 2$	$n = 1$		
	$\overline{T}_{A18}^{2}$	$\overline{T}_{A18}^{2}$	$\overline{T}_{A06}^{2}$	$\overline{T}_{A18} \cdot \overline{T}_{A06}$
$2V_n =$	$.01321124$	0	0	0
$2V_{s_1} =$	$.03271410$	$.01981953$	0	0
$2V_{s_2} =$			$.00994467$	$\overline{2} \times .00855762$
$2V_{t_1} =$	0	0	0	0

$$2V_{TOTAL} = .06581465\,\overline{T}_{A18}^{2} - 2 \times .00855762\,\overline{T}_{A18} \cdot \overline{T}_{A06} + .00994467\,\overline{T}_{A06}^{2}$$

FORM 1319A

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

REPORT NO. 7/0510/9
SHEET NO. 13·45

AIRCRAFT:		PREPARED BY	DATE
C-105	CENTRE SECTION	E. AUGUSTINE	14/OCT/55
		CHECKED BY	DATE

(ii) TORSION (AIR). – REF. PG'S. 38, 43.

	$n \geq 3$	$n = 2$					
	τ_{419}^2	τ_{419}^2	σ_{W417}^2	σ_{W405}^2	$\tau_{419}\cdot\sigma_{W417}$	$\tau_{419}\cdot\sigma_{W405}$	$\sigma_{W417}\cdot\sigma_{W405}$
$2V_R=$	·1424824	·29136222	·02427886	0	2×·0840983	0	0
$2V_{S_1}=$	·0083606	·04489950	·00317983	0	$\bar{2}$×·01194876	0	0
$2V_{S_2}=$	·0083606	·04489950	·00317983	·00317983	$\bar{2}$×·01194896	2×·00673646	$\bar{2}$×·00179272
$2V_{P_1}$	·0019210	$\overset{*}{·}$22631473	$\overset{*}{·}$01602077	$\overset{*}{·}$00400519	$\overset{*}{\bar{2}}$×·06020079	$\overset{*}{2}$×·03010039	$\overset{*}{\bar{2}}$×·00301038

ADD TO THE ABOVE THE STRAIN ENERGY IN S, DUE TO THE CONSTANT TORSIONAL SHEAR FLOW

$$= \frac{2\pi R A t}{G} \quad \text{(REF. PG. 32)}$$

$$= \cdot 01470266\,(\tau_{419})^2$$

FORM 1319A

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

REPORT NO. 7/0510/9
SHEET NO. 13-46.

PREPARED BY	DATE
R. AUGUSTINE	14/OCT/55.
CHECKED BY	DATE

AIRCRAFT: C-105

CENTRE SECTION

(ii) <u>TORSION</u> –

THE FUNCTION $2V_F$, (REF. (ii) TORSION - PG. 43) PARTIALLY ANTICIPATES THE RESULTS OF PROBLEM #A SOLUTION. THIS, BY VIRTUE OF THE FACT THAT, IN GENERAL, THE FRAME, ENERGY IS EXPRESSED IN THE C_{iR} MATRIX AS A FUNCTION OF ONE STRESS ONLY (τ_{424} – THE AVERAGE SHEAR STRESS OF THE FRAME AS AN IDEALIZED TRAPEZOIDAL PANEL). THIS STRESS (424) IN TURN IS EXPRESSED, THRU THE MEDIUM OF THE H_{iP} MATRIX, AS A FUNCTION OF THE WARPING GROUPS $\textcircled{406}$ & (119). AND SINCE $\left\{\begin{array}{l} \tau_{W417} = W_{\textcircled{406}} \\ \tau_{W405} = W_{(119)} \end{array}\right.$ (REF. PG. 12)$\Big\}$, THE TERMS IN $2V_F$, IN $(\tau_{W417})^2$, $(\tau_{W405})^2$ & $(\tau_{W417} \cdot \tau_{W405})$ (REF. PG. 43) WILL APPEAR IN THE NORMAL SOLUTION OF PROBLEM #A.

FORM 1319A

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

REPORT NO. 7/0510/9
SHEET NO. 13·48

AIRCRAFT:		PREPARED BY	DATE
C-105	CENTRE SECTION	F. AUGUSTINE	14/OCT./55
		CHECKED BY	DATE

$2V_{f_1}$ IS ACCORDINGLY CONVERTED BY WRITING—

$$\sigma_W = \left(\sigma_{W_{417}} - \tfrac{1}{2}\sigma_{W_{425}}\right)$$

$$\therefore\ 2V_{f_1} = .01602077\,\sigma_W^2 - 2\times .06020079\,\tau_{419}\cdot\sigma_W$$

THEN—

— WRITE: $\sigma_W = -.827364528\,\tau_{424}$ (REF.

— CHANGE 419 TO 434 — A NEW STRESS POINT ON THE FRAME TO REPRESENT FRAME STRESSES DUE TO THE 2ND TORSION HARMONIC

& — ADD THE TERM IN $\underset{n=2}{\tau^2_{434}}$ (REF. SHEET 45).

THEN—

$$\underset{n=2}{2V_{f_1}} = .0071674\,\tau^2_{f_{424}} + 2\times .01970778\,\tau_{f_{424}}\cdot\tau_{f_{434}} + .22621473\,\tau^2_{f_{434}}$$

FOR THE REMAINING TERMS—

SUM $2V_{S_1}$, $2V_{S_2}$, $2V_R$ FOR $n \geq 2$ + $2V_{f_1}$ FOR $n \geq 3$

& INTRODUCE THE RESULTS INTO THE

C.L. ACCORDING TO EXISTING STRESS POINTS.

(SEE FOLLOWING PG. FOR TABULATION).

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

REPORT NO. 7/0510/9
SHEET NO. 13-48.

PREPARED BY	DATE
E. AUGUSTINE	17/OCT/55
CHECKED BY	DATE

AIRCRAFT:
C-105

CENTRE SECTION

	405	406	417	418	419	424	434
405	.00317983		$\overline{.00}199287$		.00673646		
406		.00199467		$\overline{.00}853762$			
417	$\overline{.00}199287$		.03068859		$\overline{.06}020078$		
418		$\overline{.00}853762$		.00581447			
419	.00673646		$\overline{.06}020078$		.55628778		
424						.0071168377	.019707789
434						.019707784	.32631473

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.

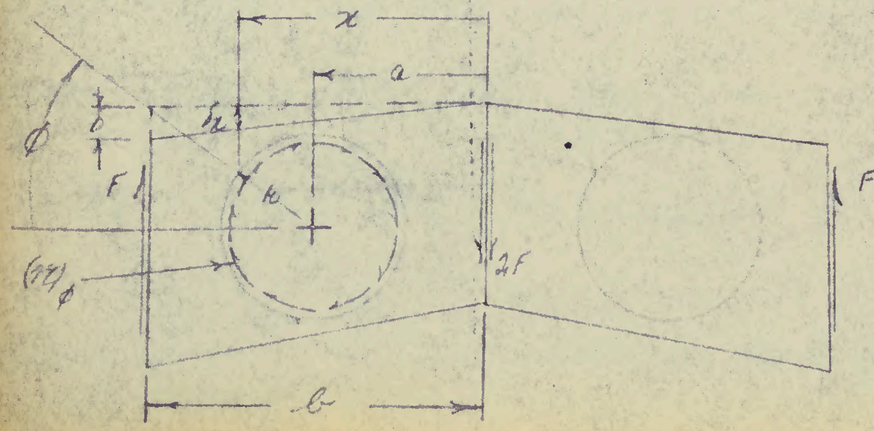


FIG. 1.

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

REPORT No. 7/0510/9
SHEET No. 13 - 49

AIRCRAFT:		PREPARED BY	DATE
C-105	CENTRE SECTION	E. AUGUSTINE	17/OCT/55
		CHECKED BY	DATE

FROM TEST RESULTS IT HAS BEEN ESTABLISHED THAT FRAME LOADING OF THE REDUNDANT TRIPLE 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 —

$$F = k \cdot \delta \qquad (k = 10 \text{ KIPS/INCH})$$

SO STRAIN ENERGY OF FRAME (ONE SIDE) —

$$V_{1f} = \tfrac{1}{2} F \cdot \delta = \tfrac{1}{2} \cdot \tfrac{1}{k} \cdot F^2 .$$

APPLY TO THE DUCT CUT-OUT, ON THE FRAME, THE GENERAL SHEAR FLOW $(\tau t)_\phi$ (REF. PG. 21.).

$$(\tau t)_\phi = 2R/h \sum_{n=1}^{\infty} \left(\frac{\sigma_{a_n} \cdot t}{n}\right) \cdot \sin n\phi + \left(\frac{\sigma_{b_n} \cdot t}{n}\right) \cdot \cos n\phi$$

NOTE : EFFECTS OF THE CONSTANT TERM K ARE IGNORED.

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

AIRCRAFT: C-105	CENTRE SECTION	REPORT No. 7/0510/9	
		SHEET No. 13 - 50	
		PREPARED BY E. BUGHSTINE	DATE 17/OCT/55
		CHECKED BY	DATE

CALCULATE THE WORK DONE ($V_{2\phi}$) BY THE SH. FLOW $(\tau t)_\phi$ DURING PERIPHERAL MOVEMENT OF THE CIRCULAR DUCT CUT OUT DUE TO DISTORTION δ.

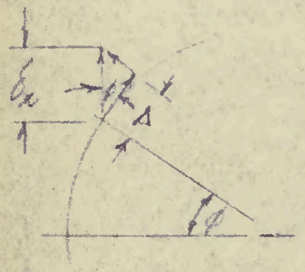


FIG. 2.

$$V_{2\phi} = \tfrac{1}{2}\int_0^{2\pi} r(\tau t)_\phi \cdot \Delta\phi \cdot d\phi \cdot (\text{REF. FIG. 2}).$$

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

$$\delta_x = x/a \cdot \delta$$

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

$$\therefore\; V_{2\phi} = \left(\frac{r^2}{4bh}\right)\cdot F\int_0^{2\pi}\left(\left(\sum\left(\frac{\tau_{Bn}t}{n}\right)\cdot \sin n\phi + \left(\frac{\tau_{An}t}{n}\right)\cdot\cos n\phi\right)(\cos\phi + r)\,d\phi\right)$$

$$= \left(\frac{\pi r^2}{4bh}\right)\cdot F\left\{a\,(\tau_{A1}\cdot t) + \frac{r}{4}\left(\frac{\tau_{A2}\cdot t}{2}\right)\right\}$$

THE TERM τ_{A1} ARISING FROM A DUCT BENDING TYPE LOADING IS IGNORED (REF. 9-47 & 10).

$$\therefore\; V_{2\phi} \simeq \frac{\pi r^3}{2\cdot 4bh}\left(\frac{\tau_{A2}\cdot t}{2}\right)\cdot F$$

FORM 1319A

REPORT NO. 7/0510/9
SHEET NO. 13-51

AIRCRAFT:		PREPARED BY	DATE
C-105	CENTRE SECTION	R. [illegible]	12/OCT/55
		CHECKED BY	DATE

FROM VIRTUAL DISPLACEMENT —

$$\delta(V_{1F} + V_{2F}) = 0$$

$$\therefore \quad F = -\left(\frac{\pi a^2}{64}\right)\left(\frac{\bar{r}_A t}{2}\right)$$

i.e. AS A CONSEQUENCE OF THE ASSUMED MODE OF DISTORTION OF THE FRAME, ONLY A SECOND HARMONIC COSINE SHEAR FLOW* APPLIED ON THE DUCT CUT-OUT DISTORTS THE FRAME & HENCE CREATES A REACTION AT FRAME SIDES.

* $(q_0)_{n=2} = 2^1/_4 \cdot \left(\frac{\bar{r}_A t}{2}\right) \cdot \cos 2\phi$ — THIS IS A DUCT HARMONIC TYPE SHEAR FLOW. (REF. PG 12.)

A. V. ROE CANADA LIMITED MALTON ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT No. 7/0510/9 SHEET No. 13-52	
AIRCRAFT:		PREPARED BY	DATE
		E. AUGUSTINE	17/OCT./58
C-105	CENTRE SECTION	CHECKED BY	DATE

EQUIVALENT MOMENT OF INERTIA OF FRAME

STRAIN ENERGY ($2V_f$) OF FRAME AS A RING —

$$2V_f = \frac{4\pi a^7}{4EI_f h^2} \sum_{n=1}^{\infty} \frac{(\sigma_{A_n} t)^2 + (\sigma_{B_n} t)^2}{n^4 (n^2-1)^2}$$

σ_{A_n} - COEFF. OF SIN TERMS

σ_{B_n} - " " COS "

SINCE THE FRAME IS ASSUMED TO DISTORT UNDER WARPING TYPE DUCT STRESS ONLY, $n=2$ (IN $2V_f$) GIVES THE ONLY TERM, $2V_f'$ THAT EXPRESSES AN ENERGY EQUIVALENT TO THAT OF FRAME DISTORTION.

$$\therefore \quad 2V_f' = \frac{\pi a^5}{4EI_f} \frac{(\tau c)^2}{36}$$

WHERE $(\tau c) = 2\frac{a}{h}\left(\frac{\sigma_{B_2} t}{2}\right)$

$$\therefore \quad 2V_f' = \frac{\pi a^7}{4EI_f h^2} \frac{(\sigma_B t)^2}{36}$$

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

AIRCRAFT: C-105	CENTRE SECTION	REPORT No 7/0510/9	
		SHEET No 13-53	
		PREPARED BY	DATE
		CHECKED BY	DATE

STRAIN ENERGY ($2V_f$) OF FRAME AS A TRAPEZOIDAL PANEL — (REF. PG. 49)

$$2V_f = (10)^{-2} \cdot F^2$$

SUBSTITUTE FOR $F = -\left(\frac{\pi R^3}{b h}\right)\left(\frac{\sigma_B \cdot t}{2}\right)$ (REF. PG. 51)

$$\therefore \quad 2V_f = (10)^{-4} \cdot \frac{\pi^2 R^6}{b^2 h^2} \cdot \frac{(\sigma_B \cdot t)^2}{4}$$

EQUATING $2V_f'$ & $2V_f$ & SOLVING FOR EQUIVALENT MOMENT OF INERTIA — I_f.

$$I_f = \frac{\pi \overset{12}{R^7}}{\underset{9\ (10.6)(10)^6\ \pi}{36\, b\, h^2}} \times \frac{4 b^2 h^2}{\pi^2 R^6} \overset{10^{-2}}{(10)^4} = \frac{R\, b^2}{9 \times 10.6\, \pi}(10)^{-2} = 1.23\ \text{in.}^4$$

FORM 1319A

	401	402	403	404	405	406	407	408	409	410	411	412	413	414	415	416	417	418	419	420	421	422	423	424	425	426	427	428	429	430	431	432	433	434		Check Sums
401	+.000783												+.000390																							+.001173
402		+.000779												+.000386																						+.001165
403			+.005850																																	+.005850
404				+.005506																																+.005506
405					+.003180												-.001792		+.006736																	+.008124
406						+.009945												-.008558																		+.001387
407							+.014702																													+.014702
408								+.005709																												+.005709
409									+.001218												+.000621															+.001839
410										+.001634												+.000834														+.002468
411											+.005484																									+.005484
412												+.001722																								+.001722
413	+.000390												+.001555												+.000386											+.002331
414		+.000386												+.001532												+.000382										+.002300
415															+.005850																					+.005850
416																+.005506																				+.005506
417					-.001792												+.030634		-.060201																	-.031359
418						-.008558												+.065816																		+.057257
419					+.006736												-.060201		+.556988																	+.503523
420																				+.005709																+.005709
421									+.000621												+.002524							+.000633								+.003778
422										+.000834												+.003389							+.000851							+.005074
423																							+.009141													+.009141
424																								+.001722										+.019708		+.021430
425													+.000386												+.002887											+.003273
426														+.000382												+.000760										+.001142
427																																				
428																					+.000633							+.004800								+.005433
429																						+.000851							+.001717							+.002568
430																														+.010447	+.002283	-.018965	-.008160			-.014395
431																														+.002283	+.008960	+.009531	+.003677			+.021451
432																														-.018965	+.009531	+.131619	+.028496			+.150681
433																														-.008160	+.003677	+.028496	+.011841			+.035854
434																								+.019708										+.226215		+.245923
Check Sums	+.001173	+.001165	+.005850	+.005506	+.008124	+.001387	+.014702	+.005709	+.001839	+.002468	+.005484	+.001722	+.002331	+.002300	+.005850	+.005506	-.031359	+.057257	+.503523	+.005709	+.003778	+.005074	+.009141	+.021430	+.003273	+.001142		+.005433	+.002568	-.014395	+.021451	+.150681	+.035854	+.245923		+1.097597

Cik MATRIX

AVRO STRUCTURE

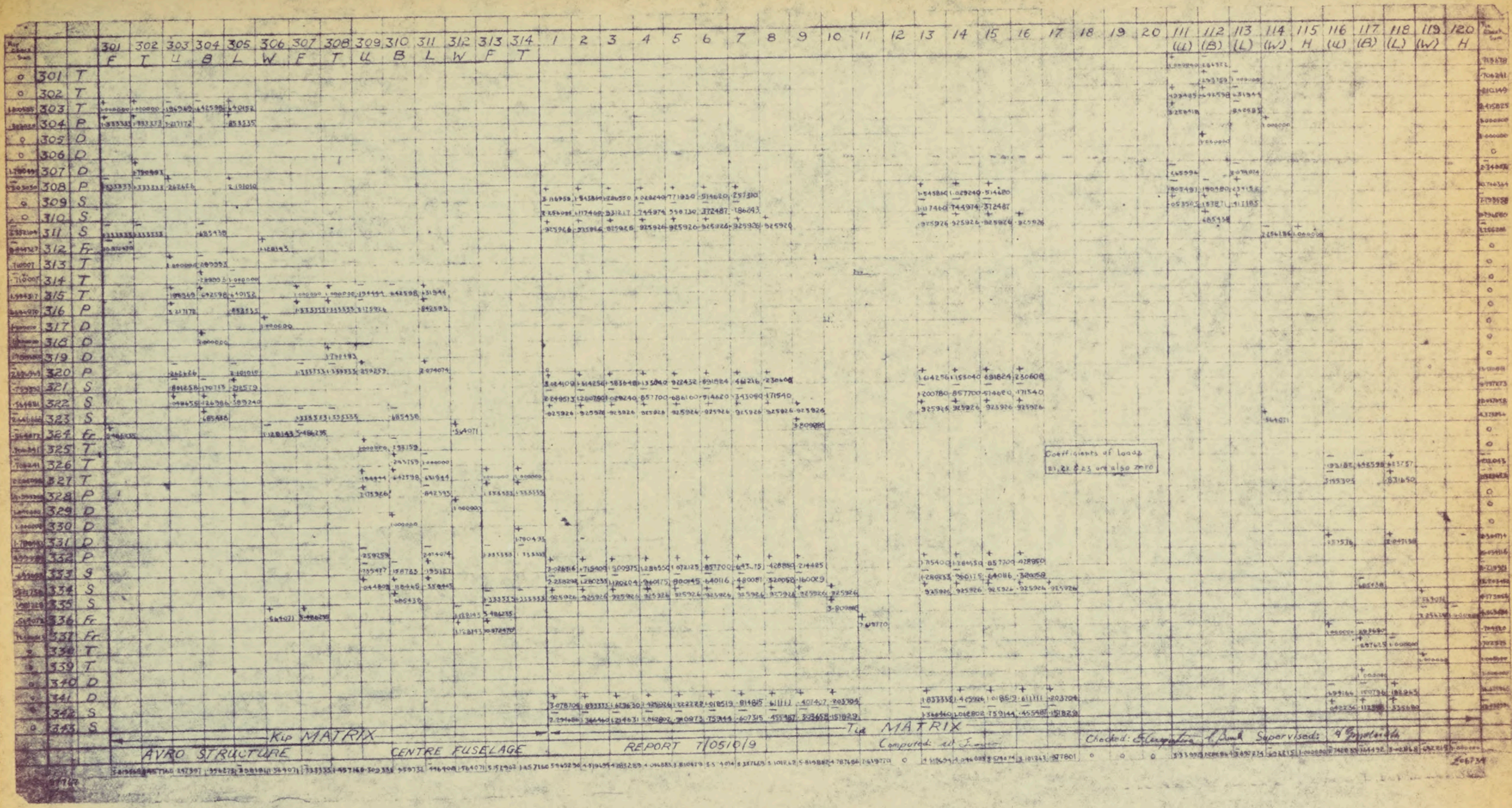
CENTRE FUSELAGE

REPORT: 7/0510/9

COMPUTED: W. GROVER, C B E a.

CHECKED: C B E a

SUPERVISED



Ref. Check Sum

301 F · 302 T · 303 U · 304 B · 305 L · 306 W · 307 F · 308 T · 309 U · 310 B · 311 L · 312 W · 313 F · 314 T · 1 · 2 · 3 · 4 · 5 · 6 · 7 · 8 · 9 · 10 · 11 · 12 · 13 · 14 · 15 · 16 · 17 · 18 · 19 · 20 · 111 (U) · 112 (B) · 113 (L) · 114 (W) · 115 H · 116 (U) · 117 (B) · 118 (L) · 119 (W) · 120 H

301 T
302 T
303 T
304 P
305 D
306 D
307 D
308 P
309 S
310 S
311 S
312 Fr
313 T
314 T
315 T
316 P
317 D
318 D
319 D
320 P
321 S
322 S
323 S
324 Fr
325 T
326 T
327 T
328 P
329 D
330 D
331 D
332 P
333 S
334 S
335 S
336 Fr
337 Fr
338 T
339 T
340 D
341 D
342 S
343 S

Coefficients of loads 21, 22 & 23 are also zero

Kip MATRIX

Tie MATRIX

AVRO STRUCTURE CENTRE FUSELAGE REPORT 7/05/0/9

Computed: [illegible]

Checked: [illegible] Supervised: [illegible]

	201	202	203	204	205	206	207	208	209	210	211	212	213	214	215	216	217	218	219	220	221	222	223	224	225	226	227	228	229	230	231	232	233	234	235	236	237	238	239	240	241	242	243	
201	+.000705												+.000442																															.001147
202		+.000725												+.000424																														.001149
203			+.007920																																									.007920
204				+.007454																																								.007454
205					+.002917												-.001040																											.001877
206						+.007914												-.006036																										.001878
207							+.019905																																					.019905
208								+.007729																																				.007729
209									+.000741												+.000425																							.001166
210										+.001211												+.000676																						.001887
211											+.009467																																	.009467
212												+.001722																																.001722
213	+.000442												+.002138												+.000538																			.003118
214		+.000424												+.001927												+.000489																		.002840
215															+.007920																													.007920
216																+.007454																												.007454
217					-.001040												+.005835												-.001040															.003755
218						-.006036												+.015829												-.006036														.003757
219																			+.019905																									.019905
220																				+.007729																								.007729
221									+.000425												+.001862												+.000511											.002798
222										+.000676												+.002926												+.000774										.004376
223																							+.009467																					.009467
224																								+.003445																				.003445
225													+.000538												+.002251													+.000552						.003341
226														+.000489												+.002022													+.000518					.003029
227																											+.007920																	.007920
228																												+.007454																.007454
229																	-.001040												+.005835											-.001040				.003755
230																		-.006036												+.015829											-.006036			.003757
231																															+.019905													.019905
232																																+.007729												.007729
233																					+.000511												+.002261									+.000607		.003379
234																						+.000774												+.003300									+.000864	.004938
235																																			+.007425									.007425
236																																				+.003445								.003445
237																																					+.001722							.001722
238																									+.000552													+.001088						.001640
239																										+.000518													+.001019					.001537
240																													-.001040											+.002917				.001877
241																														-.006036											+.007914			.001878
242																																	+.000607									+.001363		.001970
243																																		+.000864									+.001859	.002723
Check Sum	.001147	.001149	.007920	.007454	.001877	.001878	.019905	.007729	.001166	.001887	.009467	.001722	.003118	.002840	.007920	.007454	.003755	.003757	.019905	.007729	.002798	.004376	.009467	.003445	.003341	.003029	.007920	.007454	.003755	.003757	.019905	.007729	.003379	.004938	.007425	.003445	.001722	.001640	.001537	.001877	.001878	.001970	.002723	.229349

C_{ik} MATRIX

AVRO STRUCTURE

CENTRE FUSELAGE

REPORT 7/0510/9

Computed:

Checked:

Supervised:

	Row		401 F	402 T	403 U	404 B	405 L	406 W	407 F	408 T	409 U
	401	T									
	402	T									
+1.705325	403	T	+1.000000	+1.000000	-.259829	-.849980	+.833333				
-.466098	404	D	+1.333333	+1.333333	-4.243875		+1.111111				
	405	O									
	406	O									
+1.790493	407	O		+1.790493							
+.414816	408	D	-1.333333	-1.333333	+.346439		+2.735049				
	409	S									
	410	S									
-3.914400	411	S	-1.333333	-1.333333		-.927978					
-9.445138	412	Fr	-10.977470					+1.527332			
+.702375	413	T			+1.000000	-.297625					
-.698407	414	T				+.301593	-1.000000				
[illegible]	415	T			+.259829	-.849980	-.833333		+1.000000	+1.000000	-.256410
[illegible]	416	D			+4.243875		-1.111110		+1.333333	+1.333333	[illegible]
+1.000000	417	O						+1.000000			
+1.000000	418	O				+1.000000					
+1.790493	419	O								+1.790493	
[illegible]	420	D			-.346439		2.735049		-1.333333	-1.333333	+.341880
-.468451	421	S			.671445	-.146820	+.175794				
[illegible]	422	S			+.040819	+.109939	+.327256				
-1.085215	423	S				+.556787			.800000	.800000	
[illegible]	424	Fr	+10.977470					-3.054671	-10.977470		
1.000000	425	T									+1.000000
	426	T									
	427	R									
[illegible]	428	S									-.655957
[illegible]	429	S									+.099801
	430	Bd									
	431	Bd									
	432	Bd									
	433	Bd							-1.440000	1.440000	
[illegible]	434	Frt								+1.790493	
[illegible]			-.833333	+1.457160	+.369338	+.595296	-.901960	1.527332	-2.212420	+2.340986	-3.[illegible]

← Kio MATRIX →

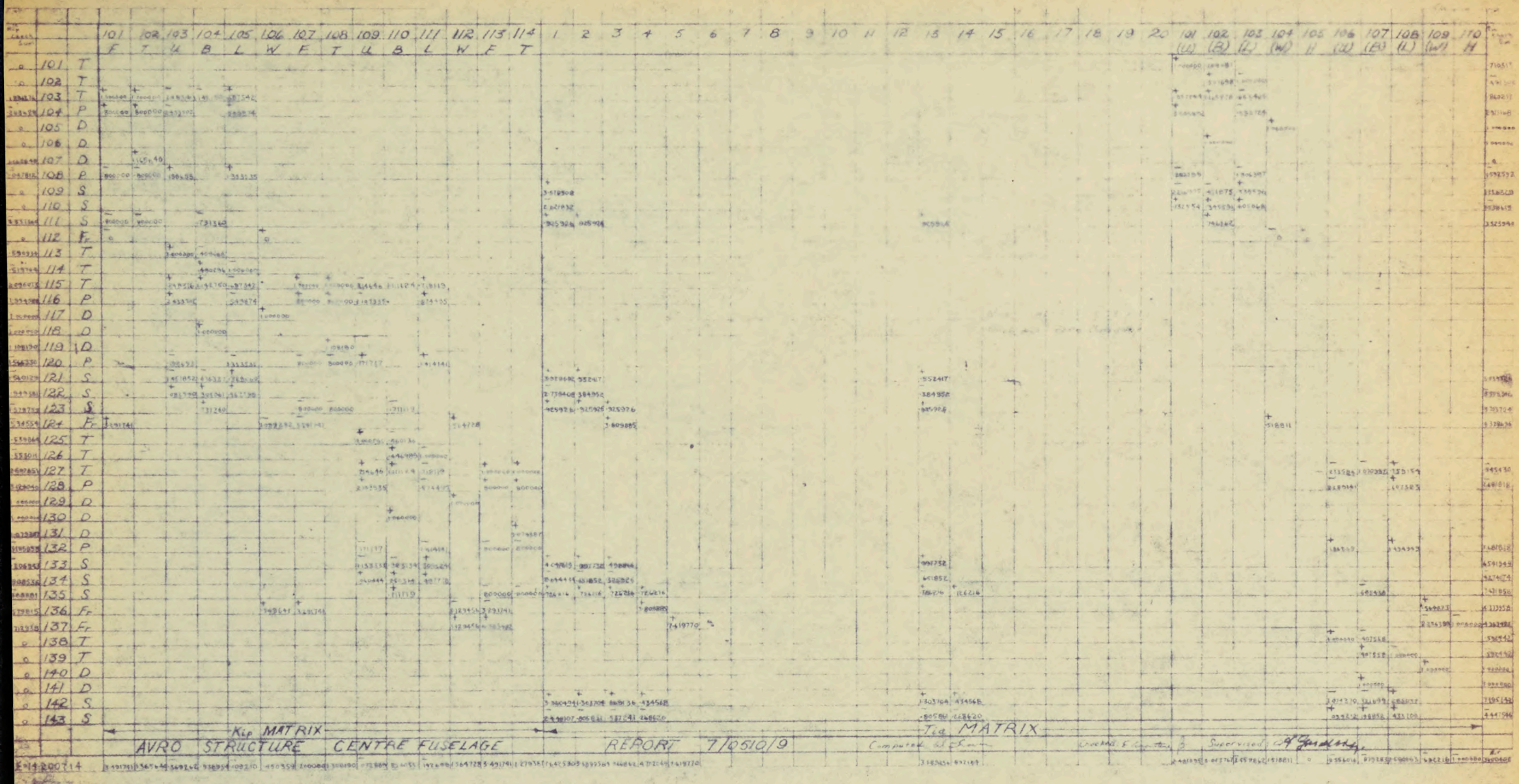
Row	1 <	2	3	4	5	6	7	8	9 2a	10	11	12
401 T												
402 T												
403 T												
404 D												
405 O												
406 O												
407 O												
408 D												
409 S	+[illegible]	1.833333	-1.629630	+1.425926	+1.222222	+1.018519	.814815	+[illegible]	+.407407	+.203704		
410 S	-2.291686	-1.366460	-1.214631	-1.062802	-.910973	-.759144	-.607315	-.455487	-.303658	-.151829		
411 S	+.925926	+.925926	+.925926	+.925926	+.925926	+.925926	+.925926	+.925926	+.925926	+.925926	+.925926	
412 Fr												
413 T												
414 T												
415 T												
416 D												
417 O												
418 O												
419 O												
420 D												
421 S	+3.144016	+1.951485	+1.793153	+1.534821	+1.336489	+1.138156	+.939824	+.741492	+.543160	+.344828	+.146495	
422 S	-2.341790	-1.438402	-1.290102	-1.143001	-.995510	-.847600	-.699899	-.552199	-.404498	-.256798	-.109097	
423 S	+.555556	.555556	+.555556	+.555556	+.555556	+.555556	+.555556	+.555556	+.555556	+.555556	.555556	+.555556
424 Fr												+.749770
425 T												
426 T												
427 R												
428 S	+3.757474	+2.057117	+1.860776	+1.664436	+1.468095	+1.271754	+1.075413	+.879072	+.682731	+.486390	+.290049	+.165026
429 S	-2.421227	-1.529022	-1.387085	-1.237148	-1.091211	-.945274	-.799337	-.653400	-.507463	-.361526	-.215589	-.107794
430 Bd												
431 Bd												
432 Bd												
433 Bd												
434 Frt												
Check Sums	+3.904373	+2.969533	+2.816623	+2.663714	+2.510574	+2.357893	+2.204983	+2.052071	+1.899161	+1.746251	+1.593340	+8.112056

Row	13	14	15	16	17	18 >	19 C'	20 S'	21 B'	22 L	23 H
401 T											
402 T											
403 T											
404 D											
405 O											
406 O											
407 O											
408 D											
409 S	+1.833333	+1.425926	+1.018519	+.611111	+.203704						
410 S	-1.366460	-1.062802	-.759144	-.455487	-.151829						
411 S	+.925926	+.925926	+.925926	+.925926	+.925926						
412 Fr											
413 T											
414 T											
415 T										+.827778	
416 D										+1.103704	
417 O											
418 O											
419 O											
420 D										+2.716809	
421 S	+1.951485	+1.534821	+1.138156	+.741492	+.344828						
422 S	-1.438402	-1.143001	-.847600	-.552199	-.256798						
423 S	+.555556	+.555556	+.555556	+.555556	+.555556	+.555556					
424 Fr											
425 T										-1.000000	
426 T											
427 R											
428 S	+2.057117	+1.664436	+1.271754	+[illegible]	-.486390	+.145024				+.172869	
429 S	-1.529022	-1.237148	-.945274	-.653400	-.361526	-.107794				-.316285	
430 Bd							+1.000000				
431 Bd								+1.000000			
432 Bd									+1.000000		
433 Bd											+1.000000
434 Frt											
Check Sums	+2.469533	+2.665746	+2.357893	+2.052071	+1.746251	.592786	+1.000000	+1.000000	+1.000000	+4.157445	1.000000

Row	116 (U)	117 (B)	118 (L)	119 (W)	120 H	Check Sums
401 T	+1.000000	-.295679				+.704321
402 T		+.297625	-1.000000			-.702375
403 T	+.261538	+.849980	-.844444			+.287074
404 D	+4.271795		1.125926			+1.165069
405 O				+1.000000		+1.000000
406 O		+1.000000				+1.000000
407 O						
408 D	-.346784		-2.771510			-3.120228
409 S	-.694[illegible]	-.150796	+.182963			+6.675966
410 S	+.042296	+.112995	+.335680			-12.432596
411 S		+.927978				+15.742796
412 Fr				-.399183	+1.000000	+.600816
413 T						
414 T						
415 T						+.827778
416 D						+1.103704
417 O						
418 O						
419 O						
420 D						+2.716809
421 S						+19.214[illegible]
422 S						+14.117116
423 S						+10.000000
424 Fr				+1.527332		+9.167107
425 T						1.000000
426 T						
427 R						
428 S						+3.114393
429 S						-15.369[illegible]
430 Bd						+1.000000
431 Bd						+1.000000
432 Bd						+1.000000
433 Bd						+1.000000
434 Frt						
Check Sums	+4.511[illegible]	+.746503	-5.223257	-2.128165	1.000000	[illegible]

Tio MATRIX

REPORT 7.0510/9   Computed: W. G[illegible] B. E. G.   Checked: B. E. G.   Supervised:



101 102 103 104 105 106 107 108 109 110 111 112 113 114 1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18 19 20 101 102 103 104 105 106 107 108 109 110

F T U B L W F T U B L W F T (U) (B) (L) (W) U (U) (B) (L) (W) H

101 T
102 T
103 T
104 P
105 D
106 D
107 D
108 P
109 S
110 S
111 S
112 F
113 T
114 T
115 T
116 P
117 D
118 D
119 D
120 P
121 S
122 S
123 S
124 Fr
125 T
126 T
127 T
128 P
129 D
130 D
131 D
132 P
133 S
134 S
135 S
136 Fr
137 Fr
138 T
139 T
140 D
141 D
142 S
143 S

Kip MATRIX

Tiq MATRIX

AVRO STRUCTURE CENTRE FUSELAGE REPORT 7/0510/9

Computed ... Checked ... Supervised ... Goodhugh

E = 14.200714

K_ip Check Σ			201	202	203	204	205	206	207	208	209	210	211	212	213	214	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	106	107	108	109	110	111	112	113	114	115	T_ia Check Sum
			F	T	U	B	L	W	F	T	U	B	L	W	F	T																					(U)	(B)	(L)	(W)	H	(U)	(B)	(L)	(W)	H	
0	201	T																																			+1.000000	−.407558									.592442
0	202	T																																				+.407558	−1.000000								−.592442
1.789561	203	T	+1.000000	+1.000000	−.190025	−.642598	+.562184																														+.145152	+.642598	−.455498								.327258
.312499	204	P	+1.333333	+1.333333	−3.103746		+.749579																														+2.289141		−.607323								1.681818
0	205	D																																						+1.000000							1.000000
0	206	D																																				+1.000000									1.000000
1.790493	207	D		+1.790493																																											0
−.568181	208	P	−1.333333	−1.333333	+.253367		+1.845118																														−.156869		−1.494949								−1.651818
0	209	S															+3.960494	+1.303704	+.869136	+.434568									+1.303704	+.434568							−1.074370	−.321699	+.285037								7.195142
0	210	S															−2.448107	−.805861	−.537241	−.268620									−.805861	−.268620							+.054215	+.138852	+.453700								−4.487543
−2.629103	211	S	−1.045752	−1.045752		−.537599											+.726216	+.726216	+.726216	+.726216	+.726216								+.726216	+.726216								+.537599									5.621111
−9.744327	212	Fr	−10.872470					+1.128143																																0	+1.000000						1.000000
.699409	213	T			+1.000000	−.300591																																									0
−.669789	214	T				+.330211	−1.000000																																								0
2.023674	215	T			+.190025	+.642598	−.562184		+1.000000	+1.000000	−.203283	−.642598	+.599116																																		0
2.499369	216	P			+3.103746		−.749579		+1.333333	+1.333333	−3.320286		+.798822																																		0
1.000000	217	D						+1.000000																																							0
1.000000	218	D				+1.000000																																									0
1.790493	219	D								+1.790493																																					0
−2.587777	220	P			−.253367		−1.845118		−1.333333	−1.333333	+.211044		+1.966330																																		0
−1.189724	221	S			−1.213910	−.268983	+.293169										+3.662551	+1.448560	+1.086420	+.724280	+.362140								+1.448560	+.724280																	9.456791
.696641	222	S			+.063402	+.171522	+.461717										−2.343539	−.926804	−.695103	−.463402	−.231701								−.926804	−.463402																	−6.050755
−2.091504	223	S				+.537599			−1.045752	−1.045752		−.537599					+.726216	+.726216	+.726216	+.726216	+.726216	+.726216							+.726216	+.726216	+.726216																6.535944
−.564072	224	Fr	+5.486235					−1.128143	−5.486235					+.564071								+3.809885																		+.564071							4.373956
.719033	225	T									+1.000000	−.280967																																			0
−.690144	226	T										+.309856	−1.000000																																		0
2.246765	227	T									+.203283	+.642598	−.599116		+1.000000	+1.000000																										−.198499	−.642598	+.631944			−.209153
5.188130	228	P									+3.320286		−.798822		+1.333333	+1.333333																										−3.258418		+.842593			−2.415825
1.000000	229	D												+1.000000																																	0
1.000000	230	D										+1.000000																																			0
1.790493	231	D														+1.790493																															0
−4.844040	232	P									−.211044		−1.966330		−1.333333	−1.333333																										+.265995		+2.074024			2.340019
−1.039801	233	S									−1.076528	−.222032	+.258759				+3.333333	+1.499653	+1.199727	+.899796	+.599864	+.299932							+1.499653	+.899796	+.299932																10.531686
.628423	234	S									+.040131	+.152064	+.436228				−2.282913	−1.027077	−.821662	−.616246	−.410831	−.205415							−1.027077	−.616246	−.205415																−7.212882
−1.181208	235	S										+.485458			−1.333333	−.333333	+.925926	+.925926	+.925926	+.925926	+.925926	+.925926	+.925926						+.925926	+.925926	+.925926												−.685418				8.573842
−.564072	236	Fr						+.564071	+5.486235					−1.128143	−5.486235								+3.809885																						+.564071		4.373956
12.100613	237	Fr												+1.128143	+10.972470									+7.619770																					0	+1.000000	8.619770
0	238	T																																								+1.000000	−.286527				.713473
0	239	T																																									+.293759	−1.000000			−.706241
0	240	D																																											+1.000000		1.000000
0	241	D																																									+1.000000				1.000000
0	242	S															+3.116959	+1.543860	+1.286550	+1.029240	+.771930	+.514620	+.257310						+1.543860	+1.029240	+.514620											−.905497	−.193480	+.334152			10.843364
0	243	S															−2.256085	−1.117460	−.931217	−.744974	−.558730	−.372487	−.186243						−1.117460	−.744974	−.372487											+.053505	+.157871	+.417185			−7.773556
Σ 8.873714			−5.531987	1.744741	−.150508	.933059	.245114	1.864071	−.045752	1.744741	.015347	1.104740	−.305013	1.564071	5.152902	1.457160	7.121251	4.296939	3.834968	3.373000	2.911030	5.698677	4.806878	7.619770	0	0	0	0	4.296939	3.373000	1.888792	0	0	0	0	0	[illegible]	[illegible]	2.654021	1.564071	1.000000	3.045914	.373208	3.199948	.564071	1.000000	Σ 53.572860

K_ip MATRIX

T_ia MATRIX

AVRO STRUCTURE

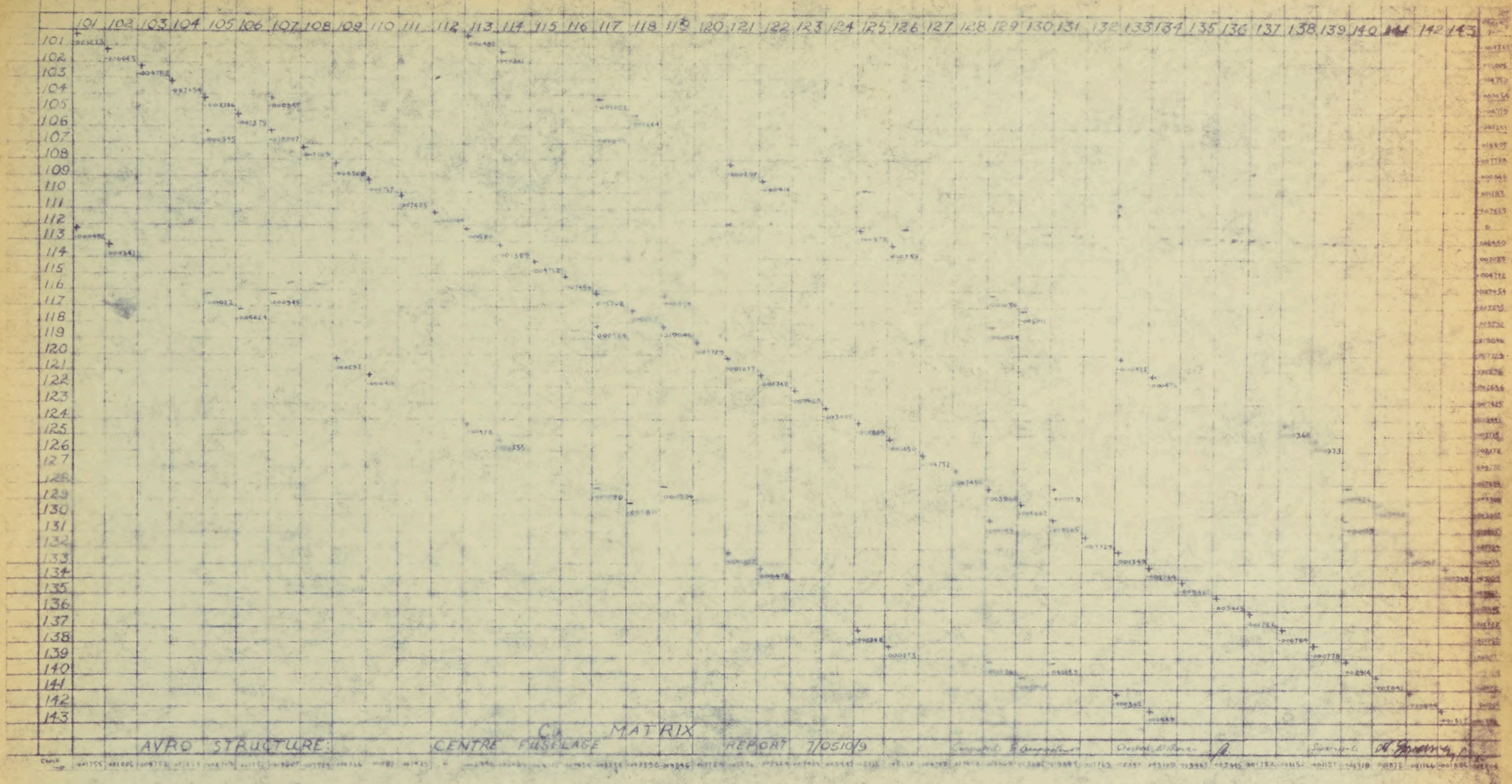
CENTRE FUSELAGE

REPORT 7/0510/9

Computed:

Checked: E.

Supervised: A. Gryglew



AVRO STRUCTURE: CENTRE FUSELAGE C_A MATRIX REPORT 7/0510/9

	301	302	303	304	305	306	307	308	309	310	311	312	313	314	315	316	317	318	319	320	321	322	323	324	325	326	327	328	329	330	331	332	333	334	335	336	337	338	339	340	341	342	343	Check Sum
301	+.001097											+.000543																														[illegible]
302		+.001062											+.000536																													[illegible]
303			+.007920																																								[illegible]
304				+.007454																																							.007454
305					+.002817											−.001040																										.001877
306						+.007914											−.006036																									[illegible]
307							+.019905																																				.019905
308								+.001729																																			.001729
309									+.001217											+.000693																						.001970
310										+.001778											+.000945																					.002723
311											+.007425																																.007425
312												+.001728																															[illegible]
313	+.000543											+.002159											+.000536																		[illegible]
314		+.000536											+.002165											+.000536																	.003237
315															+.007920																												.007920
316																+.007454																											[illegible]
317					−.001040											+.005835											−.001040														[illegible]
318						−.006036											+.015827											−.006036													[illegible]
319																			+.019905																								.019905
320																				+.007729																							.007729
321									+.000693											+.002945											+.000760										.004398
322										+.000945											+.003948											+.001020									[illegible]
323																							+.007425																				[illegible]
324																								+.003445																			.003445
325													+.000536											+.002128												+.000531					.003196
326														+.000536											+.002131												+.000529				.003196
327																											+.007920																.007920
328																												+.007454															.007454
329																	−.001040											+.005835										−.001040			.003755
330																		−.006036											+.015829										−.006036		.003757
331																															+.019905												.019915
332																																+.007729											.007729
333																					+.000760											+.003182								+.000809	.004731
334																						+.001020											+.004235								+.001085	.006340
335																																			+.007425								.007425
336																																				+.003445							.003445
337																																					+.001722						.001722
338																									+.000531												+.001057					.001588
339																										+.000529												+.001048				.001577
340																													−.001040										+.002917			.001877
341																														−.006036										+.007914		.001878
342																																	+.000809								+.001681	.002890
343																																		+.001085								+.002258	.003342
Check Sum	.001640	.001598	.007920	.007454	.001877	.001878	.019905	.007729	.001970	.002725	.007425	.001722	.003238	.003237	.007920	.007454	.003755	.003757	.019905	.007729	.004398	.005913	.007425	.003445	.003196	.003196	.007920	.007454	.003755	.003797	.019905	.007729	.004751	.006340	.007425	.003445	.001722	.001588	.001577	[illegible]	.001878	.002490	.003342	[illegible]

AVRO STRUCTURE

CENTRE FUSELAGE

C_{ik} MATRIX

REPORT 7/0510/9

Computed: E. [illegible]

Checked:

Supervised:

ESS REPORT

/0510/9

ummary

ENTRE
SELAGE
ALYSIS

FIDENTIAL

y authority of
Date 30 Sept 01
Signature
/ Rank / Appointment