STRESS REPORT

7/0510/13

Issue 2

LOAD SYSTEM
OF THE
GENERAL STRESS
ANALYSIS OF
THE A/C

~~CONFIDENTIAL~~

Classification changed / Changed to UNCLAS
By authority of AVES
Date 30 Sept 60
Signature [illegible]
Unit / Rank / Appointment AVES

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

ANALYZED

AIRCRAFT: C 105

REPORT NO. 7/0510/13

Classification cancelled / Changed to UNCLASS
By authority of AVRS
Date 30 Sept 96
Signature [signature]
Unit / Rank / Appointment AVRS 5

FILE NO:

NO. OF SHEETS:

CONFIDENTIAL

TITLE: Load System of the General Stress Analysis of the Aircraft

NOTE : The load point indices, a as used in this report are not the same as the load point indices, A as used in report 7/0510/3 entitled, "General Aircraft Analysis For Symmetrical Loading Cases".

PREPARED BY Alex. Grzedzielski, R. N. Shearly DATE May 1956

CHECKED BY [signature] DATE May 1956

SUPERVISED BY DATE

APPROVED BY DATE

ISSUE No.	REVISION No.	REVISED BY	APPROVED BY	DATE	REMARKS
2	105	R. Shearly	[signature]	Sept. 1956	Altered to handle any number of cases, with each case handled separately. Sheets revised 3 to 10.

FORM 1316 A

15865958

AVRO AIRCRAFT LIMITED
MALTON - ONTARIO
TECHNICAL DEPARTMENT

REPORT No. 7/0510/13

SHEET No. 1

AIRCRAFT: C 105

PREPARED BY	DATE
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SUMMARY

This report establishes unit load coordinates to be used with the general stress analysis. It suggests also a method of obtaining a balanced load distribution, compatible with the general procedure.

A) Unit load coordinates.

B) Balanced load system.

C) Use of a digital computer.

A) UNIT LOAD COORDINATES.

Table I gives coordinates of 91 unit loads with respect to two systems of reference; to the general D.O. system, and to the system adopted in the wing stress analysis for convenience of mathematical formulation. Any system can be used for balancing the aircraft. Unit loads 90 and 91 are reactions for the load system of the general stress and displacement analysis. Hence all computed displacements have to be referred to the 90 and 91 loading points as a datum. Care should be taken that all loads are properly balanced, in other words, that the loads 90 and 91 are as required, since otherwise a considerable error may result in the vicinity of those loads as concerns all web shearing stresses.

FORM 1319A

AVRO AIRCRAFT LIMITED
MALTON - ONTARIO
TECHNICAL DEPARTMENT

AIRCRAFT:		REPORT No. 7/0510/13	
C 105		SHEET No. 2	
		PREPARED BY	DATE
		CHECKED BY	DATE

References:

Avro Gen/1090/336 Theory of Multi-Spar and Multi-Rib Wing Structures, June 1955; Alex. Grzedzielski.

Avro 7/0510/7 Wing Analysis, June 1955; V.F.Gardener, R.N.Shearly and D.L.Turner.

Avro 7/0510/8 Tank 3 Analysis, March 1955; Alex. Grzedzielski and W.Roberts.

Avro 7/0510/9 Centre Fuselage Analysis, Dec. 1955; E.Augustine, V.F.Gardener and C.Burrell.

FORM 1319A

B) BALANCED LOAD SYSTEM.

At present the general stress analysis neglects all stress and displacement due to what can be referred to as drag loads. For that reason all unit loads of the analysis are perpendicular to the wing chord plane and also to the fuselage axis. Some effects of the drag loads can not be neglected, however. The fin drag load is reacted at loading points 67 and 70. The wing drag load* generates a bending moment in the chord plane and because of the wing anhedral, kink loads appear in loading points 64, 65, 66, 67 and possibly in 74. Engine mount reactions add a considerable pitching moment. Similarily the fuselage drag creates a moment to be represented by an equivalent set of loads. It is understood that the latter effect can be accounted for in an approximate manner only. Sept/56

In general airloads are not equilibrated without inertia loads. Suppose therefore that in each loading point are known:

a) the magnitude of the airlift, real or equivalent,

b) the associated lumped mass.

Then the total loads (air and inertia) can be determined in the following way. Denote

x_a	the coordinate of a,	
m_a	the lumped mass at a,	
P_a	lift load at a,	(Positive down)
Q_a	total load (lift and inertia) at a.	(positive down) Sept/56
$M = \text{Sum } m_a$	total mass,	
$S = \text{Sum } x_a m_a$	total mass moment,	
$I = \text{Sum } x_a^2 m_a$	total moment of inertia,	

* For a first approximation the wing drag load affect is neglected. Sept/56

x_m	$= S/M$	coordinate of center of gravity,	*
x_g	$= \sqrt{I/M}$	radius of gyration.	*
(P)	$= \text{Sum } P_a$	total lift,	
(xP)	$= \text{Sum } x_a P_a$	total lift moment,	
x_p	$= (xP)/(P)$	coordinate of center of pressure.	*
u		linear acceleration perpendicular to the chord plane, (Positive down)	*
γ		angular acceleration in pitching motion. (Positive nose down)	*

* in reference to the system of coordinates

Sept/56

According to the principles of dynamics, two equations can be written and u and γ can be determined.

Equilibrium of forces perpendicular to the wing chord plane

$$\text{Sum } (u + x_a\gamma)\, m_a = \text{Sum } P_a \qquad \text{or} \qquad uM + \gamma S = (P)$$

Equilibrium of moments around the pitching axis

$$\text{Sum } (x_a u + x_a^2\gamma)\, m_a = \text{Sum } x_a P_a \qquad \text{or} \qquad uS + \gamma I = x_p (P)$$

Solving the above two equations for u and γ and substituting Mx_g^2 and Mx_m for I and S resp. yields

$$u = \frac{(P)}{M} \frac{x_g^2 - x_p x_m}{x_g^2 - x_m^2}$$

$$\gamma = - \frac{(P)}{M} \frac{x_m - x_p}{x_g^2 - x_m^2}$$

The linear acceleration of the center of gravity due to the angular acceleration about the origin is

$$x_m \gamma = -\frac{(P)}{M} \frac{x_m^2 - x_p x_m}{x_g^2 - x_m^2}$$

and consequently the total linear acceleration of the center of gravity is

$$u_m = u + x_m \gamma = \frac{(P)}{M}$$

The total load at a can be obtained from the following equation

$$Q_a = P_a - (u + x_a \gamma) m_a$$

which, when the above expressions for u and γ are substituted, becomes

$$Q_a = P_a - \frac{(P)}{M}\left[\frac{x_g^2 - x_p x_m - (x_m - x_p) x_a}{x_g^2 - x_m^2} \right] m_a$$

The above calculations are required for any particular loading case.

The choice of which loading cases are to be considered is not within the scope of this report. In the next section there are a few words dealing with the identification of loading cases and information to ensure that for a given set of airloads the appropriate set of lumped weights is used to form a particular loading case.

Sheet 5b

Sept/56

C) USE OF A DIGITAL COMPUTER.

This section deals with the preparation of data and with the programming of a digital computer to carry out the computations.

I INPUT DATA (Punched cards)

1 set of arms x_a

3 sets of weights W_a

k sets of lift loads P_a

where $a = 1 - 91$ inclusive

k = the number of symmetric loading cases considered.

The loading cases are identified by an identification mark as shown below:

Case ✕✕ . ✕✕ (*)

1st number — 2nd number — letter

where (i) the first number denotes the flight envelope case

(ii) the second number denotes the sub case, which depends on CG position and variation of aerodynamic coefficients due to manufacturing tolerances

(iii) the letter inside the parentheses denotes whether there is a pitching acceleration or not.

To ensure that the correct set of lumped weights is used with any given set of lift loads the following information is given. If the second number denoting the sub case is $\begin{Bmatrix} 5 \text{ or } 8 \\ 6 \text{ or } 9 \\ 7 \text{ or } 10 \end{Bmatrix}$ use the set of lumped weights for CG at $\begin{Bmatrix} 28\% \ \bar{c} \\ 30\% \ \bar{c} \\ 32\% \ \bar{c} \end{Bmatrix}$.

Secr/9.

II COMPUTATIONS

The following calculations will be made for each case:

1) $m_a = W_a/g$

2) $M = \text{Sum } m_a$ print

3) $S = \text{Sum } x_a m_a$

4) $I = \text{Sum } x_a^2 m_a$

5) $x_m = S/M$ print

6) $x_g^2 = I/M$ print

7) $A = x_m^2$

8) $B = x_g^2 - A$

9) $(P) = \text{Sum } P_a$ print

10) $(xP) = \text{Sum } x_a P_a$

11) $x_p = (xP)/(P)$ print

12) $u_m = (P)/M$ print

13) $C = x_p x_m$

14) $D = A - C$

15) $E = D/B$

Sept/9.

16) $x_m Y = -u_m E$ print

17) $F = x_g^2 - C$

18) $G = x_m - x_p$

19) $H = F/B$

20) $J = G/B$

21) $K = H\,u_m$

22) $L = J\,u_m$

23) $N_a = K - Lx_a$ For each point a

24) $R_a = N_a m_a$ For each point a

25) $Q_a = P_a - R_a$ For each point a print

26) $(Q) = \text{Sum } Q_a$

27) $(xQ) = \text{Sum } x_a Q_a$

Checks: (Q) and (xQ) should be equal to zero.

These computations will be repeated for each case considered.

III SUGGESTED OUTPUT FORMAT

It is desirable to have the printed results of the calculations on paper, size 8 1/2" x 11", from which reproductions can be obtained readily. The information to be printed in the following form, one case to a sheet.

Sept/2.

Case XX . XX (*)

(W)	x_m	x_g^2		
(P)	x_p		u_m	$x_m \gamma$
Q_1	Q_2	Q_3	Q_4	Q_5
Q_6	–	–	–	–
–	–	–	–	Q_{90}
Q_{91}				

where (W) = Sum W_a

IV ADDITIONAL INFORMATION

x_a will be given in inches, W_a and P_a will be given in lbs.

Q_a to be printed in kip units.

$$g = 386.4 \text{ in/sec}^2$$

Range of values

$$-75 \text{ inches} < x_a < 650 \text{ inches}$$

$$46{,}000 \text{ lbs} < \text{Sum } W_a < 50{,}000 \text{ lbs}$$

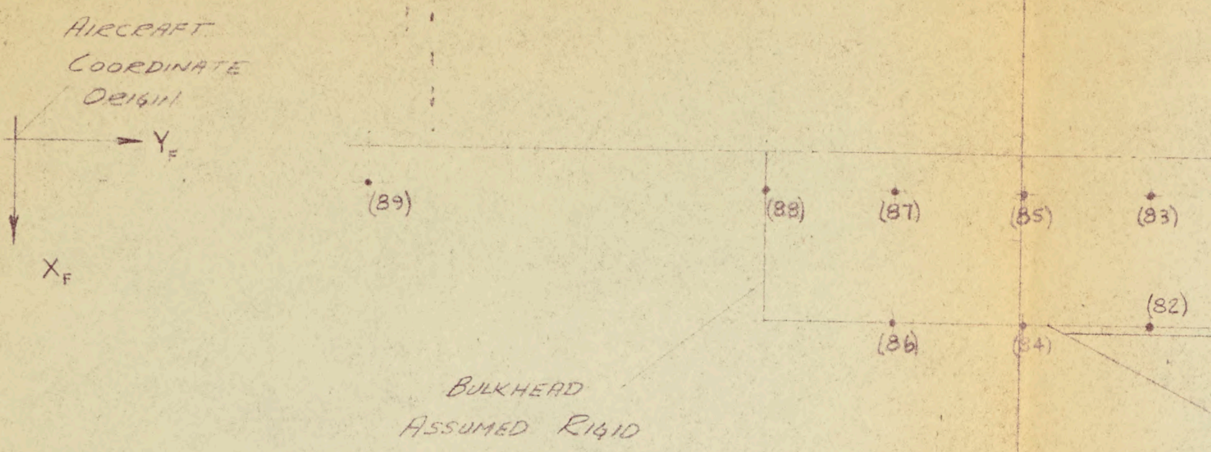
$$-200{,}000 \text{ lbs} < \text{Sum } P_a < 400{,}000 \text{ lbs}$$

$$200 \text{ inches} < x_m < 225 \text{ inches}$$

$$275 \text{ inches} < x_g < 300 \text{ inches}$$

$$50 \text{ inches} < x_p < 512 \text{ inches}$$

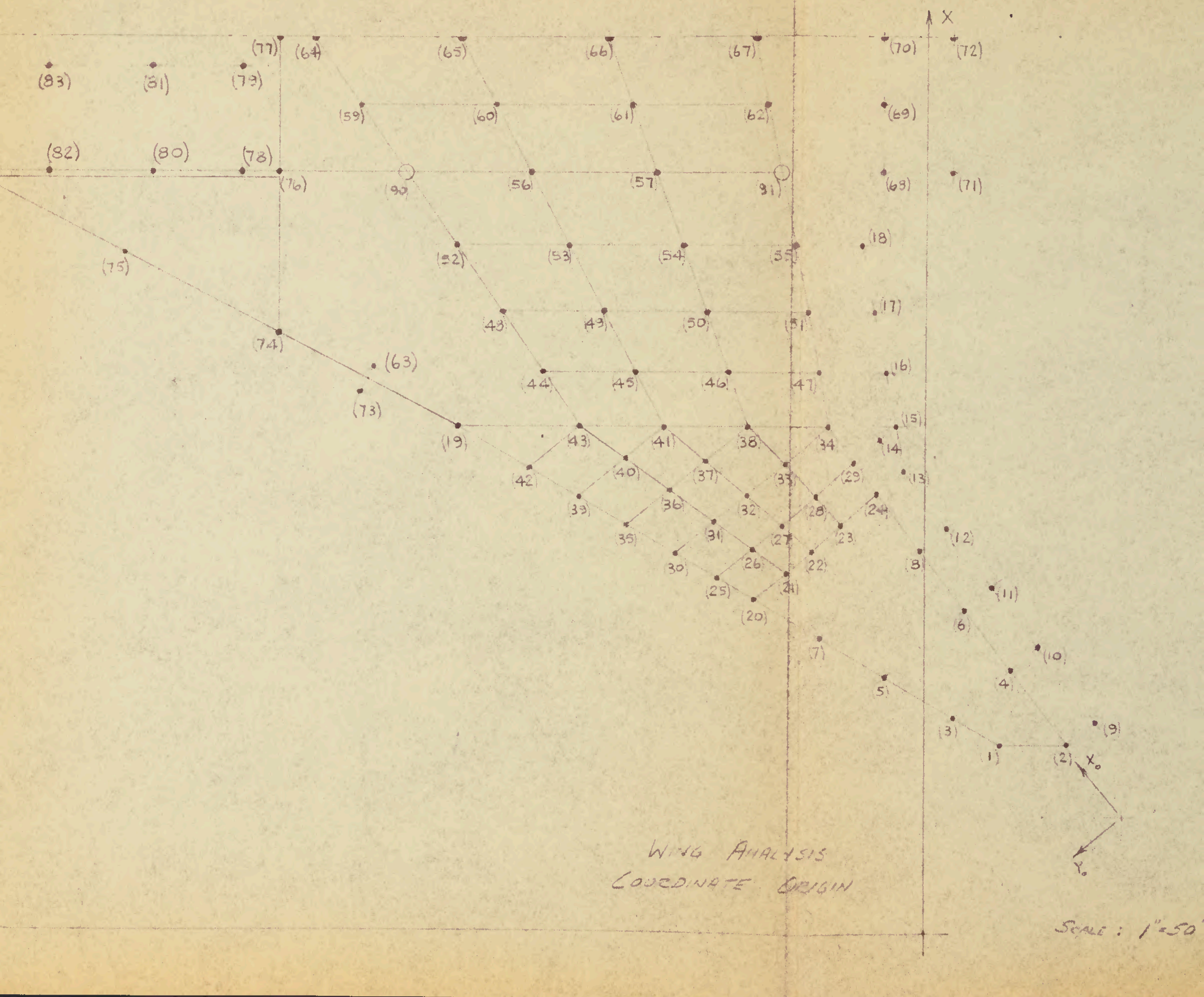
LOAD



SEPT/56

NOTE! THE LOAD POINT INDICES, a AS SHOWN HERE ARE <u>NOT</u> THE SAME AS THE LOAD POINT INDICES, A AS USED IN REPORT 7/0510/3 ENTITLED GENERAL AIRCRAFT ANALYSIS FOR SYMMETRICAL LOADING CASES.

LOAD POINTS FOR GENERAL STRESS ANALYSIS
OF THE AIRCRAFT
FIG. 1



STRESS ANALYSIS

...AFT

REPORT No 7/05 10/13

SHEET 10

DATE MAY 1956

PREPARED BY R.N. SHERRIFF

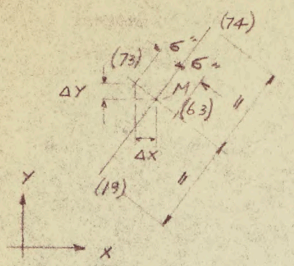
[illegible]

SCALE: 1" = 50"

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

REPORT No. 7/0510/13
SHEET No. 11

AIRCRAFT:		PREPARED BY	DATE
C-105	LOAD SYSTEM of GENERAL STRESS ANALYSIS	R.H. SHEARLY	MAY 1956
		CHECKED BY	DATE



FINDING X & Y FOR POINTS a = 73 & 63 KNOWING X & Y FOR POINTS a = 19 & 74.

POINT	X	Y
19	215.5596	200.2698
74	254.8909	276.5201

FOR POINT M

$X = \frac{1}{2}(215.5596 + 254.8909) = 235.22525$

$Y = \frac{1}{2}(200.2698 + 276.5201) = 238.39495$

FINDING COORDINATES OF POINTS a = 63 & 73.

$X_{74} - X_{19} = 254.8909 - 215.5596 = 39.3313$
$Y_{74} - Y_{19} = 276.5201 - 200.2698 = 76.2503$

DISTANCE BETWEEN POINTS a = 19 & 74 = $\sqrt{39.3313^2 + 76.2503^2} =$

$= 85.7966165$

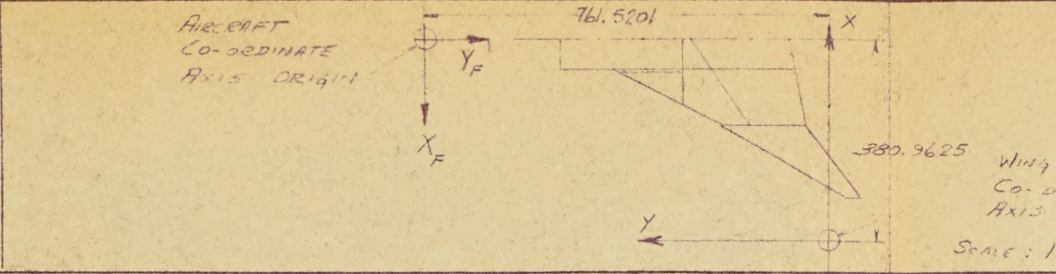
$\Delta X = \frac{6 \times 76.2503}{85.7966165} = 5.33240$

$\Delta Y = \frac{6 \times 39.3313}{85.7966165} = 2.75055$

$X_{63} = 235.22525 + 5.33240 = 240.5576$
$Y_{63} = 238.39495 - 2.75055 = 235.6444$

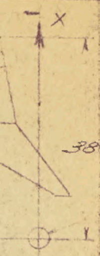
$X_{73} = 235.22525 - 5.33240 = 229.8928$
$Y_{73} = 238.39495 + 2.75055 = 241.1455$

FORM 1319A



LOAD POINT (a)	1 X	2 Y	3 X_F = 380.9625 − X	4 Y_F = 761.5201 − Y	5	6 LOAD POINT (a)	7 X	8 Y	9 X_F = 380.9625 − X	Y_F = 761.5… − Y
1	80.2204	‾31.7519	300.7421	793.2720		31	175.2265	91.5321	205.7360	669.
2	80.2204	‾60.9053	300.7421	822.4254		32	186.3011	78.0988	194.6614	683.
3	91.6120	‾12.2222	289.3505	773.7423		33	199.5178	62.0670	181.4447	699.
4	111.8211	‾36.7356	269.1414	798.2557		34	215.5536	42.6084	165.4029	718.
5	108.7974	17.2401	272.1651	744.2800		35	173.6993	128.5068	207.2632	633.
6	137.2570	‾17.2811	243.7055	778.8012		36	188.6708	110.3465	192.2917	651.
7	125.9828	46.7024	254.9787	714.8177		37	200.9298	95.4764	180.0327	666.
8	162.6929	2.1735	218.2696	759.3466		38	215.5596	77.7308	165.4029	683.
9	89.7625	‾72.4798	291.2000	833.9999		39	185.6972	149.0759	195.2653	612.
10	121.3632	‾48.3101	259.5993	809.8302		40	202.1152	129.1608	178.8473	632.
11	146.7991	‾28.8556	234.1634	790.3757		41	215.5596	112.8532	165.4029	648.
12	172.2350	‾9.4010	208.7275	770.9211		42	197.6952	169.6448	183.2673	591.
13	196.8497	9.4255	184.1128	752.0946		43	215.5596	147.9754	165.4029	613.5
14	209.9389	19.4366	171.0236	742.0835		44	238.5602	163.7649	142.4023	597.7
15	215.5596	12.6084	165.4029	748.9117		45	238.5602	124.8949	142.4023	636.6
16	238.5602	17.1548	142.4023	744.3653		46	238.5602	86.0248	142.4023	675.
17	264.0150	22.1683	116.9475	739.3518		47	238.5602	47.1548	142.4023	714.3
18	292.1859	27.7546	88.7766	733.7655		48	264.0150	181.2389	116.9475	580.
19	215.5596	200.2698	165.4029	561.2503		49	264.0150	138.2214	116.9475	623.
20	142.6134	75.2137	238.3491	686.3064		50	264.0150	95.2038	116.9475	666.
21	153.8371	61.5993	227.1254	699.9208		51	264.0150	52.1863	116.9475	709.3
22	163.0274	50.4516	217.9351	711.0685		52	292.1859	200.5774	88.7766	560.9
23	173.9952	37.1477	206.9673	724.3724		53	292.1859	152.9698	88.7766	608.
24	187.3076	21.0000	193.6549	740.5201		54	292.1859	105.3622	88.7766	656.
25	151.4569	90.3747	229.5056	671.1454		55	292.1859	57.7546	88.7766	703.
26	163.7468	75.4671	217.2157	686.0530		56	323.3626	169.2920	57.5999	592.
27	173.8101	63.2605	207.1524	698.2596		57	323.3626	116.6046	57.5999	644.
28	185.8199	48.6928	195.1426	712.8213		58				
29	200.3968	31.0111	180.5657	730.5090		59	352.1625	241.7497	28.8000	519.7
30	161.7015	107.9379	219.2610	653.5822		60	352.1625	184.3697	28.8000	577.

FORM 1544



380.9625 WING ANALYSIS CO-ORDINATE AXIS ORIGIN

SCALE : 1" = 300"

TABLE 1

* FOR (a) = 68 - 72 INCL. & 76 - 89 INCL.

X = 380.9625 - X_F

Y = 761.5201 - Y_F

AVRO AI…

MA…

ENG…

AIRCRAFT -

WEIGHT -

C. G. POSITION

8	9	10	11	12	13	14	15	16	17	18	19	20	21
Y	X_F = 380.9625 − X	Y_F = 761.5201 − Y		LOAD POINT (a)	X *	Y *	X_F = 380.9625 − X	Y_F = 761.5201 − Y		LOAD POINT (a)	X	Y	X_F = 380.9625 − X
91.5321	205.7360	669.9880		61	352.1625	126.9898	28.8000	634.5303		91	323.3626	63.3171	57.5999
78.0988	194.6614	683.4213		62	352.1625	69.6098	28.8000	691.9103					
62.0670	181.4447	699.4531		63	240.5576	235.6444	140.4049	525.8757					
42.6084	165.4029	718.9117		64	380.9625	261.5201	0	500.0000					
128.5068	207.2632	633.0133		65	380.9625	199.4476	0	562.0725					
110.3465	192.2317	651.1736		66	380.9625	137.3751	0	624.1450					
95.4764	180.0327	666.0437		67	380.9625	75.3026	0	686.2175					
77.7308	165.4022	683.7893		68	323.3626	19.0201	57.5999	742.5000					
149.0759	195.2653	612.4442		69	352.1625	19.0201	28.8000	742.5000					
129.1608	178.8473	632.3593		70	380.9625	19.0201	0	742.5000					
112.8532	165.4029	648.6669		71	323.3626	$\overline{1}0.9799$	57.5999	772.5000					
169.6448	183.2673	591.8753		72	380.9625	$\overline{1}0.9799$	0	772.5000					
147.9754	165.4029	613.5447		73	229.6928	241.1455	151.2697	520.3746					
163.7649	142.4023	597.7552		74	254.3909	276.5201	126.0716	485.0000					
124.8949	142.4023	636.6252		75	289.1267	342.8946	91.8358	418.6255					
86.0248	142.4023	675.4953		76	323.3626	276.5201	57.5999	485.0000					
47.1548	142.4023	714.3653		77	380.9625	276.5201	0	485.0000					
181.2389	116.9475	580.2812		78	323.3626	292.7701	57.5999	468.7500					
138.2214	116.9475	623.2987		79	368.3626	292.7701	12.5999	468.7500					
95.2038	116.9475	666.3163		80	323.3626	331.0201	57.5999	430.5000					
52.1863	116.9475	709.3338		81	368.3626	331.0201	12.5999	430.5000					
200.5774	88.7766	560.9427		82	323.3626	375.0201	57.5999	386.5000					
152.9698	88.7766	608.5503		83	368.3626	375.0201	12.5999	386.5000					
105.3622	88.7766	656.1579		84	323.3626	419.0201	57.5999	342.5000					
57.7546	88.7766	703.7655		85	368.3626	419.0201	12.5999	342.5000					
169.2920	57.5999	592.2281		86	323.3626	463.0201	57.5999	298.5000					
116.6046	57.5999	644.9155		87	368.3626	463.0201	12.5999	298.5000					
				88	368.3626	507.0201	12.5999	254.5000					
241.7497	28.8000	519.7704		89	368.3626	641.5201	12.5999	120.0000					
184.3697	28.8000	577.1504		90	323.3626	221.9794	57.5999	539.5407					

- 89 INCL.

AVRO AIRCRAFT LIMITED
MALTON, ONTARIO
ENGINEERING DIVISION

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

REPORT NO. - 7/0510/13
SHEET - - 12
DATE - - MAY 10/56
CHECKED BY R.H. SHEARLEY
PREPARED BY - R.H. SHEARLEY

17	18	19	20	21	22	23	24	25	26	27	28	29
	LOAD POINT (a)	X	Y	X_F = 380.9625 − X	Y_F = 761.5201 − Y							
	91	323.3626	63.9171	57.5999	697.6030							

REFERENCE

COMPONENT	LOAD POINTS (a)	SOURCE
WING	1-57 INCL., 59-62 INCL. 64-67 INCL., 90 & 91	REPORT 7/0510/7 FIG. 9
FRONT-SPAR BEAM	63 & 73	SHEET 11
TANK 3	74 & 75	REPORT 7/0510/8 FIG 1 (A-BACK OF REPORT)
CENTER FUSELAGE	76-89 INCL.	REPORT 7/0510/9 FIG. 2
REAR FUSELAGE	68-72 INCL.	REPORT 7/0510/12 FIG. 2

O. R. L.