STRESS REPORT

7/0510/13

<u>Issue 2</u>

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

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

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

ANALYZED

AIRCRAFT: C 105

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Date 30 Sept 96
Signature [signature]
Unit / Rank / Appointment AVRS

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 DATE

SUPERVISED BY [signature] DATE May 56

APPROVED BY DATE

ISSUE No.	REVISION No.	REVISED BY	APPROVED BY	DATE	REMARKS
2	1 & 5	R.Shearly	[signature]	Sept. 1956	Altered to handle any number of cases, with each case handled separately. Sheets revised 3 to 10.

FORM 1316A

AVRO AIRCRAFT LIMITED
MALTON - ONTARIO
TECHNICAL DEPARTMENT

REPORT NO. 7/0510/13

SHEET NO. 1

AIRCRAFT: C 105

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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

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.

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) * Sept/56

γ angular acceleration in pitching motion. (Positive nose down) *

* in reference to the system of coordinates

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

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 $\begin{Bmatrix} 28\% \ \bar{c} \\ 30\% \ \bar{c} \\ 32\% \ \bar{c} \end{Bmatrix}$.

Serr/52

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) $E = D/B$

FORM 1319A

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Case ※※ . ※※ (*)

(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.

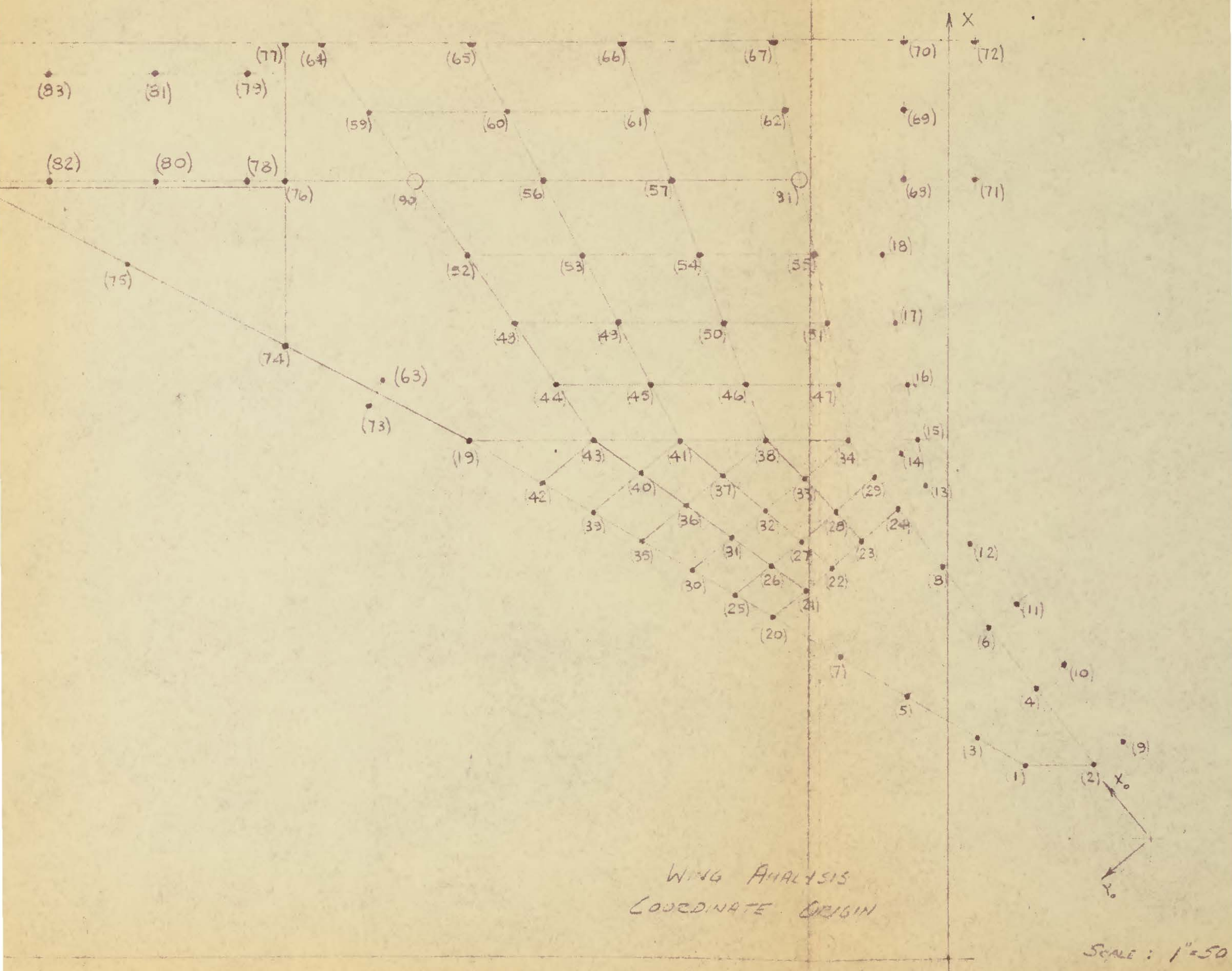
$$g = 386.4 \text{ in/sec}^2$$

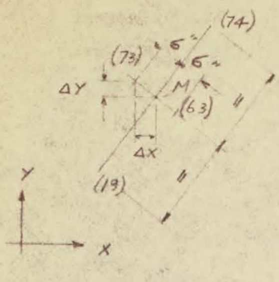
Range of values

$$-75 \text{ inches} < x_a < 650 \text{ inches}$$
$$46{,}000 \text{ lbs} < \text{Sum } W_a < 50{,}000 \text{ lbs}$$
$$-200{,}000 \text{ lbs} < \text{Sum } P_a < 400{,}000 \text{ lbs}$$
$$200 \text{ inches} < x_m < 225 \text{ inches}$$
$$275 \text{ inches} < x_g < 300 \text{ inches}$$
$$50 \text{ inches} < x_p < 512 \text{ inches}$$

FORM 1319 A

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





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

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

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

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 = $\sqrt{39.3313^2 + 76.2503^2} =$

$$= 85.7966165$$

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

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

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

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

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	63.9171	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	165.4029	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	118.8473	632.3593		70	380.9625	19.0201	0	742.5000					
112.8532	165.4029	648.6669		71	323.3626	-10.9799	57.5999	772.5000					
169.6448	183.2673	591.8753		72	380.9625	-10.9799	0	772.5000					
147.9754	165.4029	613.5447		73	229.8928	241.1455	151.0697	520.3746					
163.7649	142.4023	597.7552		74	254.3909	276.5201	126.5716	485.0000					
124.8949	142.4023	636.6252		75	289.1267	342.8946	91.8358	418.6255					
36.0248	142.4023	675.4953		76	323.3626	276.5201	57.5999	485.0000					
47.1549	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					
152.9638	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					