AVRO AIRCRAFT

TECHNICAL REPORT

ANALYZED

A. V ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C 105

REPORT NO: 7/0510/17 Vol I

FILE NO

NO. OF SHEETS:

Sect	0	13 + 5
	1	63
	2	17
	3	30
	4	32
	5	122
	6	3
	7	54
	12	FIGURES
	6	MATRIX ZONE

TITLE:

Wing Analysis

CONFIDENTIAL

Classification cancelled / Changed to UNCLASS
By authority of AVRS
Date 30 Sept 96
Signature
Unit / Rank / Appointment AVRSS

PREPARED BY R.N. Shearly, D.L. Turner, V.F. Gardener DATE June 55

CHECKED BY R.N.S., D.L.T., V.F.G. DATE July '55

SUPERVISED BY [signature] DATE

APPROVED BY DATE

ISSUE No	REVISION No	REVISED BY	APPROVED BY	DATE	REMARKS
2	84	R.N. SHEARLY		MAY 1956	SHEET 0-2 REVISED SHEETS 8-1 TO 8-21 INCLUSIVE ADDED.
					15865756

FORM 1316A

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

AIRCRAFT: C105	Wing Analysis	REPORT NO. 7/0510/7 SHEET NO. 0-1	
		PREPARED BY: D. Turner	DATE: 22 June 55
		CHECKED BY	DATE

Index

CONFIDENTIAL

Reference Reports	0-3
Index of Figures	0-4
Index of Matrices	0-5
Introduction	0-6
Explanatory Notes	0-9
Computation Program	1
Geometry	1-1
Wing Tip	1-2
Outer Wing	1-21
Inner Wing	1-24
Centre Section	1-32
Revision	1-47
Transformation of Coordinates – O/W to I/W	1-57
Wing Tip	2-1
Elastic Influence Coefficients	2-2
Statically Determinate Stresses	2-9
Loads at Rib 18	2-15
Elastic Influence Coefficients	3-1
Outer Wing	3-2
Inner Wing	3-6
Centre Section	3-9
Revision	3-15
Redundant Stress Coefficients	4-1
Outer Wing	4-3
Transport Joint	4-5
Inner Wing	4-22
Centre Section	4-24
Statically Determinate Stresses	5-0
Introduction to Technique	5-1
G_1 Wing Tip – Outer Wing Interaction	5-8
T_2 Outer Wing Stress – O/W Load	5-11
T_1' Outer Wing Stress – Wing Tip Loads	5-24
H_1 Inner Wing Interaction – Outer Wing Interaction	5-43

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

REPORT NO. 7/0510/7
SHEET NO. 0-2

AIRCRAFT: C105 | Wing Analysis

PREPARED BY: [signature] | DATE: 22 June '55
CHECKED BY: | DATE:

Index (Cont'd)

Statically Determinate Stresses (cont'd)

G_2	Inner Wing Interaction - Outer Wing Loads	5-45
T_2'	Inner Wing Stress - Inner Wing Interaction	5-52
T_3	Inner Wing Stress - Inner Wing Load	5-62
G_3	Inner Wing - Centre Section Interaction	5-69
H_2	Centre Section Interaction - Inner Wing Loads	5-71
T_3'	Centre Section Stress - Centre Section Interaction	5-74
T_4	Centre Section Stress - Centre Section Loads.	5-83
	Intercomponent Interaction Loads	5-98

T_{ia} & Kip Factors to Account for Geometry Changes. 6-1

Miscellaneous Calculations 7-1

Boundary Conditions (T/J - ℄) 7-2

Wing Tip Energy Expressions 7-21

Checking Techniques 7-32

Errors 7-47

Figures 8-1

Figures 1-13 8-1

Matrices C_{ik} Zones 1 & 3 8-14

Kip Zones 1 & 3 8-16

T_{ia} Zones 1 & 2 8-18

Check Figures 1 & 2 8-20

FORM 1319A

A.V.ROE CANADA LIMITED MALTON – ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7/0510/7 SHEET NO. 0-4	
		PREPARED BY: V F GARDENER	DATE: FEB 16 '55
AIRCRAFT C105	Index of Figures	CHECKED BY:	DATE:

INDEX OF FIGURES

FIG 1. NUMBERING CODE (Preliminary)

FIG 2 WING CO-ORDINATE SYSTEM (Preliminary)

FIG 3 OUTER WING & TIP DATA DRAWING

FIG 4 WING TIP DATA DRG & SECTION PROPERTIES.

FIG 5 OUTER WING SECTION PROPERTIES

FIG 6 INNER WING SECTION PROPERTIES (PRELIMINARY AREAS & THICKNESSES)

FIG 7 CENTRE SECTION WING & FUSELAGE SECTION PROPERTIES (PRELIMINARY AREAS & TH.)

FIG 8 Revised Numbering Code

FIG 9. Wing Co-ordinate System (Revised)

FIG 10. INNER WING & CENTER SECTION WING SECTION PROPERTIES (Revised)

Check Fig. 1 Outer Wing Tia check.

CHECK FIG. 2 INNER WING & CENTER SECTION WING TIA CHECK

Note: Figures 4, 5, 8, 9 & 10 only are required for the interpretation of the results of this analysis.

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C105	Reference Reports	REPORT NO. 7/0510/7	SHEET NO. 0-3
		PREPARED BY: D L Turner	DATE: 5 May '55
		CHECKED BY:	DATE:

Reference Reports.

Gen/1030/307	An Introduction to Oblique Co-ordinates
Gen/1030/329	Stress Analysis of Multi-Spar & Multi-Rib Wing Structures — Part 1
Gen/1030/330	Part 2
Gen/1030/331	Part 3
Gen/1030/332	Part 4
7/0510/2	Wing Analysis
7/0510/4	Incorporating The L/E Structure into the Wing Analysis
Gen/1030/311	Theoretical Investigation of the C100 Wing.
P/Geom/32	C105 Geometry.
7/0556/27	Analysis of Bulkhead at Sta. 485.
7/0510/7 Vol. II	Computation of Preliminary Matrices
7/0510/7 Vol. III	Wing Analysis — Results.

FORM 1319 A

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

AIRCRAFT:		REPORT NO. 7/0510/7	
C105	Wing Analysis	SHEET NO. 0-5	
		PREPARED BY	DATE
		D L Turner	22 June '55
		CHECKED BY	DATE

Index of Matrices

Preliminary Matrices

Fig P-1 Matrices H_1, H_2, G_2 & G_3

P-2 T_2, T_1'

P-3 T_3, T_2'

P-4 T_4, T_3'

P-5 G_1

Primary Matrices

C_{ix}

Zone 1 Wing Tip & Outer Wing

Zone 3 Inner Wing & Centre Section

T_{ia}

Zone 1 Wing Tip & Outer Wing

Zone 2 Inner Wing & Centre Section

Kip

Zone 1 Outer Wing

Zone 3 Inner Wing & Centre Section

ORM 1319A

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT C105	Wing Analysis	REPORT NO. 7/0510/7	SHEET NO. 0-6
		PREPARED BY	DATE
		K.L. Turner	22 June 55
		CHECKED BY	DATE

Introduction

In this report the wing of the C105 aircraft is analyzed, and the stresses and deflections at the intersections of a network of ribs and spars are calculated. The method used is that developed in reports Gen/1030/325-332 inclusive. These reports should be referred to when making use of this report.

The wing, having a low aspect ratio delta planform of low thickness-chord ratio, with relatively thick skins, is considered to behave as a plate, (a two dimensional problem), rather than as a beam, (a one dimensional problem), as is customary in dealing with straight or swept wings having relatively little taper.

The method as developed in the above mentioned reports, is an adaptation of the Southwell formulation of plate theory, extended, by means of a trapezoidal coordinate system, to take into account the spanwise taper (in thickness) of the wing, and the fanline arrangement of spars with parallel ribs. Using this method, a net, approximating the actual locations of ribs and spars, can be formed, loads can be applied at the intersection points, and deflections and rib and spar stresses can be determined at these points.

In brief, the method is as follows: first an approximate plate moment distribution (based on unit loads at the intersection points) is assumed, which satisfies equilibrium conditions exactly, but which does not necessarily produce compatible strains. This distribution is then corrected by a series of properly chosen self-equilibrating groups whose magnitude is determined by the minimum energy theorem. Stresses may then be determined, and, if required, deflections may be calculated using a dummy load method.

As suggested in the reports developing the analysis, matrix methods are used throughout, the high speed automatic computer doing the major proportion of the work. The body of this report consists primarily of the formation of the required matrices, (Elastic Influence Matrix, Stress-Redundant Load Matrix, and Statically Determinate Stress-Load Matrix) The actual calculation of the deflections and stresses is done by the high speed computer.

For convenience, the matrices are formed using unit loads at the assumed load points. Thus a unit load-deflection matrix, and a unit load-stress matrix are determined by the machine. These matrices, when multiplied by the load matrix for any case, give the required deflections and stresses.

The analysis, as carried out in this report, is designed to be used in combination with the results of reports; 7/0510/8 — Tank 3 Analysis, 7/0510/9 - Centre Fuselage Analysis, 7/0556/27 - Analysis of Bulkhead at Sta. 485, 7/0510/ — Rear Fuselage Analysis, and the analysis of the front spar beam. The result of this combined analysis brings in the effect of each of the components of the aircraft on all other components, thus presenting a complete picture of the stress and deflection distribution throughout the aircraft. The individual analyses, of which this report is one, present only the effects on the particular component of the loads applied to that component.

To make provision for combining the results of this analysis with the analyses of other components, a set of unit "interaction" loads has been incorporated at appropriate locations, to allow the wing structure to be coupled with the remaining structure of the aircraft. These forces are treated in the same way as the basic unit loads applied to the structure. Details of the methods used in developing the analysis of the complete aircraft are given in report

A. V. ROE CANADA LIMITED MALTON - ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7/0510/7 SHEET NO. 0-8	
		PREPARED BY	DATE
AIRCRAFT: C105	Wing Analysis	DL Turner	22 June '55
		CHECKED BY	DATE

For the purposes of this analysis, the effective wing structure is considered to consist of the structure between the front and rear spars inboard of rib 18 for the outer wing, and the structure between the main and rear spars for the inner wing and centre section. The structure between the main and front spars of the inner wing is considered ineffective due to the large wheel well cut-outs, however the fuselage side rib and the ℄ rib between the main spar and bulkhead 485 are included in the wing structure.

The wing tip (outboard of rib 18) is included in the analysis. However the trapezoidal plate theory, as used for the remaining wing structure, is not used here, as it is felt that the conventional engineers' theory of bending and torsion may be applied to this region with sufficient accuracy. (This is borne out by the results of report 7/0510/2). Accordingly this has been done.

In this report the symmetric cases only are considered, hence the calculations are made for one side of the structure only, a factor of two, where required, serving to account for the complete structure.

FORM 1319A

A. V. ROE CANADA LIMITED MALTON - ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7/0510/7 SHEET NO. 0-9	
AIRCRAFT: C105	Wing Analysis	PREPARED BY	DATE
		D. Turner	22 June '55
		CHECKED BY	DATE

Explanatory Notes.

Initially this analysis was intended to include in it the front spar beam, the rear fuselage, (including fin effect) bulkhead 485, and through the results of report 7/0510/8, the effect of tank 3. Much of the calculation was done on this basis. Then, due to delay in obtaining information concerning the rear fuselage, it was decided to reduce the scope of this report to cover the wing proper only, and to set it up as a partial problem, the results to be combined with the results of separate analyses of the centre fuselage, rear fuselage, (with fin effect), bulkhead 485, tank 3 and the front spar beam. As a result, those portions of the report, relating to components other than the wing, which had been completed, were scrapped. However, in perusing the report, it will be noticed that, due to the interlocking of components, occasional reference to some of this work may appear. This should not detract from the general continuity of the work.

It should be noted also, that due to some changes in skin thickness and rib & spar cap areas at a late date, the completed Tia & Kip matrices (based on the original data) have been factored where necessary, and some elements of the Cik matrix have been recalculated. Thus it may be said that some changes in the structure were made after the matrices were completed, and that corrections have been made according to these changes.

Since much of the geometry of the wing remains the same as that used in report 7/0510/2, many elements in the matrices were obtained by simple factoring of the data calculated in report 7/0510/2. Hence it is imperative when using this report to have report 7/0510/2 available for reference.

FORM 1219A

A. V. ROE CANADA LIMITED MALTON · ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7/0510/7 SHEET NO. 0-10	
AIRCRAFT: C105	Wing Analysis	PREPARED BY D L Turner	DATE 22 June '55
		CHECKED BY	DATE

Explanatory Notes (Cont'd)

For the purposes of this analysis, the complete wing has been considered as pinned at four points, on the fuselage side at the main and rear spars. These points serve as the reaction points for all applied loads.

The major proportion of the elements of the T_{ia} matrix have been determined by use of the I.B.M computer (ref. sect. 5). Results of this calculation appear in report 7/0510/7 Vol II "Computation of Preliminary Matrices - Wing Analysis" and, of course, in the T_{ia} matrix.

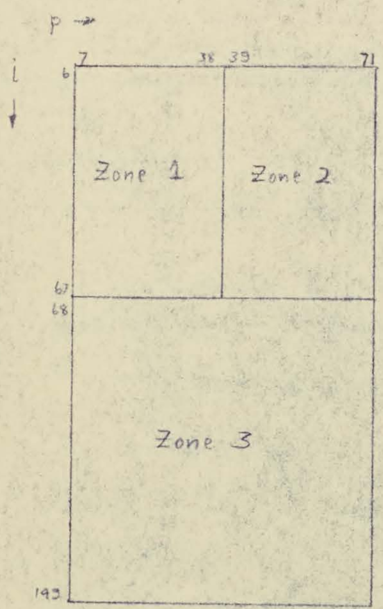
The results of the operations carried out on the primary matrices of this report (T_{ia}, K_{ep} & C_{in}) appear in report 7/0510/7 Vol III and its associated drawings.

FORM 1319

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

AIRCRAFT: C105	Wing Analysis	REPORT NO. 7/0510/7	SHEET NO. 0-12
		PREPARED BY D L Turner	DATE 22 June '55
		CHECKED BY	DATE

Key to Kip Matrix



Non-zero Elements

Zone 1	104
Zone 2	0
Zone 3	196
	300.

Total Elements 9360

plus checks

FORM 1318 A

COMPUTATION PROGRAM

A) Given Matrices:

C_{ik} elastic influence coefficients (symmetric)

K_{ip} stress to redundant moment

T_{ia} stat. det. stress to load

δ_{ik} unit matrix (Kronecker Delta)

The indices of these matrices are given by two sets of numbers, the set nearer the body of the matrix is a preliminary set which is retained for refering to the calculations of the non zero elements, the other set (the circled set) will be the indices referred to in the following pages dealing with the computation program.

These indices have the range and are defined, as follows:

1) loading point indices

a, b = 1, 2, 3, ... 79

2) stress section indices

i, k = 1, 2, 3, ... 138

3) redundant moment indices

p, q, r = 1, 2, 3, ... 62

Indices of the same group are interchangeable, the first index denoting the row, the second index the column of the matrix.

The matrix elements are given in 6 separate sheets, to be fitted together according to the key below.

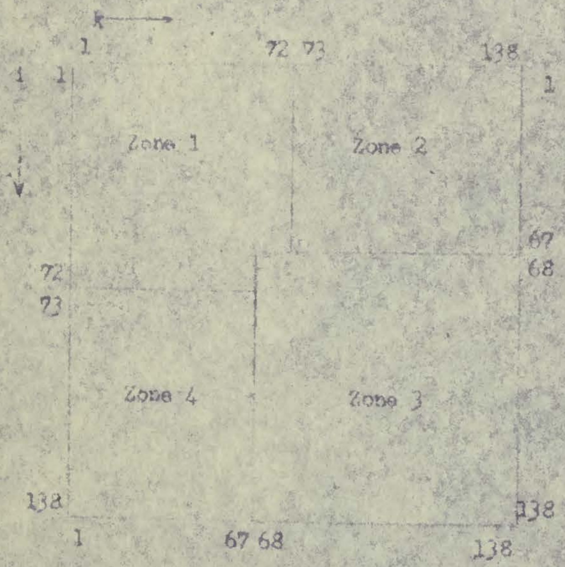
The C_{ik} matrix is divided into 2 sheets. This matrix is made up of 19,044 elements, 878 being non zero elements.

The K_{ip} matrix is divided into 2 sheets. The matrix is made up of 8,556 elements, 300 being non zero elements.

The T_{ik} matrix is divided into 2 sheets. The matrix is made up of 10,902 elements, 2,677 being non zero elements.

Check sums are provided for each matrix. They are either partial sums as indicated in each sheet or total sums containing all elements of the corresponding row or column.

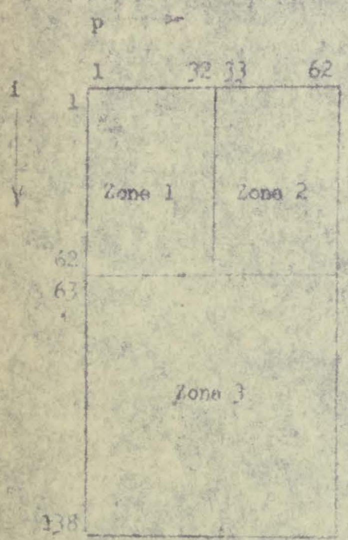
Key to the C_{ik} matrix



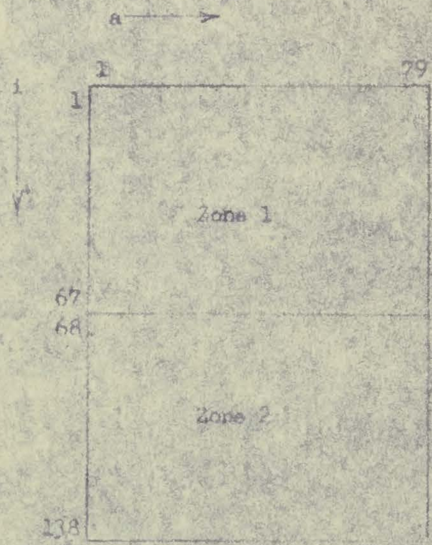
Non zero elements:	Zone 1	413
	Zone 2	0
	Zone 3	465
	Zone 4	0
	Total	878

Elements total 19044

Key to the K_{ip} matrix.



Key to the T_{ia} matrix.



Non zero elements

Zone 1	104		Zone 1	607
Zone 2	0		Zone 2	2070
Zone 3	196			
Total	300		Total	2677

Total elements

8556   10902

B) Elementary Operations:

Transpose to A_{ip} is A_{pi}

Matrix product of A_{ak} and B_{ki} is C_{ai}, denoted as

$$A_{ak} B_{ki} = C_{ai}$$

Inverse to D_{pq} is D^{-1}_{qp}

C) Results to be printed:

1) Results:

Redundancies to Unit Load	F_{pa}
Stress to Unit Load	S_{ia}
Displacement to Unit Load	Z_{ab}

2) Checks:

Inversion error	$D_{pr} D^{-1}_{rq} - \delta_{pq}$
Symmetry of D_{pq}	
Symmetry of Z_{ab}	

D) Operations:

1) Multiply	$H_{ip} = C_{ik} K_{kp}$	
2) Multiply	$D_{pq} = K_{pi} H_{iq}$	print
3) Invert	$D_{pq} \rightarrow D^{-1}_{qp}$	
4) Perform	$D_{pr} D^{-1}_{rq} - \delta_{pq}$	print

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT		REPORT NO. 7/0510/7 Extract	
		SHEET NO. 5	
	WING ANALYSIS	PREPARED BY	DATE
		R.N.Shearly	July 8/55
		CHECKED BY	DATE
		J.G.	

5) Multiply $M_{pi} = D^{-1}_{pq} H_{qi}$

6) Multiply $F_{pa} = -M_{pi} T_{ia}$ print

7) Multiply $N_{ia} = K_{ip} F_{pa}$

8) Add $S_{ia} = T_{ia} + N_{ia}$ print

9) Multiply $O_{ai} = T_{ak} C_{ki}$

10) Multiply $Z_{ab} = O_{ai} S_{ib}$ print

The above computation will be checked:

a) by applying check sums in the conventional way,

b) by symmetry of D_{pq} in most significant figures,

c) by inversion error of D^{-1}_{qp},

d) by symmetry of Z_{ab} in four significant figures,

e) by analysis of the final result.

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT C 105	Wing Tip	REPORT NO 7/0510/7	SHEET NO 1-2 *
		PREPARED BY	DATE
		[signature]	16 May '55
		CHECKED BY	DATE
		R.N. Sh[illegible]	[illegible] /55

Wing Tip Geometry

Values of I_i as calculated from data given in fig 5 are as follow: (ref sht. 1-3)

$I_{l/s} = 16.1830$ in^4 $I_{Rfr/s} = 79.3369$ in^4

$I_{M/s} = 59.8217$ in^4 $I_{R/s} = 37.9600$ in^4

$I_{Ffr/s} = 71.7220$ in^4

$$* \quad Y_0 = \frac{\Sigma Y_i I_i \cos^2\theta_i}{\Sigma I_i \cos^2\theta_i}$$

where I_i is the moment of inertia of the relevant spar at rib 18

This equation is solved in the table on sht. 1-3

Note: A small error is introduced here in that the focus of the wing tip trapezoid is different from that of the outer wing. Also, the above values of I_i are taken on right cross sections of outer wing spars at rib 18, rather than on cross sections perpendicular to the wing tip trapezoid continuation of the spars. These effects are neglected.

For reference, see fig. 4 and

* This equation determines the location of the shear centre at rib 18 (ref.

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

REPORT NO.	7/0510/2
SHEET NO.	1-5
PREPARED BY	DATE
D.L. Turner	10 Jan. '55
CHECKED BY	DATE
R.N. SHEPPEY R.N.S.	MAY 6/55

AIRCRAFT: C105

Wing Tip

Wing Tip Geometry.

X_o ordinates.

From report 7/0510/2 sht. 1-15

$$X_{o_2} = 173.3489$$

$$X_{o_1} = 156.8819$$

From Fig. 3

$$X_{o_E} = X_{o_1} - 14.50$$
$$= 156.8819 - 14.50$$
$$= 142.3819 \text{ in.}$$

$$X_{o_D} = X_{o_E} - 32.00$$
$$= 142.3819 - 32.00$$
$$= 110.3819$$

$$X_{o_C} = X_{o_D} - 32.00$$
$$= 110.3819 - 32.00$$
$$= 78.3819$$

To obtain the X_o ordinates at the wing tip, the following equation is used:

$$X_{o_t} = X_o' \cos\alpha_o + Y_o' \sin\alpha_o \qquad \text{(ref. Rep. 7/0510/2 sht 1-13)}$$

where $X_o' = X_{o_1}' = X_{o_2}' = 166.0304 - 135.3296 = 30.7008$
(wing tip X_o' ordinate, ref 7/0510/2 sht. 1-9)

$$\therefore X_{o_t} = 30.7008 \times .77163 + Y_o' \times .63608$$
$$= 23.6897 + .63608\, Y_o'$$

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

AIRCRAFT: C105	Wing Tip	REPORT NO. 7/0510/7	
		SHEET NO. 1-7	
		PREPARED BY	DATE
		D L Turner	11 Jan. '55
		CHECKED BY	DATE
		R.N. SHEARLY R.N.	MAY 7/55

Wing Tip Geometry

Y_o ordinates

From report 7/0510/2 sht. 1-13

$$Y_o = X_o \tan\theta$$

Hence $Y_{o(2)} = 38.6262 \tan\theta_1$

$= -38.6262 \times .03653$ (ref. 7/0510/2 sht. 1-19)

$= -1.4110$

$Y_{o(1)} = 57.1701 \tan\theta_s$

$= 57.1701 \times .36877$

$= 21.0826$

$Y_{o(3)} = 78.3819 \times .36877 = 28.9049$

$Y_{o(4)} = -78.3819 \times .03653 = -2.8633$

$Y_{o(5)} = 110.3819 \times .36877 = 40.7055$

$Y_{o(6)} = -110.3819 \times .03653 = -4.0323$

$Y_{o(7)} = 142.3819 \times .36877 = 52.5062$

$Y_{o(8)} = -142.3819 \times .03653 = -5.2012$

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

AIRCRAFT: C105	Wing Tip	REPORT No. 7/0510/7	SHEET No. 1-8
		PREPARED BY: D. Turner	DATE: 11 Jan. 55
		CHECKED BY: R. N. SHEARLY	DATE: MAY 6/55

Wing Tip Geometry

$$Y_{o(9)} = Y_{o(2)} - 15.00 = -1.4110 - 15.00 = -16.4110$$

$$Y_{o(10)} = Y_{o(4)} - 15.00 = -2.8633 - 15.00 = -17.8633$$

$$Y_{o(11)} = Y_{o(6)} - 15.00 = -4.0323 - 15.00 = -19.0323$$

$$Y_{o(12)} = Y_{o(8)} - 15.00 = -5.2012 - 15.00 = -20.2012$$

Summarizing X_o & Y_o ordinates for wing tip load points:

load point	X_o	Y_o
1	57.1701	21.0826
2	38.6262	-1.4110
3	78.3819	28.9049
4	78.3819	-2.8633
5	110.3819	40.7055
6	110.3819	-4.0323
7	142.3819	52.5062
8	142.3819	-5.2012
9	38.6262	-16.4110
10	78.3819	-17.8633
11	110.3819	-19.0323
12	142.3819	-20.2012

A.V.ROE CANADA LIMITED MALTON – ONTARIO TECHNICAL DEPARTMENT (Aircraft)		REPORT NO. 7/0510/7 SHEET NO. 1-9	
		PREPARED BY: D L Turner	DATE: 11 Jan. '55
AIRCRAFT C105	Wing Tip	CHECKED BY: R. N. SHEARLY RNS	DATE: MAY 6/55

Wing Tip Geometry.

X_o & Y_o ordinates of the locus of shear centres.

The locus of shear centres has been located by calculating the Y_o ordinate at rib 18 (ref. sht. 1-2 et seq) and joining this point to the intersection of the front spar ℄ and the aileron hinge line.

To locate this intersection geometrically, determine the equations of the front spar and of the aileron hinge line, from which the intersection point can be determined.

Having this information, and the coordinates of the shear centre at rib 18, the equation of the locus of shear centres can be written, from which the required ordinates may be calculated.

This work is done below.

Equation of aileron hinge line.

At $X_o' = 166.0304 - 15.0366 = 150.9938$
$Y_o' = 608.903 - 512.6667 = 96.2364$ } i/b end of aileron

At $X_o' = 30.7008$
$Y_o' = 25.3111$ } o/b end of aileron

See data drawing. (Fig. 4)

Now from Rep. 7/0510/2 sht 1-13

$$X_o = X_o' \cos \delta_o + Y_o' \sin \delta_o$$
$$= .77163\, X_o' + .63608\, Y_o'$$

A.V.ROE CANADA LIMITED
MALTON – ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT C105	Wing Tip	REPORT NO. 7/0510/7 SHEET NO. 1-10	
		PREPARED BY: B L Turner	DATE: 12 Jan. '55
		CHECKED BY: R. N. SHERRY RNS.	DATE: MAY 6/55

Wing Tip Geometry.

and $Y_0 = -X_0' \sin\delta_0 + Y_0' \cos\delta_0$
$= -.63608\, X_0' + .77163\, Y_0'$ follows.

hence at the i/b end of the aileron

$X_0 = 150.9938 \times .77163 + 96.2364 \times .63608$
$= 116.5113 + 61.2140$
$= 177.7253$

$Y_0 = -150.9938 \times .63608 + 96.2364 \times .77163$
$= -96.0441 + 74.2589$
$= -21.7852$

and at the o/b end of the aileron

$X_0 = 30.7008 \times .77163 + 25.3111 \times .63608$
$= 23.6897 + 16.0999$
$= 39.7896$

$Y_0 = -30.7008 \times .63608 + 25.3111 \times .77163$
$= 0$

A.V.ROE CANADA LIMITED MALTON – ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7/0510/7 SHEET NO. 1-11	
		PREPARED BY: D.L. Turner	DATE: 12 Jan. '55
AIRCRAFT C105	Wing Tip	CHECKED BY: R.N. Shearly	DATE: May 6/55

Wing Tip Geometry

From this data the equation of the aileron hinge line may be written:

$$Y_0 + 21.7852 = \frac{21.7852}{39.7896 - 177.7253}(X_0 - 177.7253)$$

$$Y_0 = -\frac{21.7852}{137.9357}(X_0 - 177.7253) - 21.7852$$

$$= -.1579\,X_0 + 6.2776$$

The equation of the front spar may be written from the following information:

At the intersection of F/S & T/J rib:

$X_0 = 309.1772$ ref Rep. 7/0510/2 shts 1-9

$Y_0 = 114.0153$ 1-15 & 1-16

At $X_0 = 0$ $Y_0 = 0$

hence $Y_0 = \frac{114.0153}{309.1772} X_0 = .3688\, X_0$

A.V.ROE CANADA LIMITED
MALTON — ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/7
SHEET NO. 1-12

AIRCRAFT C105	Wing Tip	PREPARED BY: D.T. Turner	DATE: 12 Jan. '55
		CHECKED BY: R.N. SHEARLY	DATE: May 6/55

Wing Tip Geometry.

The location of the intersection of the front spar ℄ and the aileron hinge line may now be determined.

$Y_0 = .3688\ X_0$ F/S

$Y_0 = -.1579\ X_0 + 6.2776$ ail. hinge line.

$.3688\ X_0 + .1579\ X_0 = 6.2776$

$$X_0 = \frac{6.2776}{.5267} = 11.9187$$

$$X_0 = \frac{Y_0}{.3688} = 2.7128\ Y_0$$

$$X_0 = -\frac{Y_0}{.1579} + \frac{6.2776}{.1579} = -6.3331\ Y_0 + 39.7568$$

$2.7128\ Y_0 + 6.3331\ Y_0 = 39.7568$

$$Y_0 = \frac{39.7568}{9.0459} = 4.3950$$

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

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

AIRCRAFT		PREPARED BY	DATE
C105	Wing Tip	[signature]	16 May 55
		CHECKED BY	DATE
		R. N. Sheehy	MAY 20/55

Wing Tip Geometry

To determine the equation of the locus of shear centres.

At rib 18

$X_0 = 173.3489$ sht 1-5
$Y_0 = 25.5202$ sht 1-3

At intersection of aileron hinge line & front spar.

$X_0 = 11.9187$
$Y_0 = 4.3950$

Hence

$$Y_0 - 25.5202 = \frac{4.3950 - 25.5202}{11.9187 - 173.3489}(X_0 - 173.3489)$$

from which

$$Y_0 = .130863\,X_0 + 2.8353$$

The X_0 ordinates of the ribs are known, hence the Y_0 ordinates of the shear centre locus - rib intersections may be calculated from the above.
This calculation is done in the table on sht. 1-14.

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/7
SHEET NO. 1-14 *

PREPARED BY	DATE
D.L. Turner	16 May '55
CHECKED BY	DATE
R. N. Sheasby	MAY 20/55

AIRCRAFT: C105

Wing Tip

Wing Tip Geometry.

(1)	(2)	(3)	(4)
X loc'n	X_0	$.130863\,X_0$	Y_S $2.8353 + (3)$
X_{0_2}	173.3489	22.6850	25.5203
X_{0_E}	142.3819	18.6325	21.4678
X_{0_D}	110.3819	14.4449	17.2802
X_{0_C}	78.3819	10.2573	13.0926

FORM 1373A

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C105	Wing Tip	REPORT No 7/0510/7	SHEET No 1-17 *
		PREPARED BY [signature]	DATE 16 May 55
		CHECKED BY [signature]	DATE May 20/55

Wing Tip Geometry

Determination of b_r, (distance from F/s to R/s at rib 18), h_r, (wing thickness at shear centre at rib 18), & A_r (enclosed area of wing at rib 18)

At load points 22 & 23 h values are 6.4481 & 6.2826 respectively. (ref Rep. 7/0510/2 fig. 4)

Assume h varies linearly between points 22 & 23 (in agreement with the basic hypotheses of the analysis)

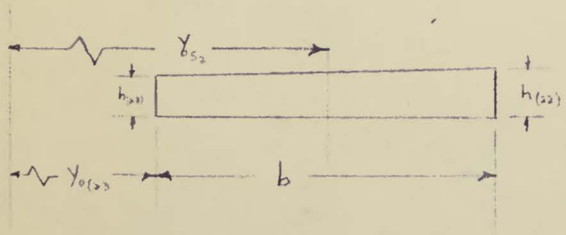
$$\Delta h = 6.4481 - 6.2826 = .1655 \text{ in.}$$

b = 17.2413 in (Ref rep. 7/0510/2 fig. 4) is the ribwise dist. between points 22 & 23.

$Y_{o(23)} = 14.5942$ (rep. 7/0510/2 fig. 4)

$Y_{oS_2} = 25.5203$ sht. 1-14

$Y_{oS_2} - Y_{o(23)} = 25.5203 - 14.5942 = 10.9261$



Then $h_r = 6.2826 + \frac{10.9261}{17.2413} \times .1655 = 6.3875$ in.

FORM 1018A

A.V.ROE CANADA LIMITED MALTON – ONTARIO TECHNICAL DEPARTMENT (Aircraft)		REPORT NO. 7/0510/2 SHEET NO. 1-18	
		PREPARED BY: D L Turner	DATE: 14 Jan. 55
AIRCRAFT C105	Wing Tip	CHECKED BY: R.N. SHEARLY RNS.	DATE: MAY 10/55

Wing Tip Geometry

From Rep 7/0510/2 fig 4

$b_r = 20.9266 + 17.2913 + 14.4469 + 17.6435$

$= 70.2583$ in.

A_r is considered as made up of a series of trapezoidal areas A_i where

$$A_i = b_{cell}\left(\frac{h_1 + h_2}{2}\right)$$ where h_1 & h_2 are h values at the two ends of the cell.

$$A_r = \sum A_i = 20.9266\ \frac{5.3324 + 6.2826}{2} + 17.2413\ \frac{6.2826 + 6.4481}{2}$$

$$+ 14.4469\ \frac{6.4481 + 5.9584}{2} + 17.6435\ \frac{5.9584 + 3.7033}{2}$$

$A_r = 121.5312 + 109.7469 + 89.6177 + 85.2331$

$= 406.1289$

AVRO AIRCRAFT LIMITED TECHNICAL DEPARTMENT (Aircraft)		REPORT NO. 7/0510/7 SHEET NO. 1-19 *	
		PREPARED BY	DATE
AIRCRAFT C105	Wing Tip	D L Turner	16 May 55
		CHECKED BY	DATE
		R N Alcock	MAY 20/55

Wing Tip Geometry

Calculation of t values

From Report 7/0510/2 sht. 1-28

$t = .00008821\, Y_o + .0004691\, X_o + .0774$

t will be determined by calculating t values along the locus of shear centres, and averaging values at each end of each bay.

hence: $t_1 = (t_{X_{st}} + t_C)\frac{1}{2}$ $\qquad t_2 = (t_C + t_D)\frac{1}{2}$

$t_3 = (t_D + t_E)\frac{1}{2}$ $\qquad t_4 = (t_E + t_{X_2})\frac{1}{2}$.

	(1) Y_o	(2) X_o	(3) $.00008821\, Y_o$	(4) $.0004641\, X_o$	(5) $.0774 + (3) + (4)$	(6) t_{av}
$t_{X_{st}}$	9.0148	47.2210	.0007952	.0219153	.1001	
						.1075
t_C	13.0326	78.3819	.0011543	.0363770	.1143	
						.1226
t_D	17.2802	110.3819	.0015243	.0512282	.1302	
						.1378
t_E	21.4678	142.3819	.0018937	.0660794	.1454	
						.1528
t_{X_2}	25.5203	173.3489	.0022511	.0804512	.1601	

A.V.ROE CANADA LIMITED MALTON – ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7/0510/7 SHEET NO. 1-21	
		PREPARED BY: VF GARDENER	DATE: JAN 25 '55
AIRCRAFT C105	OUTER WING	CHECKED BY: [signature] RNJ	DATE: 4 MAR 55

THE OUTER WING ASSUMED STRUCTURE IS BASICALLY THE SAME AS FOR REPORT 7/0510/2.

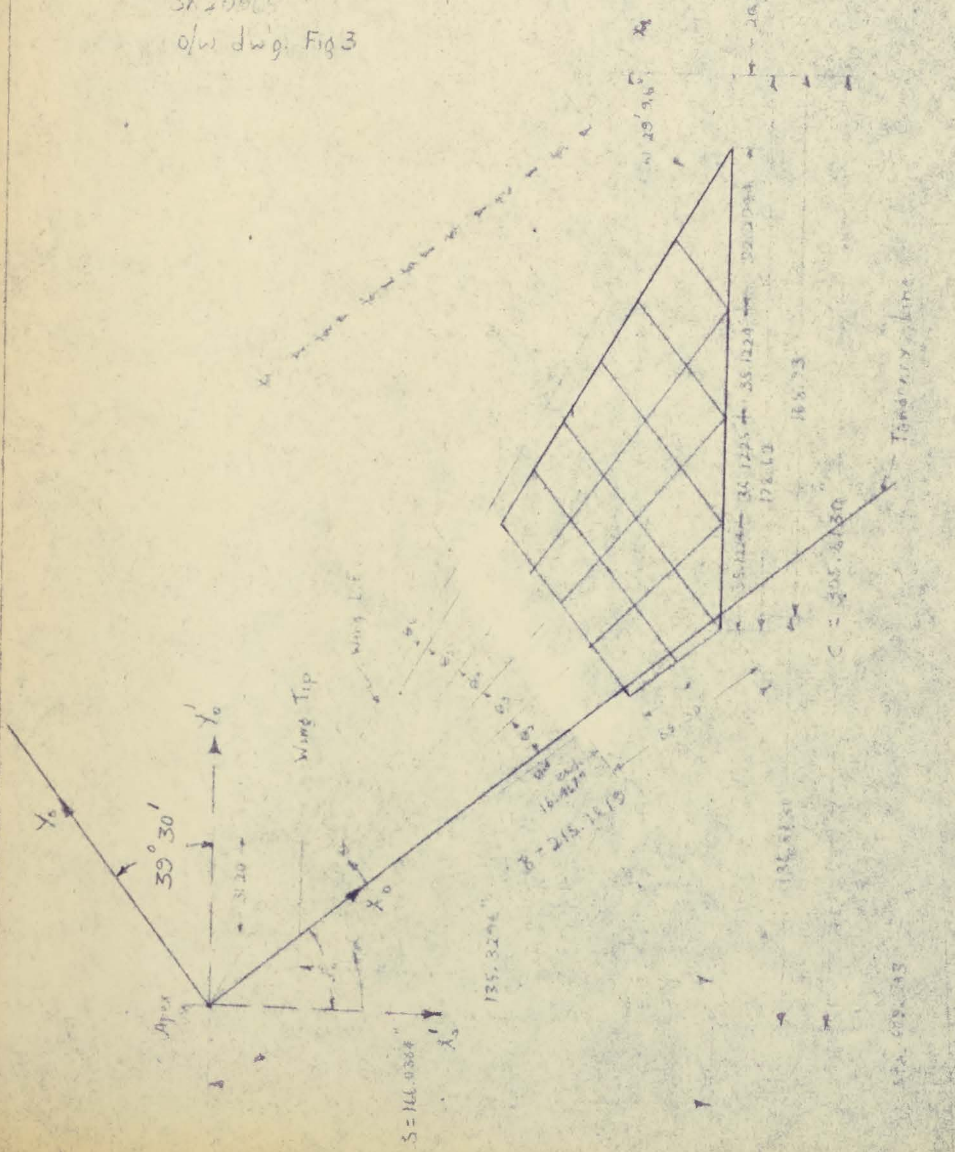
THE DIFFERENCES ARE THAT NOW THE FRONT SPAR IS EXTENDED INBOARD TO THE TRANSPORT JOINT AND THE STRUCTURE BETWEEN X_{01} AND X_{02} IS INCLUDED IN THE STATICALLY DETERMINATE WING TIP.

A. V. ROE CANADA LIMITED MALTON - ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7/0510/7 SHEET NO. 1-22	
		PREPARED BY	DATE
AIRCRAFT C105	Outer Wing	D.L. Turner	15 April '54
		CHECKED BY:	DATE
		V.F.G. REVISED	JAN 25 '55

Outer Wing Geometry

Data obtained from Drawings

7-4960-0001
SK 30865
o/w dwg. Fig 3



Note: The ℄ shown here is not the ℄ of the basic equations

A.V. ROE CANADA LIMITED MALTON – ONTARIO TECHNICAL DEPARTMENT (Aircraft)		REPORT NO. 7/0510/7 SHEET NO. 1-27	
AIRCRAFT C.105	INNER WING	PREPARED BY: V.F. GARDENER	DATE: JAN 25 55
		CHECKED BY: [signature]	DATE: 4 May 55

INNER WING GEOMETRY

SPAR AREAS – SKIN THICKNESSES AS FOR CENTRE SECTION
SEE p.

MAIN SPAR

$A_{117} = \left(\frac{1}{2} \times 52.6874 \cdot .1832\right) \times \cos \overset{.824438}{34°28.11'} = 4.1092 \ \square''$

$A_{107} = \frac{1}{2} \times 47.6073 \times .1832 \times .824438 = 3.7130 \ \square''$

$A_{97} = \frac{1}{2} \times 43.0171 \times .1832 \times .824438 = 3.3550 \ \square''$

$A_{87} = \frac{1}{2} \times 38.8695 \times .1832 \times .824438 = 3.0315 \ \square''$

$A_{73} = \frac{1}{2} \times 35.1218 \times .1832 \times .824438 = 2.7392 \ \square''$

NB THESE AREAS ARE FOR SKIN ONLY. FOR ADDITIONAL SPAR CAP AREA SEE p. 1-29 & 1-30

FWD C/S

$A_{118} = 52.6874 \times .1833 \times \cos \overset{.885934}{27°38.02'} = 8.5560 \ \square''$

$A_{108} = 47.6073 \times .1833 \times .885934 = 7.7310 \ \square''$

$A_{98} = 43.0171 \times .1833 \times .885934 = 6.9856 \ \square''$

$A_{88} = 38.8695 \times .1833 \times .885934 = 6.3121 \ \square''$

$A_{76} = 35.1218 \times .1833 \times .885934 = 5.7035 \ \square''$

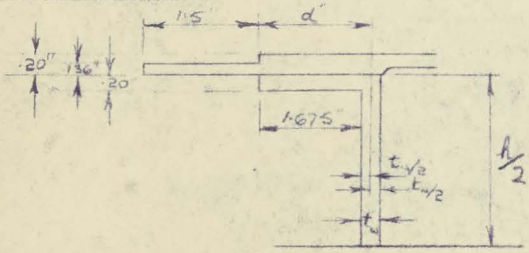
A.V.ROE CANADA LIMITED
MALTON – ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT C.105	INNER WING.	REPORT NO. 7/0510/7 SHEET NO. 1-29	
		PREPARED BY: V F GARDENER	DATE: JAN 28 '55
		CHECKED BY: D L Dunne	DATE: 4 May '55

INNER WING GEOMETRY

TOTAL MAIN SPAR AREA [INCLUDING SPAR CAPS]



$\underline{A_{117}}$ $t_w = \cdot125"$ $d = 1.675 + .0625 = 1.7375"$ $h = 15.0010"$

$$A_{117} = 4.1092 + \left(\frac{15.0010}{2} \times .125\right) + (1.675 \times .2) + (1.7375 \times .2) + (1.5 \times .136)$$

$$= 5.9333 \ \square"$$

$\underline{A_{107}}$ $t_w = \cdot15"$ $d = 1.675 + .075 = 1.75"$ $h = 13.3734"$

$$A_{107} = 3.7130 + \left(\frac{13.3734}{2} \times .15\right) + (1.675 \times .2) + (1.75 \times .2) + (1.5 \times .136)$$

$$= 5.6050 \ \square"$$

$\underline{A_{37}}$ $t_w = \cdot24"$ $d = 1.675 + .12 = 1.795"$ $h = 11.9242"$

$$A_{37} = 3.3550 + \left(\frac{11.9242}{2} \times .24\right) + (1.675 \times .2) + (1.795 \times .2) + (1.5 \times .136)$$

$$= 5.6839 \ \square"$$

A.V. ROE CANADA LIMITED MALTON – ONTARIO TECHNICAL DEPARTMENT (Aircraft)		REPORT NO. 7/0510/7 SHEET NO. 1-30	
		PREPARED BY: V.F. GARDENER	DATE: JAN 31 '55
AIRCRAFT C105	INNER WING	CHECKED BY: D.L. Turner	DATE: 4 May '55

INNER WING GEOMETRY.

TOTAL MAIN SPAR AREAS

$\underline{A_{87}}$ $\quad t_w = .5 \quad d = 1.675 + .25 = 1.925'' \quad h = 10.6146$

$$A_{87} = 3.0315 + \left(\frac{10.6146}{2} \times .5\right) + (1.675 \times 2) + (1.925 \times 2) + (1.5 \times .136)$$

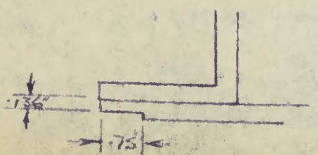
$$= 6.6091 \ \square''$$

$\underline{A_{73}}$ $\quad t_w = .5 \quad d = 1.675 + .25 = 1.925'' \quad h = 9.4132$

$$A_{73} = 2.7392 + \left(\frac{9.4132}{2} \times .5\right) + (1.675 \times .2) + (1.925 \times 2) + (1.5 \times .136)$$

$$= 6.0210 \ \square''$$

NB In the Region Between Fuselage Side & 50" Inboard of Transport Joint, the Structure is as Shown Below & Not as on Previous Page. Effect is Considered Negligible Since Areas Used Are Those For Bottom Surface Only And Top Surface Effective Areas Are Greater



A.V.ROE CANADA LIMITED
MALTON – ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO.	7/0510/7
SHEET NO.	1-31
PREPARED BY: V F GARDENER	DATE: JAN. 25 '55
CHECKED BY: DL Turner	DATE: 4 May '55
AIRCRAFT: C.105	INNER WING.

INNER WING GEOMETRY

RIB AREAS

TRANSPORT JOINT

RIB CAP AREAS WERE OBTAINED FROM WING GROUP (JAN 14 '55)

$A_{74} = 8.25 + \frac{1}{2} \times 23.0006 \times .150 \times .75 + 5.000 \times .212 = 10.6038 \square''$

$A_{75} = 5.50 + \frac{1}{2} \times 23.0006 \times .150 \times .75 + 5.000 \times .199 = 7.7888 \square''$

$A_{78} = 4.50 + \frac{1}{2} \times 23.0006 \times .150 \times .75 + 5.000 \times .187 = 6.7288 \square''$

$A_{81} = 4.10 + \frac{1}{2} \times 23.0006 \times .150 \times .75 + 5.000 \times .174 = 6.2638 \square''$

$A_{80} = 6.50 + \frac{1}{2} \times 23.0006 \times .064 \times .75 + 5.000 \times .231 = 8.2070 \square''$

THE OTHER RIB AREAS ARE OBTAINED BY FACTORISING THE VALUES OF 7/0510/2 BY THE RATIO OF SKIN THICKNESSES $\frac{.150}{.204} = .73529$

$A_{93} = A_{91} = A_{89} = 3.7069 \times .73529 = 2.7256 \square''$

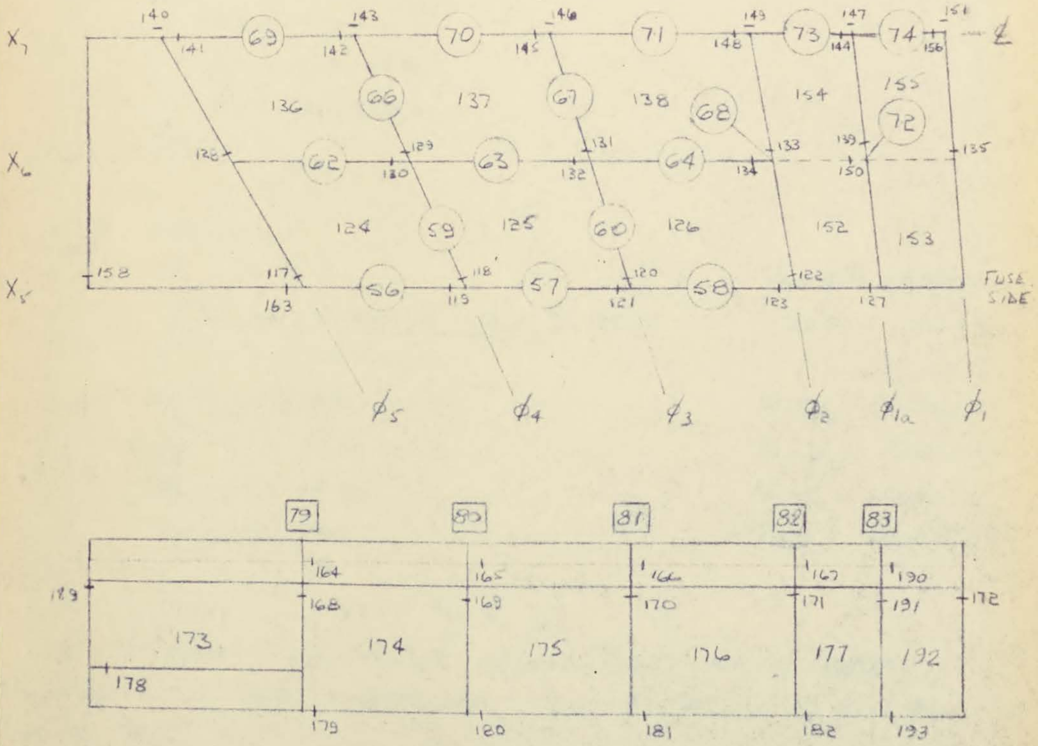
$A_{103} = A_{101} = A_{99} = 4.1024 \times .73529 = 3.0165 \square''$

$A_{113} = A_{111} = A_{109} = 4.5401 \times .73529 = 3.3383 \square''$

$A_{123} = A_{121} = A_{119}$ See centre section calculations

A.V.ROE CANADA LIMITED MALTON – ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7/0510/7 SHEET NO. 1-33	
AIRCRAFT C105	CENTRE SECTION	PREPARED BY: V F GARDENER	DATE: JAN 19 '55
		CHECKED BY: R. M. Sherard	DATE: JUNE 10/55

INDEX CODING



(p), (q) REDUNDANCIES (56)-(74), (79)-(83)

i, k STRESS POINTS 117-156, 158, 163-182, 189-193

A.V. ROE CANADA LIMITED MALTON – ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7/0510/7 SHEET NO. 1-34	
		PREPARED BY: V F GARDENER	DATE: JAN 18 '55
AIRCRAFT C 105	CENTRE SECTION	CHECKED BY: R. N. [illegible] R.N.	DATE: JUNE 10/55

CENTRE SECTION GEOMETRY (WING)

X ORDINATES

$X_5 = 323.3625''$

$X_6 = 352.1625''$

$X_7 = 380.9625''$

ϕ ORDINATES

$\phi_1 = 3^\circ\ 3.04'$ TAN $\phi_1 = .053296$

$\phi_{1a} = \tan^{-1}\left(\frac{761.5201 - 711.3213}{380.9625}\right) = 7^\circ\ 25.07'$ TAN $\phi_{1a} = .130193$

$\phi_2 = 11^\circ\ 10.87'$ TAN $\phi_2 = .197664$

$\phi_3 = 19^\circ\ 43.76'$ TAN $\phi_3 = .360600$

$\phi_4 = 27^\circ\ 38.02'$ TAN $\phi_4 = .523536$

$\phi_5 = 34^\circ\ 28.11'$ TAN $\phi_5 = .686472$

Y ORDINATES $Y = X \tan\phi$.

IN GEOMETRY CALCULATIONS INDIVIDUAL POINTS ARE NUMBERED ACCORDING TO THEIR X ϕ CO-ORDS. E.G., AT $X = X_2$, $\phi = \phi_1$ THE POINT IS CALLED POINT 2,1

POINT	TAN ϕ	Y	POINT	TAN ϕ	Y	POINT	TAN ϕ	Y
5,1	.053296	17.2339	6,1	.053296	18.7688	7,1	.053296	20.3038
5,1a	.130193	42.0997	6,1a	.130193	45.8492	7,1a	.130193	49.5988
5,2	.197664	63.9171	6,2	.197664	69.6098	7,2	.197664	75.3026
5,3	.360600	116.6045	6,3	.360600	126.9898	7,3	.360600	137.3751
5,4	.523536	169.2919	6,4	.523536	184.3697	7,4	.523536	199.4476
5,5	.686472	221.9793	6,5	.686472	241.7497	7,5	.686472	261.5201

A.V. ROE CANADA LIMITED
MALTON – ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

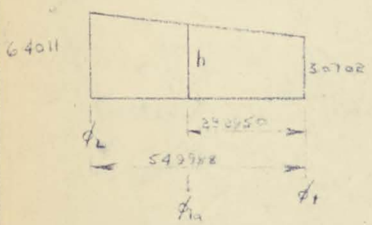
REPORT NO.	7/0510/7
SHEET NO.	1-35
PREPARED BY: V.F GARDENER	DATE: JAN 19 55
CHECKED BY: NOT USED P.M.J.	DATE: JUNE 10/55
AIRCRAFT: C 105	CENTRE SECTION

CENTRE SECTION GEOMETRY (WING)

DETERMINATION OF 'h' VALUES (WING THICKNESS) AT POINTS (7,1a) (6,1a) (5,1a)

WING SURFACE AFT OF REAR SPAR $\phi_1 < \phi < \phi_2$ IS FLAT PLANE I.E THERE IS A STRAIGHT LINE VARIATION OF 'h' BETWEEN THE REAR SPAR & THE AFT BEAM (FALSE SPAR)

POINT (7,1a)

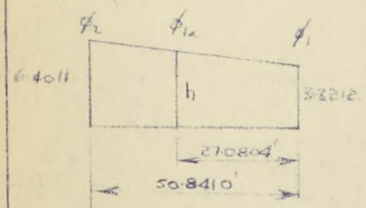


$$h = 3.0702 + \frac{3.3309 \times 29.2950}{54.9988}$$

1.7742

$$= 4.8444''$$

POINT (6,1a)

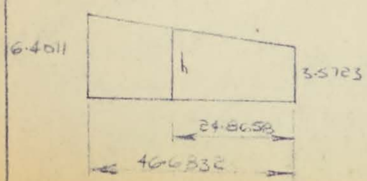


1.6405

$$h = 3.3212 + \frac{3.0799 \times 27.0804}{50.8410}$$

$$= 4.9617''$$

POINT (5,1a)



1.5068

$$h = 3.5723 + \frac{2.8288 \times 24.8658}{46.6832}$$

$$= 5.0791''$$

A.V.ROE CANADA LIMITED MALTON – ONTARIO TECHNICAL DEPARTMENT (Aircraft)		REPORT NO. 7/0510/7 SHEET NO. 1-40	
		PREPARED BY: V F GARDENER	DATE: JAN 19 55
AIRCRAFT C105	CENTRE SECTION	CHECKED BY: [illegible signature]	DATE: JUNE 10/55

CENTRE SECTION GEOMETRY (WING)

SPAR AREAS

FORWARD OF THE REAR SPAR, THE SPAR AREAS WERE BASED ON THE SKIN AREA ALONE ASSUMING SKINS FULLY EFFECTIVE AND USING EQUIVALENT THICKNESSES SUPPLIED BY THE WING GROUP

MAIN SPAR $\bar{t} = \cdot 1892$

F.S. C/S $\bar{t} = \dfrac{\cdot 1892 + \cdot 1775}{2} = \cdot 1833''$

AFT C/S $\bar{t} = \dfrac{\cdot 1775 + \cdot 1765}{2} = \cdot 1770$

REAR SPAR $\bar{t} = \cdot 1765''$

MAIN SPAR

SKIN FORWARD OF MAIN SPAR IS ·064 THICK AND WILL BE ASSUMED TO BE 50% EFFECTIVE

X_7 $A_{140} = \left(\frac{1}{2} \times 60{\cdot}0262 \times \cdot 1892 + \frac{1}{4} \times 60{\cdot}0515 \times \cdot 064\right) \times \cos 34^\circ 28' 11''$ ·824438

$= 5{\cdot}6600\ \square''$ 599

X_6 $A_{122} = \left(\frac{1}{2} \times 57{\cdot}38 \times \cdot 1892 + \frac{1}{4} \times 57{\cdot}38 \times \cdot 064\right) \times \cdot 824438 = 5{\cdot}2321\ \square''$

X_5 $A_{117} = \left(\frac{1}{2} \times 52{\cdot}6874 \times \cdot 1892 + \frac{1}{4} \times 52{\cdot}6874 \times \cdot 064\right) \times \cdot 824438 = 4{\cdot}8042\ \square''$

NB THESE AREAS DO NOT INCLUDE SPAR CAP AREA.
SEE p. 1-41

A.V.ROE CANADA LIMITED MALTON – ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7/0510/7 SHEET NO. 1-42	
AIRCRAFT C.105	CENTRE SECTION	PREPARED BY: V.F. GARDENER	DATE: JAN 19 55
		CHECKED BY: [illegible] PAS	DATE: JUNE 10/55

CENTRE SECTION GEOMETRY (WING)

SPAR AREAS

FWD C/S

$X_7 \quad A_{143} = 62.0725 \times .1833 \times \overset{.885934}{\cos 27^\circ 38.02'} = 10.0801\ \square''$

$X_6 \quad A_{129} = 57.3800 \times .1833 \times .885934 = 9.3181\ \square''$

$X_5 \quad A_{118} = 52.6874 \times .1833 \times .885934 = 8.5560\ \square''$

AFT C/S

$X_7 \quad A_{146} = 62.0725 \times .1770 \times \overset{.940707}{\cos 19^\circ 47.76'} = 10.3354\ \square''$

$X_6 \quad A_{131} = 57.3800 \times .1770 \times .940707 = 9.5541\ \square''$

$X_5 \quad A_{120} = 52.6874 \times .1770 \times .940707 = 8.7727\ \square''$

REAR SPAR

DUE TO ELEVATOR JACK ACCESS DOOR, SKIN AFT OF REAR SPAR WILL BE CONSIDERED AT 50% EFFECTIVE FOR SPANWISE LOADING

R/S CAP AREA OF .90 $\square''$ WILL ALSO BE INCLUDED

SKIN THICKNESS

$X_5 \quad t = .12''$

$X_6 \quad t = \dfrac{\left(\frac{.25+.12}{2}\right) + .12}{2} = .1525''$

$X_7 \quad t = \frac{1}{2}(.25 + .12) = .185''$

$X_7 \quad A_{149} = \left(\frac{1}{4} \times 25.7038 \times .185 + \frac{1}{2} \times 62.0725 \times .1765 \overset{+.90}{}\right) \times \overset{.981019}{\cos 11^\circ 10.87'} = 7.4231\ \square''$

$X_6 \quad A_{133} = \left(\frac{1}{4} \times 23.7606 \times .1525 + \frac{1}{2} \times 57.38 \times .1765 \overset{+.90}{}\right) \times .981019 = 6.7393\ \square''$

$X_5 \quad A_{122} = \left(\frac{1}{4} \times 21.8174 \times .12 + \frac{1}{2} \times 52.6874 \times .1765 + .90\right) \times .981019 = 6.1122\ \square''$

N.B. R/S CAP AREA SHOULD NOT BE MULTIPLIED BY COS α. ERROR NEGLIGIBLE

AVRO AIRCRAFT LIMITED — TECHNICAL DEPARTMENT (Aircraft)		REPORT NO. 7/0510/7 *	
		SHEET NO. 1-52	
AIRCRAFT		PREPARED BY	DATE
C-105	GEOMETRY	R.N. SHEARLY	JUNE 21/55
		CHECKED BY	DATE
		J. Gardener	June '55

CENTER SECTION WING

RIB FLANGE AREAS

THE VALUES BETWEEN THE MAIN SPAR & THE REAR SPAR WERE FOUND AS IN 7/0510/2 USING THE NEW THICKNESS $t = .155''$

ALONG $x = 323.3625$ BETWEEN M/S & R/S.

$$AREA = .155 \times \frac{28.80}{2} + .150 \times \frac{31.1767}{2}{}^{*} = 4.57025\ \square''$$

FORWARD OF M/S

CAP AREA (TOP) = .9950 □" (REF REPORT P. JEWEL OF WING GROUP ENTITLED FUSELAGE SIDE RIB AFT OF STN. 485.)

(BOTTOM) = .7441 □"

SKIN (TOP) = 2 × .064 × 14.4 = 1.8432 □ (ASSUMED EFF WIDTH 14.4" INB'D & OUTB'D)

(BOTTOM) = .064 × 14.4 = .9216 □" (ASSUMED EFF. WIDTH 14.4" INB'D ONLY)

TOTAL AREA (TOP) = .9950 + 1.8432 = 2.8382 □"

(BOTTOM) = .7441 + .9216 = 1.6657 □"

2.5" × .25" STEEL CAP

A_{27} = AVER. OF TOP & BOTTOM AREAS = 2.2520 □"

AFT OF R/S

CAP AREA (TOP) = $2.5 \times .25 \times \frac{30 \times 10^6}{10 \times 10^6} + (2.5 + 1.4).12 + (2.32 + 1.2).17 = 2.9414$ □"

2.5 × .12 1.4 × .12"

T 2.32" × 1.2" × .17"

(BOTTOM) = $2.4 \times .095 \times \frac{30 \times 10^6}{10 \times 10^6} + (2.4 + 1.25).17 +$

T 2.4" × 1.25" × .17"

$+ 2.4 \times .12 = 1.5925$ □"

SKIN .12" 2.4" × .095" STEEL CAP

NOTE: AFTER CONSULTATION WITH MR GRZEDZIELSKI REGARDING THE TWO ABOVE AREAS IT WAS DECIDED TO INCLUDE THE HINGE AT BOTTOM & SET

A_{123} AFT OF R/S = 3.0000 □"

ALONG $x = 352.1625$ BETWEEN M/S & R/S.

AREA = .155 × 28.80 = 4.4640 □"

ALONG $x = 380.3625$ BETWEEN M/S & R/S.

AREA = .155 × 28.80 / 2 = 2.2320 □" **

FWD OF M/S

AREA = 1.0000 □" REF. ALEX GRZEDZIELSKI

* REF. 7/0510/2 P 2-2

** ADD FIN WING INTERSECTION BOX AREA AT R/S & R/S (SEE NEXT SHEET)

FORM 1714A

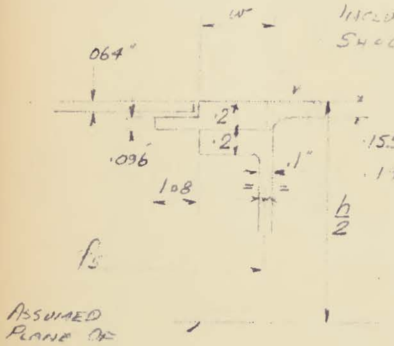
AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT		REPORT NO. 7/0510/7	
		SHEET NO. 1-34	
		PREPARED BY	DATE
C-105	GEOMETRY	R.M. SHEARLY	JUNE 6/55
		CHECKED BY	DATE
		J. Gardenas	June 55

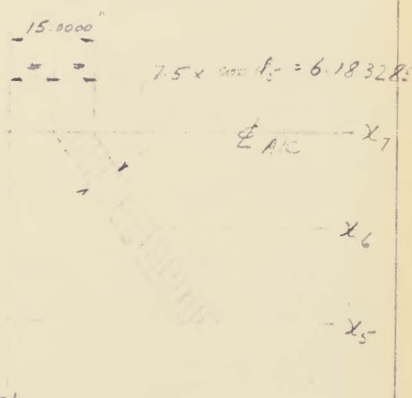
CENTER SECTION WING

SPAR FLANGE AREAS

M/S



SECTION THRU M/S CAP BETWEEN FUSELAGE SIDE & A/C CENTERLINE



PLAN VIEW OF M/S BETWEEN FUSELAGE SIDE (x_5) & A/C ℄(x_7) WITH PORTION OF .064" SKIN CONSIDERED EFFECTIVE SHOWN SHADED.

DIMENSION LABELLED W ABOVE IS CONSTANT AT 1.13" BETWEEN x_7 & x_6 AND IS ASSUMED TAPER LINEARLY BETWEEN x_6 & x_5 ATTAINING THE VALUE 1.78" AT x_5

M/S x_7

AREA FWD OF $\mathcal{L}_S$ $\quad .40 \times 1.08 + .096 \times 1.08 + .064 \times (6.183285 - 1.08) + .05 \times \frac{17.7644 - .8}{6} = 1.004691$ □"

AREA AFT OF $\mathcal{L}_S$ $\quad .20 \times .05 + .05 \times \frac{17.7644 - .8}{6} + .1135 \times 31.03625 \times .024453 = 5.10421$ □"

TOTAL AREA M/S AT ℄, A_{M7} $= 6.10890$ □"

FORM 1194

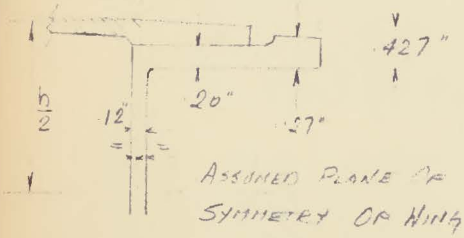
AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

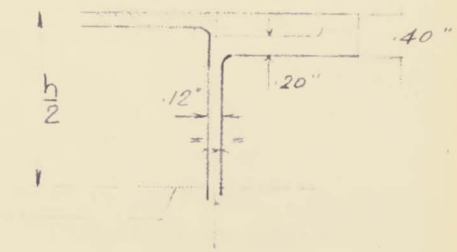
REPORT NO. 7/2610/7			
SHEET NO. 1-56			
AIRCRAFT C-105	GEOMETRY	PREPARED BY R.N. SHEARLY	DATE JUNE 6/55
		CHECKED BY D.J. Gardener	DATE June '55

CENTER SECTION WING
R/S

SKIN PANEL INCLUDES SHADED PORTION



SKIN PANEL INCLUDES SHADED PORTION



ACTUAL REAR SPAR CAP AREA

ASSUMED REAR SPAR CAP AREA

R/S x_7

AREA FWD OF ℄ $= \frac{1}{2} \times .1992 \times 31.03625 + .20 \times .06 + .06 \times \frac{6.4011 - .8}{6} =$
$= 6.13308$ □"

AREA AFT OF ℄ $= .06 \times \frac{6.4011 - .8}{6} + .40 \times 2.84 =$
$= 1.19201$ □"

TOTAL AREA R/S AT ℄, $A_{144} = 7.32509$ □"

x_6 $A_{133} = \frac{1}{2} \times .1992 \times 28.6899 + .20 \times .06 + .12 \times \frac{6.4011 - .8}{6} + .40 \times 2.84 =$
$= 6.96657$ □"

x_5 $A_{122} = \frac{1}{2} \times .1992 \times 26.3436 + .20 \times .06 + .12 \times \frac{6.4011 - .8}{6} + .40 \times 2.84 =$
$= 6.40805$ □"

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

REPORT NO. 7/0510/7		
SHEET NO. 1-57		
AIRCRAFT: C105	Outer Wing	PREPARED BY: D L Turner — DATE: 10 Feb. '55
		CHECKED BY: — DATE:

<u>Transformation of Outer Wing Co-ordinates into Inner Wing System.</u>

The transformation equations may be written in the form:

$$X = a\chi - b\gamma + c$$
$$Y = a\gamma + b\chi + d$$

where X & Y are the inner wing co-ordinates
χ & γ are the outer wing co-ordinates
a, b, c & d are the transformation constants.

At the outer wing transport joint the co-ordinates of load points 19, 34, 38, 41 & 43 are known in both systems, hence two of these points will be used as a basis for solving for the transformation constants. The two points selected are load points 34 & 43

At 34 $\quad X = 215.5596 \quad Y = 42.6084$
$\quad\quad\quad \chi = 208.8917 \quad \gamma = -7.6308$

At 43 $\quad X = 215.5596 \quad Y = 147.9754$
$\quad\quad\quad \chi = 275.9137 \quad \gamma = 73.6662$

ref figs 5 & 6

Substituting these values in the transformation equations

(1) $\quad 215.5596 = 208.8917\,a + 7.6308\,b + c$

(2) $\quad 42.6084 = -7.6308\,a + 208.8917\,b + d$

(3) $\quad 215.5596 = 275.9137\,a - 73.6662\,b + c$

(4) $\quad 147.9754 = 73.6662\,a + 275.9137\,b + d$

FORM 1319A

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

AIRCRAFT: C105	Outer Wing	REPORT No. 7/0510/7	
		SHEET No. 1-58	
		PREPARED BY	DATE
		D L Turner	10 Feb. 55
		CHECKED BY	DATE

Transformation of Outer Wing Co-ordinates

Combine (1) & (3) and (2) & (4)

(5) $0 = 67.0220\,a - 81.2970\,b$ (3)-(1)

(6) $105.3670 = 81.2970\,a + 67.0220\,b$ (4)-(2)

from (5) & (6)

$$a = \frac{81.2970}{67.0220}\,b = 1.212989764\,b$$

$$b = \frac{105.3670}{67.0220 + 81.2970 \times 1.212989764} = \frac{105.3670}{165.6344288} = .636141898$$

$$a = 1.212989764 \times .636141898 = .771633611$$

from (1)

$$215.5596 = 208.8917 \times .771633611 + 7.6308 \times .636141898 + c$$

$$c = 215.5596 - 161.187857 - 4.854272$$

$$= 49.5175$$

from (2)

$$42.6084 = -7.6308 \times .771633611 + 208.8917 \times .636141898 + d$$

$$d = 42.6084 + 5.88818 - 132.88476$$

$$= -84.3882$$

So that

$$X = .771633611\,x - .636141898\,y + 49.5175$$

$$Y = .636141898\,x + .771633611\,y - 84.3882$$

FORM 1319A

Transformation of Outer Wing Co-ordinates.

This transformation is used in preparing the table of wing co-ordinates given on figures 2 & 9.
Calculations used in the preparation of the table are given on the following sheets.

It should be noted that c & d should represent the ordinates of the origin of the outer wing co-ordinate system in the inner wing system; and that a & b represent the cos & sin of the angle of rotation.
From fig 6 the co-ordinates required are given as

$c = 49.5292$
$d = -84.3846$

as compared to the above values

$c = 43.5175$
$d = -84.3862$

Sines & cosines agree to four figures.

It is felt, that for the purpose of balancing the Q loads on the structure, it is preferable to use the given transformation. A more accurate transformation cannot be made, as basic geometry does not give enough significant figures.

Wing Analysis

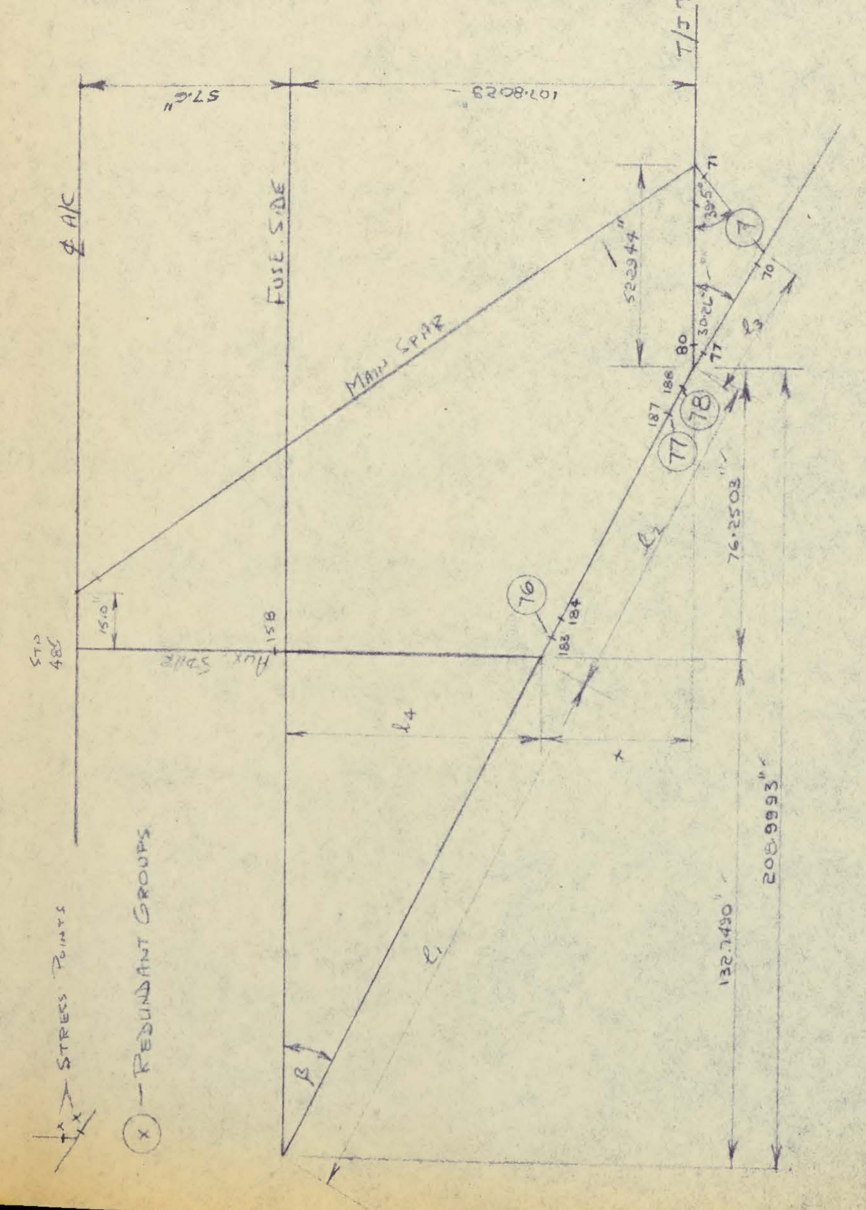
Wing Coordinate System

Coordinate Values at Load Points 1 to 43

5	6	7	8	9	10	11	12	13	14	15	16	17	18	19
.636141898 X_o	.771633611 Y_o	X	Y		Load Point Number	X_o	Y_o	.771633611 X_o	.636141898 Y_o	.636141898 X_o	.771633611 Y_o	X	Y	
.636141898 × 1	.771633611 × 2	3 - 4 + 49.5175	5 + 6 -84.3882					.771633611 × 11	.636141898 × 12	.636141898 × 11	.771633611 × 12	13 - 14 + 49.5175	15 + 16 -84.3882	
56.3683	[illegible]	80.2204	-31.7519		15									
4.5717	-1.0888	80.2204	-60.9093		16									
9.8620	2.2304	91.6120	12.2222		17									
9.8620	-2.2094	111.5211	-36.7356		18									
0.2186	31.4097	108.7974	17.2401	←	19	309.1772	114.0193	238.5715	72.5299	196.6806	87.9780	215.5591	200.2104	
70.2186	-3.1115	137.2570	-17.2811		20	173.3489	63.7259	133.7618	40.6659	110.2745	49.3274	142.6134	75.2137	
70.5751	40.5155	125.9828	46.7024		21		46.2824		29.4422		35.7130	153.8371	61.5993	
10.5751	-4.0134	162.6929	2.1735		22		31.8355		20.2519		24.5653	163.0274	50.4516	
4.5717	-12.1633	89.7625	-72.4798		23		14.9942		9.2840		11.2614	173.9953	37.1477	
49.8620	-14.7889	121.3632	-48.3101		24	173.3489	-6.3324	133.7618	-4.0283	110.2745	-4.8863	187.3076	21.0000	
10.2186	-14.1860	146.7991	-28.8556		25	189.8159	69.9984	146.4683	44.5289	120.7498	54.0131	151.4569	90.3747	
10.5751	-15.5879	172.2350	-9.4010		26		50.6789		32.2390		39.1055	163.7468	75.4611	
10.2745	-16.4608	196.8497	9.4255		27		34.8597		22.1757		26.8989	173.8101	63.2605	
20.7498	-16.9250	209.9389	19.4366		28	189.8159	15.9806	146.4683	10.1659	120.7498	12.3312	185.8199	48.6928	

A.V.ROE CANADA LIMITED MALTON – ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7/0510/7 SHEET NO. 1-62	
		PREPARED BY: V F GARDENER	DATE: JAN 25 '55
AIRCRAFT C.105	LEADING EDGE	CHECKED BY: WMHG	DATE: May 5 '55

LEADING EDGE GEOMETRY (Front Spar Beam)



DIMENSIONS GIVEN ABOVE ARE TAKEN FROM 7/0510/4, P 2-42 ET SEQ & BATH DRG.

A.V.ROE CANADA LIMITED
MALTON – ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT C105	LEADING EDGE	REPORT NO. 7/0510/7 SHEET NO. 1-63	
		PREPARED BY: V F GARDENER	DATE: JAN 25 '55
		CHECKED BY: [illegible]	DATE: May 5, '55

LEADING EDGE GEOMETRY

$$x = \frac{107.8029 \times 76.2503}{208.9993} = 39.3303''$$

$$l_4 = 107.8029 - 39.3303 = 68.4726''$$

$$l_1 + l_2 = \left(107.8029^2 + 208.9993^2\right)^{1/2} = 235.16414$$

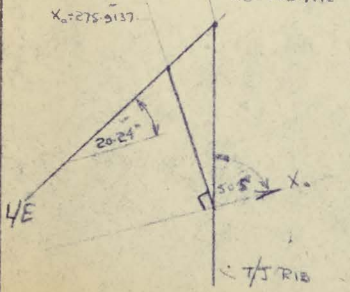
$$l_1 = \left(68.4726^2 + 132.7430^2\right)^{\frac{1}{2}} = 149.36738$$

$$l_2 = 85.7962''$$
$$l_1 = 149.3680''$$

$$\tan\beta = \frac{107.8029}{208.9993} = .515805$$

$$\beta = 27^\circ\ 17.09'$$

THE LENGTH l_3 IS CALCULATED FROM OUTER WING DATA



$$\delta X_o = 309.1772 - 275.9137 = 33.2635''$$

$$l_3 = \frac{33.2635}{\cos 20.24^\circ} = 35.4527''$$

.93823

A.V.ROE CANADA LIMITED MALTON – ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7/0510/7 SHEET NO. 2-2	
		PREPARED BY: D Turner	DATE: 19 Jan. '55
AIRCRAFT C105	Wing Tip	CHECKED BY: D Hjardene	DATE: Apr 28 '55

<u>Elastic Influence Coefficients</u> (Wing Tip)

Working Formula (ref. sht. 7-19 et seq.)

$$V = \frac{1}{2}\left\{\frac{4LI_r}{3Eh_r^2}\,\frac{t}{t_r}\,\Delta\xi\left[\xi_2\sigma_2^2 + \bar{\xi}_{av}\,\sigma_1\sigma_2 + \xi_1\sigma_1^2\right] + \frac{J_r b_r^2 L}{GA_r^2}\,\frac{t}{t_r}\,\xi_{av}\,\Delta\xi\,T^2\right\}$$

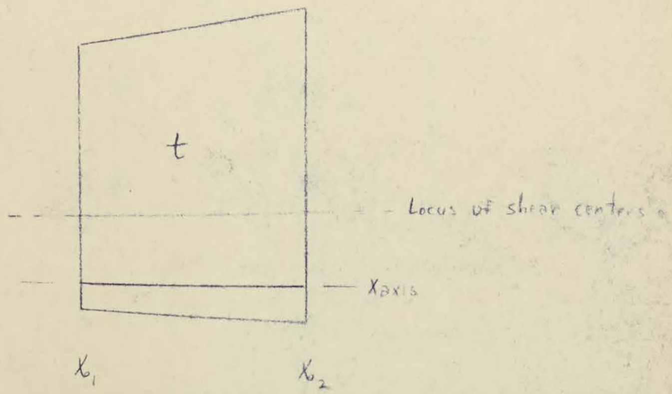
where $\xi = \frac{x}{L}$ $\qquad \bar{\xi}_{av} = \xi_{av}\left(1 - 2\frac{\Delta\xi^2}{\xi_{gm}^2}\right)$

$\xi_{gm}^2 = \xi_1\xi_2 \qquad t$ = average t in bay

Subscript r denotes values at rib 18 (i/b boundary)

This equation applies for each bay of the wing tip structure. Values of the elastic influence coefficients for each bay are calculated on the following sheets. A dummy matrix is included.

Reference should be made to the sketch on sht. 2-3



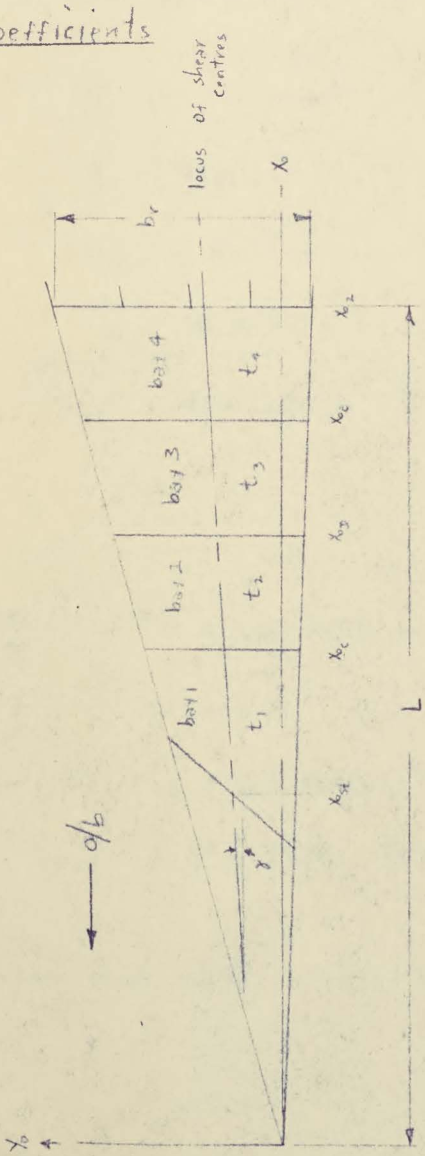
A.V.ROE CANADA LIMITED MALTON – ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**	REPORT NO. 7/0510/7 SHEET NO. 2-3	
	PREPARED BY: D L Turner	DATE: 17 Jan. '55
AIRCRAFT C105 — Wing Tip	CHECKED BY: D Ghandemer DRJ	DATE: Apr 28 '55

Elastic Influence Coefficients

Wing Tip

Data required at i/b boundary are:

t_r, h_r (at locus of shear centres)
I_r, J_r, A_r. A_r is wing enclosed area.



AVRO AIRCRAFT LIMITED TECHNICAL DEPARTMENT (Aircraft)		REPORT NO 7/0510/7 SHEET NO 2-4 *
AIRCRAFT: C105	Wing Tip	PREPARED BY D.F. Turner — DATE 16 May 55 CHECKED BY R.N. Sheaty DLD — DATE MAY 27/55

Elastic Influence Coefficients

Summary of Basic Data Required.

$L = X_{o_2} = 173.3489$ $\qquad I_r = 251.936 \text{ in}^4$ $\qquad$ (ref. sht. 1-3)

$X_{o_{st}} = 47.2210$ $\qquad t_r = .160 \text{ in.}$

$X_{o_C} = 78.3819$ $\qquad h_r = 6.3875 \text{ in.}$

$X_{o_D} = 110.3819$ $\qquad b_r = 70.2583 \text{ in.}$

$X_{o_E} = 142.3819$ $\qquad A_r = 406.1289 \text{ in}^2$

$X_{o_2} = 173.3489$

$t_1 = .1075 \qquad t_2 = .1226 \qquad t_3 = .1378 \qquad t_4 = .1528$

$$J_r = \frac{2A_r^2 t_r}{b_r} = \frac{2 \times 406.1289^2 \times .160}{70.2583} = 751.2459 \text{ in}^4$$

$$Z_r = \frac{2I_r}{h_r} = \frac{2 \times 251.936}{6.3875} = 78.8841 \text{ in}^3$$

$$\theta_r = 2A_r t_r = 2 \times 406.1289 \times .160 = 129.9613 \text{ in}^3$$

$$\frac{X_{o_{st}}}{L} = .2724 \qquad \frac{X_{o_C}}{L} = .4522 \qquad \frac{X_{o_D}}{L} = .6368 \qquad \frac{X_{o_E}}{L} = .8214 \qquad \frac{X_{o_2}}{L} = 1.0000$$

Data here is obtained from shts. 1-3, 1-5, 1-15, 1-17, 1-18 & 1-19.

Determination of Bay Constants Required for the Calculation of the Wing Tip Elastic Influence Coefficients.

A. V. R

AIRCRAF

WEIGHT

C. G. PO

	1	2	3	4	5	6	7	8	9	10	
Bay	t	ξ_2	ξ_1	$\Delta\xi$ (2) - (3)	ξ_{av} $\frac{1}{2}[(2)+(3)]$	ξ_{gm}^2 (2) × (3)	$\Delta\xi^2$ $(4)^2$	$\frac{\Delta\xi^2}{\xi_{gm}^2}$ (7)/(6)	2 × (8)	1 - (9)	(6)
1	.1075	.4522	.2724	.1728	.3623	.1232	.0323	.2622	.5244	.4756	.17
2	.1226	.6368	.4522	.1846	.5445	.2880	.0341	.1184	.2368	.7632	.4
3	.1378	.8219	.6368	.1846	.7291	.5231	.0341	.0652	.1304	.8696	.6
4	.1528	1.0000	.8214	.1786	.9107	.8214	.0319	.0388	.0776	.9224	.8

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT C105	Wing Tip	REPORT NO 7/0510/7	
		SHEET NO 2-6 *	
		PREPARED BY D.L. Turner	DATE 16 May 55
		CHECKED BY R.M. Steele	DATE May 24/55

Elastic Influence Coefficients

Energy Constants.

From data on the previous sheets:

$$\frac{4LI_r}{3Eh_r^2} = \frac{4\times173.3489\times251.936}{3\times10.5\times10^6\times6.3825} = \frac{174,691.3139}{1285.2049\times10^6} = 135.9249\times10^{-6}$$

$$\frac{J_r b_r^2 L}{GA_r^2} = \frac{751.2459\times76.2583^2\times173.3489}{4.0\times10^6\times406.1289^2} = \frac{642.8335}{659,762.733} = 974.3404\times10^{-6}$$

From these figures, and data obtained from sht. 2-5, the elastic influence coefficients may be calculated for each bay.

	Bay 1	Bay 2	Bay 3	Bay 4
$\frac{t}{t_r}\Delta\xi\,\xi_2$	.0546	.0901	.1306	.1706
135.9249×10^{-6} x above	7.4215×10^{-6}	12.2468×10^{-6}	17.7518×10^{-6}	23.1888×10^{-6}
$\frac{t}{t_r}\Delta\xi\,\bar{\xi}_{av}$	.0208	.0588	.1008	.1433
135.9249×10^{-6} x above	2.8272×10^{-6}	7.9924×10^{-6}	13.7012×10^{-6}	19.4780×10^{-6}
$\frac{t}{t_r}\Delta\xi\,\xi_1$	.0329	.0690	.1013	.1401
135.9249×10^{-6} x above	4.4719×10^{-6}	8.6392×10^{-6}	13.7692×10^{-6}	19.0431×10^{-6}
$\frac{t}{t_r}\Delta\xi\,\xi_{av}$	.0438	.0770	.1159	.1554
974.3404×10^{-6} x above	42.6761×10^{-6}	75.0242×10^{-6}	112.9261×10^{-6}	151.4125×10^{-6}

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C105	Wing Tip	REPORT NO 7/0510/7	SHEET NO 2-7 *
		PREPARED BY: D L Turner	DATE: 16 May 55
		CHECKED BY: R D Sheely	DATE: May 24/55

Elastic Influence Coefficients

From these figures the elements of the C_{ik} matrix for the wing tip are determined:

Bending $(\times 10^6)$

	σ_7^2	σ_9^2	$\sigma_9\sigma_7$	σ_{11}^2	$\sigma_{11}\sigma_9$	σ_{13}^2	$\sigma_{13}\sigma_{11}$
Bay 1	7.4215						
2	8.6992	12.2468	7.9924				
3		13.7692		17.7518	13.7012		
4				19.0431		23.1888	19.4780
Total	16.1207	26.0160	7.9924	36.7949	13.7012	23.1888	19.4780
or	16.1207	26.0160	2×3.9962	36.7949	2×6.8506	23.1888	2×9.7390

Torsion $(\times 10^6)$

$$42.6761\,T_6^2 + 75.0242\,T_8^2 + 112.9261\,T_{10}^2 + 151.4125\,T_{12}^2$$

Summarizing

$$V = \tfrac{1}{2}\Big[.016121\,\sigma_7^2 + .026016\,\sigma_9^2 + 2\times.003996\,\sigma_9\sigma_7 + .036795\,\sigma_{11}^2$$

$$+ 2\times.006851\,\sigma_{11}\sigma_9 + .023189\,\sigma_{13}^2 + 2\times.009739\,\sigma_{13}\sigma_{11} + .042676\,T_6^2$$

$$+ .075024\,T_8^2 + .112926\,T_{10}^2 + .151412\,T_{12}^2\Big]$$

for load in kips.

A.V.ROE CANADA LIMITED
MALTON – ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO.	7/0510/7
SHEET NO.	2-8
PREPARED BY:	[illegible signature]
DATE:	17 Jan '55
CHECKED BY:	[illegible signature]
DATE:	Apr 28 '55
AIRCRAFT	C105
	Wing Tip

Elastic Influence Coefficients

Dummy C_{ik} matrix.

$i \downarrow$ / $K \rightarrow$	6	7	8	9	10	11	12	13
6	T							
7		σ		σ				
8			T					
9		σ		σ		σ		
10					T			
11				σ		σ		σ
12							T	
13						σ		σ

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C105	Wing Tip	REPORT NO. 7/0510/7	SHEET NO. 2-9
		PREPARED BY: D.L. Turner	DATE: 24 Feb. '55
		CHECKED BY: V. Gardener	DATE: Ap. 29 '55

Statically Determinate Stresses.

Working Formulae. (see

(1) $M_i = \Sigma (x_{o_s} - x_{o_Q}) Q_a$ where x_{o_s} is x_o ordinate of locus of shear centres at i

(2) $T_i = \Sigma (y_{o_s} - y_{o_Q}) Q_a$

x_{o_Q} is x_o ordinate of load Q_a

y_{o_s} is y_o ordinate of locus of shear centres at i.

y_{o_Q} is y_o ordinate of load Q_a

(3) $\sigma_i = \frac{M_i h_i}{2 I_i} \sec \gamma = \frac{M_i}{Z_i} \sec \gamma$

(4) $\tau_i = \frac{T_i}{2 A_i t_i} = \frac{T_i}{\theta_i}$

Z_i & θ_i are defined on sht. 2-11

γ is angle between the x_o axis and the locus of shear centres.

see sketches shts 1-20, 2-3 fig 4

In this section of the report, all wing tip loads are transferred along their appropriate ribs (or equivalent) to the local shear centre. This transfer results in a vertical shear and a torque (vector perp. to rib) at the local shear centre.

This torque is transferred inboard as a constant torque to the transport joint rib. These torque values are determined in the upper table on sht. 2-12. (using eq'n 2 above)

The vertical shear is transferred inboard to the transport joint creating bending moments (vectors parallel to ribs) at each bending stress point. These bending moments are determined in the lower table on sht. 2-12 (using eq'n 1 above)

From these torques & moments the stresses at the various stress points are determined using equations (3) & (4). This is done on sht. 2-13

Note: M +ve for tension in upper wing surface
σ_1 +ve for tensile stress in upper upper wing surface
T +ve for nose up torque
τ +ve for shear stress in upper skin due to T +ve
Q +ve load down.

FORM 1319A

AVRO AIRCRAFT LIMITED TECHNICAL DEPARTMENT (Aircraft)		REPORT NO. 7/0510/7 SHEET NO. 2-11 *
AIRCRAFT: C105	Wing Tip	PREPARED BY: [illegible] DATE: 16 May 55 CHECKED BY: [illegible] DATE: [illegible]

Statically Determinate Stress Coefficients

Determination of Z & θ Values.

From sht 7-20

$$Z(x_0) = Z_r \left(\frac{x_0}{l}\right)^2 \frac{t}{t_r} = Z_r \xi^2 \frac{t}{t_r}$$

$$\theta(x) = \theta_r \left(\frac{x_0}{l}\right)^2 \frac{t}{t_r} = \theta_r \xi^2 \frac{t}{t_r}$$

From sht. 2-4 $\quad Z_r = 78.8841 \quad \theta_r = 129.9613 \quad t_r = .160$

x_0	ξ	ξ^2	t	$\frac{t}{t_r}$	$\xi^2 \frac{t}{t_r}$	$Z(x)$ $Z_r \xi^2 \frac{t}{t_r}$	$\theta(x)$ $\theta_r \xi^2 \frac{t}{t_r}$
$x_{0_{ST}}$	.2724	.0742	.1001	.6252	.0464	3.6602	6.0302
x_{0_C}	.4522	.2095	.1143	.7177	.1968	11.5802	19.0783
x_{0_D}	.6368	.4055	.1302	.8132	.3298	26.0160	42.8612
x_{0_E}	.8214	.6747	.1454	.9082	.6128	48.3402	79.6403
x_{0_2}	1.0000	1.0000	.1601	1.0000	1.0000	78.8841	129.9613

Data obtained from shts 1-13, 2-3 & 2-4

...s at Stress Points

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

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

REPORT NO. - 7/0[illegible]
SHEET - - 2-12
DATE - - 16 M[illegible]
PREPARED BY - [illegible]

6	7	8	9	10	11	12	13	14	15	16	17	18	19
Q_6	Q_7	Q_8	Q_9	Q_{10}	Q_{11}	Q_{12}							
4.0323 17.2802	-32.5062 +21.4635	5.2012 +21.9638	16.4110 +7.8500	17.8633 +13.0323	19.0323 +17.2802	20.2012 +21.96[illegible]	$-Y_{a4}+Y_{a5}$						
—	—	—	24.3010	—	—	—	ref.	sht. 1-8, 1-19	& 1-16				
—	—	—	24.3010	30.3559	—	—							
1.3125	—	—	24.3010	30.3559	36.3125	—							
1.3125	31.0384	26.6690	24.3010	30.3559	36.3125	41.6690			123.[illegible]				
							$X_5 - X_6$						
			33.7557				$78.3819 - X_6$						
			71.7557	32.0000			$110.3819 - X_6$						
2.0000			103.7557	64.0000	32.0000		$142.3819 - X_6$						
.9670	30.9670	30.9670	134.7227	94.9670	62.9670	30.9670	173.3[illegible]		953.3232				
.3819	142.3819	142.3819	38.6262	78.3819	110.3819	142.3819							

Determination of Elements of T_1 Matrix (Wing Tip)

$$\sigma_i = \frac{M_i}{Z_i} \sec \gamma \qquad \tau_i = \frac{T_i}{\theta_i}$$

	1	2	3	4	5	6	7	8	9	10	
			←				Moments	Due To			
Stress	Z_i	$\frac{\sec \gamma}{Z_i}$	Q_1	Q_2	Q_3	Q_4	Q_5	Q_6	Q	Q_8	
σ_2	11.5802	.08709	21.2118	39.7557	—	—	—	—	—	—	33.
σ_5	26.0160	.03877	53.2118	71.7557	32.0000	32.0000	—	—	—	—	71.
σ_{10}	48.3402	.02086	85.2118	103.7557	64.0000	64.0000	32.0000	32.0000	—	—	103.
σ_{12}	78.5641	.01278	116.1788	134.7227	94.9670	94.9670	62.9670	62.9670	30.9670	30.9670	134
			←				Torque				
	θ_i	$\frac{1}{\theta_i}$									
τ_2	6.0302	.16583	$\overline{10.7659}$	9.3010	—	—	—	—	—	—	24.
τ_5	12.0783	.05241	$\overline{10.7659}$	9.3010	$\overline{15.8123}$	15.9559	—	—	—	—	24.
τ_{10}	42.8612	.02333	$\overline{10.7659}$	9.3010	$\overline{15.8123}$	15.9559	$\overline{23.4253}$	21.3125	—	—	24.
τ_{12}	79.6403	.01256	$\overline{10.7659}$	9.3010	$\overline{15.8123}$	15.9559	$\overline{23.4253}$	21.3125	$\overline{31.038}$	26.6620	24.
Σ											

FORM 1544

lements of T_1 Matrix (Wing Tip)

$$J_1 = \frac{T_1}{\theta_1}$$

5	6	7	8	9	10	11	12	13	14	15	16	17	18	19
Moments Due To														
Q_3	Q_4	Q_5	Q_6	Q	Q_8	Q_9	Q_{10}	Q_{11}	Q_{12}		Q_1 (2)x(3)	Q_2 (2)x(4)	Q_3 (2)x(5)	Q_4 (2)x(6
—	—	—	—	—	—	39.7557	—	—	—		1.84734	3.46232	—	—
32.0000	32.0000	—	—	—	—	71.7557	32.0000	—	—		2.06302	2.78197	1.24064	1.240
64.0000	64.0000	32.0000	32.0000	—	—	103.7557	64.0000	32.0000	—		1.77752	2.16434	1.33504	1.335
4.9670	94.9670	62.9670	62.9670	30.9670	30.9670	134.7227	94.9670	62.9670	30.9670		1.48977	1.72176	1.21368	1.213
Torque														
—	—	—	—	—	—	24.3010	—	—	—		$\overline{1}.28531$	1.59238	—	—
$\overline{5}.8123$	15.9559	—	—	—	—	24.3010	30.9559	—	—		$\overline{.5}6424$	.48797	$\overline{.8}2872$	.8362
$\overline{5}.8123$	15.9559	$\overline{23}.4253$	21.3125	—	—	24.3010	30.9559	36.3125	—		$\overline{.2}5197$	.21699	$\overline{.3}6890$	.3722
$\overline{5}.8123$	15.9559	$\overline{23}.4253$	21.3125	$\overline{31}.038[illegible]$	26.[illegible]690	24.3010	30.9559	36.3125	41.6690		$\overline{.1}3522$	.11682	$\overline{.1}9860$	.2004
											4.43671	12.49405	2.39314	5.1382
											-	-	-	-

Loads on Outer Wing at Rib 18 Shear Centre Due to Wing Tip Loads.

From the table on sht. 2-13 the moment at rib 18 (vector parallel to rib 18) may be written:

$$M_{x_{o_2}} = 116.1788\, Q_1 + 134.7227(Q_2 + Q_9) + 94.9670(Q_3 + Q_4 + Q_{10})$$

$$+ 62.9670(Q_5 + Q_6 + Q_{11}) + 30.9670(Q_7 + Q_8 + Q_{12})$$

Due to the fact that the focus of the wing tip beam trapezoid is different from the focus of the outer wing trapezoid, the torque at rib 18 given in the table on sht. 2-13 may not be used for calculating wing stresses inboard of rib 18.

The correct torque at rib 18 for this purpose may be found by transferring wing tip loads to the intersection of the relevent rib and the "adjusted locus of shear centres" and transferring the resultant torque inboard (as a constant value) to rib 18. (rather than the procedure used for calculating wing tip torques — sht. 2-9 & 2-12.

The "adjusted locus of shear centres" is defined as the straight line passing through the focus of the outer wing trapezoid ($x_o = y_o = 0$) and the shear centre at rib 18 (as determined on shts. 1-2 & 1-3)

This makes the torque acting on the outer wing at rib 18 compatible with the moments acting on the spars at rib 18.

This torque is calculated on sht. 2-17 and is

$$T_{x_{o_2}} = -12.6661\, Q_1 + 7.0975\, Q_2 - 17.3656\, Q_3 + 14.4026\, Q_4$$

$$-24.4552\, Q_5 + 20.2826\, Q_6 - 31.5449\, Q_7 + 26.1625\, Q_8$$

$$+22.0966\, Q_9 + 29.4026\, Q_{10} + 35.2826\, Q_{11} + 41.1625\, Q_{12}$$

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

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

AIRCRAFT		PREPARED BY	DATE
C105	Wing Tip	D L Turner	18 May '55
		CHECKED BY	DATE
		R. M. Sheahy	MAY 24/55

Loads on Outer Wing at Rib 18 Shear Centre Due to Wing Tip Loads.

The vertical shear at rib 18 is simply

$$P_{x_{o_2}} = \sum_{a=1}^{a=12} Q_a$$

These values are used in calculating the elements of the G_1 matrix.

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C105	Wing Tip	REPORT NO. 7/0510/7 SHEET NO. 2-17 *
		PREPARED BY: D L Turner — DATE: 18 May '55
		CHECKED BY: R N Sheardy — DATE: 20 MAY 24/55

To find the Y ordinates of the intersections of the wing tip ribs and the "adjusted locus of shear centres, and from this data to calculate $T_{X_{0_2}}$.

At $X_0 = X_{0_2} = 173.3489$ (rib 18) sht. 1-5

$Y_0 = Y_0 = 25.5202$ (at shear centre) sht. 1-3

At $X_0 = 0$ $Y_{0_{S_a}} = 0$

hence $Y_{0_{S_a}} = \frac{25.5202}{173.3489} X_0 = .1472187\ X_0$

load pt.	(1) * X_0	(2) Y_{S_a} .1472187 (1)	(3) * $-Y_a$	(4) arm (2)+(3)
1	57.1701	8.41651	-21.0826	-12.6661
2	38.6262	5.68650	1.4110	7.0975
3	78.3819	11.53928	-28.9049	-17.3656
4	78.3819	11.53928	2.8633	14.4026
5	110.3819	16.25028	-40.7055	-24.4552
6	110.3819	16.25028	4.0323	20.2826
7	142.3819	20.96128	-52.5062	-31.5449
8	142.3819	20.96128	5.2012	26.1625
9	38.6262	5.68560	16.4110	22.0966
10	78.3819	11.53928	17.8633	29.4026
11	110.3819	16.25028	19.0323	35.2826
12	142.3819	20.96128	20.2012	41.1625

* ref. sht. 1-8

From cols 2 & 3 of the above table the torque arms (col. 4) may be calculated according to:

$$arm = Y_{S_a} - Y_{0_a}$$

From this data $T_{X_{0_2}}$ as given on sht. 2-15 is taken.

Elastic Influence Coefficients.

In this section of the report the elastic influence coefficients for the major portion of the wing structure are determined. Wing tip coefficients have been calculated separately in section 2.

It should be noted that, a good many of these elements may be taken directly from report 7/0510/2, some may be found by factoring values given in report 7/0510/2, and that relatively few must be calculated from scratch.

Outer wing elements remain as given in report 7/0510/2, but terms incorporating the flexibility of the "transport joint triangle" portion of the front spar must be added. It should be noted that the outboard bay of the outer wing as defined in report 7/0510/2 is deleted here, having been incorporated in the wing tip.

Inner wing elements are all revised. This is done by factoring the elements of report 7/0510/2 to take into account the changes in skin thickness and rib & spar cap areas.

Centre section elements are all revised in the same manner as for the inner wing.

A.V.ROE CANADA LIMITED MALTON – ONTARIO TECHNICAL DEPARTMENT (Aircraft)		REPORT NO. 7/0510/7 SHEET NO. 3-2	
		PREPARED BY: V.F. GARDENER	DATE: FEB 11 '55
AIRCRAFT C105	OUTER WING	CHECKED BY: R.N. SHEARLY	DATE: APR. 28/55

ELASTIC INFLUENCE COEFFS

THE COEFFS FOR THE OUTER WING ARE THE SAME AS THOSE OF REPORT 7/0510/2 EXCEPT FOR (a) ADDITIONAL SPAR TERMS INTRODUCED BY EXTENSION OF THE FRONT SPAR INBOARD TO CUT THE TRANSPORT JOINT AT LOADING POINT 19 Note that panel 83 is considered stiff.

EXTRA TERMS ARE IN σ_{77}^2, $\sigma_{77}\sigma_{70}$, & ADDITIONAL σ_{70}^2.

(b) REVISED SPAR TERMS AT RIB ALONG x_{o_2} SEE SHT 3-4

From 7/0510/2 p 1-43

$$V = \frac{1}{2}\left[KV_6\,\sigma_{n_2}^2 + 2KV_7\,\sigma_{n_2}\sigma_{n_1} + KV_8\,\sigma_{n_1}^2\right]$$

WHERE

$$KV_6 = \frac{2A_{s_2}(X_{o_2}-X_{o_1})}{3E\cos\theta_s}\left[1+\frac{3(X_{o_2}-X_{o_1})}{4X_{o_2}}\right]$$

$$KV_7 = \frac{A_{s_m}}{3E\cos\theta_s}(X_{o_2}-X_{o_1}) \qquad A_{s_m} = \frac{(A_{s_1}+A_{s_2})}{2}$$

$$KV_8 = \frac{2A_{s_1}(X_{o_2}-X_{o_1})}{3E\cos\theta_s}\left[1-\frac{3(X_{o_2}-X_{o_1})}{4X_{o_1}}\right]$$

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No 7/0510/7	SHEET No 3-4	
PREPARED BY R.M. SHEARLY	DATE APR. 29/55	
CHECKED BY	DATE	
AIRCRAFT C-105	OUTER WING	

ELASTIC INFLUENCE COEFFICIENTS

REVISED SPAR COEFFTS

SINCE THE BAY BETWEEN x_{o_1} & x_{o_2} OF THE 7/0510/2 ANALYSIS HAS BEEN CUT OFF AND INCLUDED WITH THE WING TIP IN THIS ANALYSIS THE COEFFICIENTS ASSOCIATED WITH σ_{m_i} FOR $i = 14, 16, 18, 20$ & 22 WILL NOW CONSIST OF ONLY ONE PART (FROM THE INB'D SPAR ELEMENT)

SEE TABULATED COEFFTS BELOW REF 7/0510/2 SHT 1-71

SPAR ELEMENT	KV_8 FOR IN LB UNITS	KV_8 FOR IN KIP UNITS
14-27	2.44196×10^{-6}	.00244196
16-29	3.38632×10^{-6}	.00338632
18-31	3.40540×10^{-6}	.00340540
20-33	3.91688×10^{-6}	.00391688
22-35	2.59399×10^{-6}	.00259399

POISSON'S RATIO EFFECT

IN THE CASE OF THE COEFF'TS REPRESENTING THE POISSON'S RATIO EFFECT ALONG THE RIB AT x_{o_2} WHICH ARE SHOWN IN 7/0510/2 SHT 1-68, THE VALUE AS USED IN THE OLD REPORT WAS USED. IT WAS FELT THAT THIS WAS JUSTIFIED BY THE FACTS

1) THE SPAR STRESS AT POINTS $i = 14, 16, 18, 20$ & 22 DO EXTEND OUTB'D IN THE ACTUAL STRUCTURE.

2) THE RIB AREA INCLUDES SKIN OUTBOARD OF THE LINE x_{o_2} SUCH THAT THE WIDTH OF SKIN ASSUMED LUMPED ON LINE x_{o_2} COMES EQUALLY FROM THE INBOARD & OUTBOARD SIDE OF LINE x_{o_2}

A. V. ROE CA

MALTON,

TECHNICAL DE

AIRCRAFT -

WEIGHT -

C. G. POSITION -

C.R ELEMENTS T/I RIB

$$V = \frac{1}{2}\left[KV_{23}\,\sigma_1^2 + 2KV_{24}\,\sigma_1\sigma_2 + KV_{25}\,\sigma_2^2\right]$$ REF 7/0510/2 p2-15

$$KV_{23} = \frac{L}{2E}\left[A_1 + \tfrac{1}{3}A_2\right] \qquad KV_{24} = \frac{L}{6E}\left[A_1 + A_2\right] \qquad KV_{25} = \frac{L}{2E}\left[\tfrac{1}{3}A_1 + A_2\right]$$

$$E = 10.5 \times 10^6 \text{ LB}/\square''$$

	1	2	3	4	5	6	7	8	9	10	11	12
RIB BAY	L	$\frac{L}{2E}\times10^6$	$\frac{L}{6E}\times10^6$	A_1	A_2	$\frac{A_1}{3}$	$\frac{A_2}{3}$	$A_1+\frac{A_2}{3}$	$\frac{A_1}{3}+A_2$	A_1+A_2	$KV_{23}\times10^6$ 2 × 8	$KV_{24}\times$ 3 × 10
80-74	52.2944	2.49021	.83007	10.6038	8.2070	3.5346	2.73567	13.33947	11.74160	18.8108	33.21808	15.6142
74-75	35.1218	1.67247	.55749	7.7888	10.6038	2.59627	3.5346	11.32340	13.20007	18.3926	18.93804	10.2534
75-78	35.1218	1.67247	.55749	6.7288	7.7888	2.24293	2.59627	9.32507	10.03173	14.5176	15.59590	8.0934
78-81	35.1218	1.67247	.55743	6.2638	6.7288	2.08793	2.24293	8.50673	8.81673	12.9926	14.22725	7.2432

A.V.ROE CANADA LIMITED MALTON – ONTARIO TECHNICAL DEPARTMENT (Aircraft)		REPORT NO. 7/0510/7 SHEET NO. 3-6	
		PREPARED BY: V.F. GARDENER	DATE: FEB 11 '55.
AIRCRAFT C105	INNER WING	CHECKED BY: R.M. SHEARLY	DATE: APR. 29/55

ELASTIC INFLUENCE COEFFS

SINCE ALL THE SPAR & RIB AREAS & PANEL THICKNESS HAVE BEEN CHANGED THE ELEMENTS OF THE C_{ik} MATRIX HAVE TO BE RECALCULATED

WITH THE EXCEPTION OF THE TRANSPORT JOINT RIB, THE NEW RIB AREAS HAVE BEEN OBTAINED BY FACTORIZING THE OLD AREAS BY THE RATIO OF NEW TO OLD SKIN THICKNESS

$$\text{i.e. } \frac{.150}{.204} = .735294$$

THUS IT IS POSSIBLE TO OBTAIN THE NEW PANEL, RIB, RIB-PANEL COUPLING, SPAR-PANEL COUPLING, & POISSON'S RATIO TERMS BY MULTIPLYING THE OLD VALUES BY .735294.

THE OTHER ELEMENTS ARE ALSO OBTAINED BY MEANS OF THE RATIO OF THE TWO AREAS BUT THE INDIVIDUAL RATIO FOR EACH PARTICULAR POINT MUST BE USED
THIS IS DONE ON p 3-7 et seq.

THE T/J RIB ELEMENTS ARE CALCULATED ON p 3-5

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

AIRCRAFT: C105

WEIGHT:

C. G. POSITION:

REPORT NO.: 7/0510/7

SHEET: 3-7

DATE: FEB 7 55

PREPARED BY: VF GARDENER

5	6	7	8	9	10	11	12	13	14	15	16	17	18	19
VALUES					VALUES FROM REPORT 7/0510/5 p 2-22									
A_s	$\frac{1}{2}(3-5)$ A_{sm}	④/①	⑤/①	⑥/①	$KV_7 \times 10^6$	$KV_6 \times 10^6$	$KV_8 \times 10^6$	$KV_7 \times 10^6$ ⑨ ⑩	$KV_6 \times 10^6$ ⑧ ⑪	$KV_8 \times 10^6$				
[illegible]	6.3151	1.18128	1.21894	1.20070	46581	10.3010	8.30577	5.5330	12.5663	9.8114				
[illegible]	6.1465	1.21894	.98254	1.09749	5.4834	12.1479	9.7782	6.0246	11.3479	11.3130				
[illegible]	5.6445	.98254	.90740	.94421	[illegible]	14.3700	11.5341	6.1228	13.0333	11.3442				
[illegible]	5.7631	.90740	.89790	.90248	[illegible]	17.0129	13.6437	** 6.9257	* 15.2769	12.3803				
[illegible]	6.0078	.62911	.64633	.63804	[illegible]	17.3622	13.7481	4.9516	11.1571	8.6491				
[illegible]	6.6489	.64633	.66290	.65493	[illegible]	20.6143	16.3898	6.0047	13.6652	10.5032				
[illegible]	7.3583	.66290	.67959	.67156	[illegible]	24.6281	[illegible]	7.4280	16.7370	12.9746				
[illegible]	8.1435	.67959	.69521	.68771	[illegible]	29.4865	23.3834	9.0979	* 20.4933	15.8911				
[illegible]	6.1600	.63773	.65308	.64570	[illegible]	16.4967	[illegible]	4.7814	10.7737	8.3518				
[illegible]	6.8173	.65308	.66890	.66130	[illegible]	19.7271	15.6631	5.8563	13.1954	10.2253				
[illegible]	7.5447	.66890	.68371	.67659	[illegible]	23.6384	18.7304	7.1726	16.1618	12.5288				
[illegible]	8.3498	.68371	.69785	.69107	27122	28.3652	22.4439	8.7850	* 19.7947	15.3451				

* THESE VALUES TO BE COMBINED WITH C/S VALUES p. 3-13

** See sht. 3-21 for revised value to be used

...NERGY - SPAR 2 RIB TERMS

...RL SECTION

...6349206 $\times 10^{-6}$

$$KV_4 = \frac{2}{3E} A_R (Y_2 - Y_1) \qquad KV_5 = \frac{1}{3E} A_R (Y_2 - Y_1)$$

$$KV_6 = \frac{2A_{s_2}(x_2 - x_1)}{3E \cos \phi_s}\left[1 - \frac{3}{4}\left(\frac{x_2 - x_1}{x_2}\right)\right]$$

$$KV_7 = \frac{A_{s_m}(x_2 - x_1)}{3E \cos \phi_s} \qquad A_{s_m} = \frac{A_{s_1} - A_{s_2}}{2}$$

$$KV_8 = \frac{2A_{s_1}(x_2 - x_1)}{3E \cos \phi_s}\left[1 - \frac{3}{4}\left(\frac{x_2 - x_1}{x_1}\right)\right]$$

5	6	7	8	9	10	11	12	13	14	15	16	17	18	19
x_1	$x_2 - x_1$	$\cos \phi_s$	$\frac{2 \times 10^6}{3E \cos \phi_s}$	$\frac{1 \times 10^6}{3E \cos \phi_s}$	$A_{s_1}(x_2 - x_1)$	$A_{s_2}(x_2 - x_1)$	$A_{s_m}(x_2 - x_1)$	(8) × (11)	(9) × (12) $KV_7 \times 10^6$	(8) × (10)	$\frac{x_2 - x_1}{x_2}$	$\frac{x_2 - x_1}{x_1}$	$1 + \frac{3}{4}\left(\frac{x_2 - x_1}{x_2}\right)$	$1 - \frac{3}{4}\left(\frac{x_2 - x_1}{x_1}\right)$
23.3625	28.80	.824438	.077012	.038506	182.2406	196.6982	189.4694	15.14812	7.29571	14.03471	.081780	.087064	1.061335	.93320
"	"	.885934	.071667	.035834	246.4128	262.3613	257.3870	19.23265	9.22321	17.65967	"	"	"	"
"		.940707	.067494	.033747	252.6538	275.1581	263.9059	18.57152	8.90603	17.05262			"	"
"		.981019	.064720	.032360	176.0314	194.0918	185.0616	12.56162	5.98859	11.39275	"	"	"	"
52.1625	"	.824438	.077012	.038506	196.6982	211.1558	203.9270	16.26153	7.85241	15.14812	.075598	.081780	1.056699	.93866
"	"	.885934	.071667	.035834	268.3613	290.3069	279.3341	20.80542	10.00966	19.23265	"	"	"	"
"		.940707	.067494	.033747	275.1581	297.6595	286.4088	20.09023	9.66544	18.57152				"
"		.981019	.064720	.032360	194.0918	213.7853	203.9386	13.83618	6.59945	12.56162	"	"		

A. V. ROE C…
MALTO…
TECHNICAL…

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

STRAIN ENERGY - PANELS

Centre Section

$\frac{4G}{3E} = \frac{4}{3} \times \frac{4}{10.5} = .50794$

$KV_1 = \frac{2At}{G}\left[1 + \frac{4G}{3E}\left(\tan^2\phi_2 + \tan\phi_2 \tan\phi_1 + \tan^2\phi_1\right)\right]$

	1	2	3	4	5	6	7	8	9	10	11	12
PANEL	$\tan\phi_1$	$\tan\phi_2$	$\tan^2\phi_1$	$\tan^2\phi_2$	$\tan\phi_1 \times \tan\phi_2$	3 + 4 + 5	$\frac{4G}{3E}$. 6	1 + 7	A $\square''$	t IN	$\frac{2t}{G} \times 10^{6}$	2… 9 …
124	.523536	.686472	.274090	.471244	.359393	1.104727	.561135	1.561135	1584.96	.150	.075	118.
125	.360609	.523536	.130032	.274090	.188787	.592909	.301162	1.301162	1584.96	.150	.075	"
126	.197664	.360650	.039071	.130032	.071278	.240381	.122019	1.122019	1584.96	.150	.075	"
136	.523536	.686472						1.561135	1720.11	.150	.075	129.
137	.360600	.523536						1.301162	1720.11	.150	.075	
138	.197664	.360650						1.122019	1720.11	.150	.075	

$\frac{4G}{3L}$ $\frac{4}{3}\cdot\frac{4}{...}$ 50734

$\tan^2 \phi_1)$

A. V. ROE CANADA LIMITED

MALTON, ONTARIO

TECHNICAL DEPT. (AIRFRAME)

AIRCRAFT - - C 105

WEIGHT - -

C. G. POSITION -

REPORT NO. - 7/[illegible]/7

SHEET - - 3-10

DATE - - Feb 17 54

PREPARED BY - V F G[illegible]

	7	8	9	10	11	12	13	14	15	16	17	18	19
	$\frac{4G}{3L}\cdot$ (6)	1+(7)	n D''	t IN	$\frac{2t}{G}\times10^{6}$	$\frac{2nt}{G}\times10$ (9)×(11)	$KV_1\times10^{6}$ (8)×(12)						
7	.561135	1.561135	1584.36	.150	.075	118.8720	185.57524						
?	.301162	1.301162	1584.36	.150	.075	"	154.67173						
1	.122019	1.122019	1584.36	.150	.075	"	133.38615						
		1.561135	1720.11	.150	.075	129.00825	201.39929						
		1.301162	1720.11	.150	.075	"	167.86063						
		1.122019	1720.11	.150	.075	"	144.76003						

A. V. ROE CAN
MALTON, C
TECHNICAL DEPT

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

STRAIN ENERGY - PANEL COUPLING

CENTRE SECTION

$KV_2 = \frac{2At}{E} \tan \phi_{AV} \cdot \frac{x_2}{x_2 + x_1}$ $\qquad KV_3 = \frac{2At}{E} \tan \phi_{AV} \cdot \frac{x_1}{x_2 + x_1}$

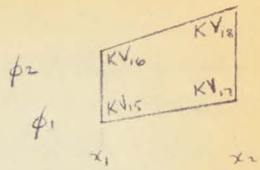
	1	2	3	4	5	6	7	8	9	10	11	12
PANEL	ϕ_{AV}	$\tan \phi_{AV}$	x_1	x_2	$x_1 + x_2$	$\frac{x_1}{x_1+x_2}$	$\frac{x_2}{x_1+x_2}$	A	t	$\frac{2t}{E} \times 10^6$	$\frac{2At}{E} \times 10^6$ (8)×(10)	(2)×(6)
124	31° 3.065'	.602075	323.3625	352.1625	675.5250	.478683	.521317	1584.96	.150	028571	45.28389	.288203
125	23° 43.89'	.439625						1584.96	.150		"	.210441
126	15° 30.315'	.277425	323.3625	352.1625	675.5250	.478683	.521317	1584.96	.150		"	.132799
136		.602075	352.1625	380.9625	733.1250	.480358	.519642	1720.11	.150		49.14526	.289212
137		.439625						1720.11	.150		"	.211177
138		.277425	362.1625	330.9625	733.1250	480358	519642	1720.11	.150	028571	"	.133263

STRAIN ENERGY – PANEL POISSON EFFECT

CENTRE SECTION

$\frac{t}{E} = \frac{150}{105 \times 10^{?}} = .01428571 \times 10^{-6}$

$\nu = \cdot 33$



A. V. ROE
TECH
AIRCRAFT -
WEIGHT -
C. G. POSITIO

PANEL	1	2	3	4	5	6	7	8	9	10	11
	x_1	x_2	$2x_1$	$2x_2$	x_1+x_2	$2x_1-x_2$	$2x_2-x_1$	A	$\frac{A}{x_1+x_2}$	$\frac{t}{E}\cdot\frac{A}{x_1+x_2}\times 10^{6}$	(10) × (6
124	323·3625	352·1625	646·7250	704·3250	675·5250	294·5625	380·9625	1584·96	234626	03351799	9·87314
125	"	"	"	"	"	"	"	"	"	"	"
126	"	"	"	"	"	"	"	"	"	"	"
136	352·1625	380·9625	704·3250	761·9250	733·1250	323·3625	409·7625	1720·11	"	"	10·83846
137	"	"	"	"	"	"	"	"	"	"	"
138	"	"	"	"	"	"	"	"	"	"	"

$KV_{15} = -\frac{t}{E}\cdot\frac{A}{x_2+x_1}(2x_1-x_2)(\nu\cos^2\phi_1-\sin^2\phi_1)$

$KV_{16} = -\frac{t}{E}\cdot\frac{A}{x_2+x_1}(2x_1-x_2)(\nu\cos^2\phi_2-\sin^2\phi_2)$

$KV_{17} = -\frac{t}{E}\cdot\frac{A}{x_2+x_1}(2x_2-x_1)(\nu\cos^2\phi_1-\sin^2\phi_1)$

$KV_{18} = -\frac{t}{E}\cdot\frac{A}{x_2+x_1}(2x_2-x_1)(\nu\cos^2\phi_2-\sin^2\phi_2)$

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/7
SHEET NO 3-15

AIRCRAFT: C105

C_{ik}

PREPARED BY: D L Turner
DATE: 17 June '55
CHECKED BY
DATE

Centre Section – Revised Coefficients

Since the calculations in the preceding pages were completed, major changes have occurred in the section properties of the centre section wing

In view of this it was decided to recalculate the elastic influence coefficients of this portion of the structure. This has been done in the pages following.

In some cases, straight factoring is all that is necessary, but in most cases, particularly in the border areas, major corrections were required. In particular, the treatment of the spars at the fuselage side should be noted. Here account is taken of the discontinuity in spar areas which occurs.

FORM 1313A

Elastic Influence Coefficients – Centre Section

Revised Values for Spars.

$A_{s_m} = \frac{A_{s_1} + A_{s_2}}{2}$

	1	2	3	4	5	6	7	8	9	10	11	
				Revised	Revised	Revised	Factors for					
Spar Bay	A_{s_1}	A_{s_2}	A_{s_m}	A_{s_1}	A_{s_2}	A_{s_m}	KV_6 $\frac{(5)}{(2)}$	KV_7 $\frac{(6)}{(3)}$	KV_8 $\frac{(4)}{(1)}$	* KV_6 $\times 10^6$	* KV_7 $\times 10^6$	*
117 – 128	6.3278	6.8298	6.5788	5.5293	5.7099	5.6196	.83603	.85420	.87381	16.07723	7.29571	1
118 – 129	8.5560	9.3181	8.93705	8.9901	9.7908	9.3904	1.05073	1.05073	1.05074	20.41228	9.22321	16
120 – 131	8.7727	9.5591	9.1634	9.6846	10.5472	10.1159	1.10394	1.10395	1.10395	19.71060	8.90603	15
122 – 133	6.1122	6.7393	6.42575	6.4080	6.8666	6.6373	1.01889	1.03292	1.04840	13.33209	5.98859	10
128 – 140	6.8298	7.3318	7.0808	5.7099	6.1089	5.9094	.83321	.83457	.83603	17.18354	7.85241	14
129 – 143	9.3181	10.0801	9.6991	9.7908	10.5915	10.1912	1.05073	1.05074	1.05073	21.98507	10.00966	18
131 – 146	9.5541	10.3354	9.94475	10.5472	11.4098	10.9785	1.10395	1.10395	1.10394	21.22933	9.66594	17
133 – 149	6.7393	7.4231	7.0812	6.8666	7.3251	7.0958	.98680	1.00206	1.01889	14.62068	6.59995	11

* ref. sht. 3-9

Final values to be used ar

on

$A_{s_1} + A_{s_2}$

A. V. ROE CANADA LIMITED

MALTON, ONTARIO

TECHNICAL DEPT. (AIRFRAME)

REPORT NO. - 7/0510/7	AIRCRAFT - - C105
SHEET - - 3-16	WEIGHT - -
DATE - - 9 June 55	C. G. POSITION -
PREPARED BY - [illegible]	

7	8	9	10	11	12	13	14	15	16	17	18	19
Factors for						Revised Values						[illegible]
KV_6 $\frac{(5)}{(2)}$	KV_7 $\frac{(6)}{(3)}$	KV_8 $\frac{(4)}{(1)}$	$^{*}KV_6$ $\times 10^6$	$^{*}KV_7$ $\times 10^6$	$^{*}KV_8$ $\times 10^6$	KV_6 $\times 10^6$	KV_7 $\times 10^6$	KV_8 $\times 10^6$				
						(7)x(10)	(8)x(11)	(9)x(12)				
.83603	.85420	.87381	16.07723	7.29571	13.09722	13.49105	6.23200	11.49998				
1.05073	1.05073	1.05074	20.41228	9.22321	16.48004	21.44779	9.69110	17.31629				
1.10394	1.10395	1.10395	19.71060	8.90603	15.91354	21.75932	9.83181	17.56725				
1.01889	1.03292	1.04840	13.33209	5.98859	10.63174	13.58393	6.18573	11.19632				
.83321	.83457	.83603	17.18359	7.85291	14.21901	14.31750	6.55339	11.88752				
1.05073	1.05074	1.05073	21.98507	10.00366	18.05302	23.10037	10.51755	18.96885				
1.10335	1.10395	1.10394	21.22933	9.66544	17.43244	23.43612	10.67016	19.24437				
.98680	1.00206	1.01889	14.62068	6.59995	11.72115	14.42769	6.61304	12.01388				

Final values to be used are given on sht 3-27

Elastic Influence Coefficients — Centre Section

Revised Values for Ribs.

$2V = \sigma_i^2 KV_{26} + 2\sigma_i\sigma_k KV_{2}$

$KV_{26} = 2\frac{Y_2-Y_1}{3E}\left[\frac{A_i^2}{(\frac{3}{4}A_i+\frac{1}{4}A_k)}\right]$ $KV_{27} = \frac{Y_2-Y_1}{3E}\left[\frac{A_i A_k}{(\frac{1}{4}A_i+\frac{1}{4}A_k)}\right]$ $KV_{28} = 2\frac{Y_2-Y_1}{3E}\left[\frac{1}{4}\right.$

	1	2	3	4	5	6	7	8	9	10	11	
Rib Box	A_i	A_k	$\frac{3}{4}A_i$	$\frac{3}{4}A_k$	$\frac{1}{4}A_i$	$\frac{1}{4}A_k$	Y_2-Y_1	$\frac{Y_2-Y_1}{3E}$ $\times 10^6$	$2\frac{Y_2-Y_1}{3E}$ $\times 10^6$	A_i^2	A_k^2	A
Blkhd -127	2.2520	2.2520	1.68900	1.68900	.56300	.56300	54.5407	1.73144	3.46288	5.0715	5.0715	5.
127 - 119	2.2520	4.5702	1.68900	3.42765	.56300	1.14255	52.6872	1.67260	3.34520	5.0715	20.8867	10.
119 - 121	4.5702	4.5702	3.42765	3.42765	1.14255	1.14255	52.6872	1.67260	3.34520	20.8867	20.8867	20
121 - 123	4.5702	3.0000	3.42765	2.25000	1.14255	.75000	52.6872	1.67260	3.34520	20.8867	9.0000	13.
M/s - 130	4.4640	4.4640	3.34800	3.34800	1.11600	1.11600	57.3798	1.82158	3.64316	19.9273	19.9273	19
130 - 132	4.4640	4.4640	3.34800	3.34800	1.11600	1.11600	57.3798	1.82158	3.64316	19.9273	19.9273	19.
132 - 134	4.4640	4.4640	3.34800	3.34800	1.11600	1.11600	57.3798	1.82158	3.64316	19.9273	19.9273	19
Blkhd - 141	1.0000	1.0000	.7500	.7500	.2500	.2500	15.0000	.47620	.95240	1.0000	1.0000	1.0
141 - 142	1.0000	2.2320	.7500	1.67400	.2500	.55800	62.0725	1.97056	3.94112	1.0000	4.9818	2.
142 - 145	2.2320	4.0930	1.67400	3.06975	.55800	1.02325	62.0725	1.97056	3.9412	4.9818	16.7526	9.
145 - 148	4.0930	2.7400	3.06975	2.05500	1.02325	.68500	62.0725	1.97056	3.94112	16.7526	7.5076	11.2

$\frac{1}{3E} = .031746 \times 10^{-6}$

FORM 1544

...fficients — Center Section

...bs.

$$2V = \sigma_1^2 KV_{26} + 2\sigma_1\sigma_2 KV_{27} + \sigma_2^2 KV_{28}$$

$$KV_{27} = \frac{Y_2 - Y_1}{3E}\left[\frac{A_1 A_2}{(\frac{1}{4}A_1 + \frac{3}{4}A_2)}\right] \qquad KV_{28} = 2\,\frac{Y_2 - Y_1}{3E}\left[\frac{A_2^2}{(\frac{1}{4}A_1 + \frac{3}{4}A_2)}\right]$$

5	6	7	8	9	10	11	12	13	14	15	16	17	18
$\frac{1}{4}A_1$	$\frac{1}{4}A_2$	$Y_2 - Y_1$	$\frac{Y_2 - Y_1}{3E}$ $\times 10^6$	$2\frac{Y_2 - Y_1}{3E}$	A_1^2	A_2^2	$A_1 A_2$	$\frac{3}{4}A_1 + \frac{1}{4}A_2$ (3)+(6)	$\frac{1}{2}(A_1 + A_2)$ $\frac{(1)+(2)}{2}$	$\frac{1}{4}A_1 + \frac{3}{4}A_2$ (4)+(5)	(10)/(13)	(12)/(14)	(11)/(15)
.56300	.56300	54.5407	1.73144	3.46288	5.0715	5.0715	5.0715	2.2520	2.2520	2.2520	2.2520	2.2520	2.2520
.56300	1.14255	52.6872	1.67260	3.34520	5.0715	20.8867	10.2921	2.8316	3.4111	3.9906	1.7910	3.0172	5.2340
1.14255	1.14255	52.6872	1.67260	3.34520	20.8867	20.8867	20.8867	4.5702	4.5702	4.5702	4.5702	4.5702	4.5702
1.14255	.75000	52.6872	1.67260	3.34520	20.8867	9.0000	13.7106	4.1776	3.7851	3.39255	4.9997	3.6223	2.6529
1.11600	1.11600	57.3798	1.82158	3.64316	19.9273	19.9273	19.9273	4.4640	4.4640	4.4640	4.4640	4.4640	4.4640
1.11600	1.11600	57.3798	1.82158	3.64316	19.9273	19.9273	19.9273	4.4640	4.4640	4.4640	4.4640	4.4640	4.4640
1.11600	1.11600	57.3798	1.82158	3.64316	19.9273	19.9273	19.9273	4.4640	4.4640	4.4640	4.4640	4.4640	4.4640
.2500	.2500	15.0000	.47620	.95240	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000
.2500	.55800	62.0725	1.97056	3.94112	1.0000	4.9812	2.2320	1.30800	1.6160	1.92400	.7648	1.3812	2.5893
.55800	1.02325	62.0725	1.97056	3.94112	4.9812	16.7526	9.1356	2.6572	3.1[illegible]	3.6278	1.8470	2.8557	4.6178
1.02325	.68500	62.0725	1.97056	3.94112	16.7526	7.5076	11.2[illegible]	3.7[illegible]	3.4165	3.0682	4.4616	3.2825	2.4391
$\times 10^{-6}$										Final	rib terms	are	th...

A. V. ROE CANADA LIMITED
MALTON, ONTARIO
ENGINEERING DIVISION — AIRCRAFT

AIRCRAFT	C105
WEIGHT	
C. G. POSITION	
REPORT NO.	7/0510/7
SHEET	3-17
DATE	20 June 55
PREPARED BY	DK [illegible]

17	18	19	20	21	22	23	24	25	26	27	28	29
		KV_{26} $\times 10^5$	KV_{27} $\times 10^5$	KV_{28} $\times 10^5$								
(12)/(14)	(11)/(15)	(9)×(16)	(8)×(17)	(9)×(18)								
2520	2.2520	7.79841	3.89920	7.79841								
.0172	5.2340	5.99125	5.04657	17.50878								
.5702	4.5702	15.28823	7.64412	15.28823								
6223	2.6529	16.72500	6.05870	8.87448								
4640	4.4640	16.26307	8.13153	16.26307								
.4640	4.4640	16.26307	8.13153	16.26307								
.4640	4.4640	16.26307	8.13153	16.26307								
0000	1.0000	.95240	.47620	.95240								
3812	2.5893	3.01299	2.72174	10.20974								
8887	4.6178	7.27925	5.69236	18.19930								
2825	2.4391	17.58370	6.46836	9.61230								

In these calculations the energy expression for a rib segment is taken from report Gen/1030/311 sht. 302 rather than from report 7/0510/2 sht. 2-14.

...ms are tabulated on sht. 3-28

Elastic Influence Coefficients – Centre Section

Revised Values for Panels, Panel Coupling, Poisson's Ratio.

A. V.

AIRCR

WEIGH

C. G.

	1	2	3	4	5	6	7	8	9	10
Panel	* KV_1 $\times 10^6$	** KV_2 $\times 10^6$	** KV_3 $\times 10^6$	KV_{15} $\times 10^6$	KV_{16} $\times 10^6$	KV_{17} $\times 10^6$	KV_{18} $\times 10^6$		Panel KV_1 $\times 10^6$	Coup KV_2 $\times 10^6$
124	185.57524	14.21334	13.05095	$\overline{.43326}$	.94785	$\overline{.56035}$	1.22587		191.76108	14.68712
125	154.67173	10.37834	9.52959	$\overline{1.74712}$	$\overline{.43326}$	$\overline{2.25958}$	$\overline{.56035}$		159.82245	10.72428
126	133.38615	6.54923	6.01366	$\overline{2.76437}$	$\overline{1.74712}$	$\overline{3.57521}$	$\overline{2.25958}$		137.83235	6.76754
136	201.39222	15.37573	14.21340	$\overline{.47562}$	1.04052	$\overline{.60271}$	1.31855		208.11260	15.88825
137	167.86063	11.22714	10.37834	$\overline{1.91794}$	$\overline{.47562}$	$\overline{2.43040}$	$\overline{.60271}$		173.45598	11.60138
138	144.76003	7.08488	6.54924	$\overline{3.03465}$	$\overline{1.91794}$	$\overline{3.84549}$	$\overline{2.43040}$		149.58539	7.32109

Due to the change in panel thickness from .150 in. to .155

factor $\frac{.155}{.150} = 1.0333\dot{3}$

Data in cols 1-7 obtained from shts $\overset{*}{3}$-10, $\overset{**}{3}$-11 & 3-1

Final elements are tabulated on shts. 3-29 & 3-30 Pan

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/7		
SHEET NO 3-19		
AIRCRAFT: C105	PREPARED BY	DATE
Cix	D L Turner	16 June 55
	CHECKED BY	DATE
	R N Sleahy	June 20/55

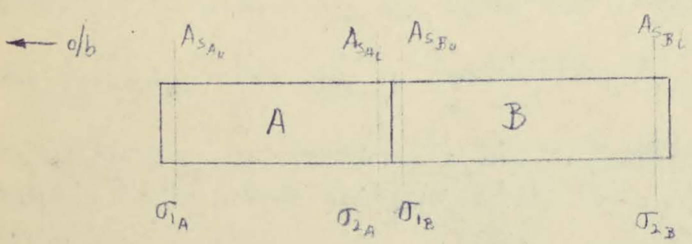
Centre Section - Spars.

Correction to spar elastic influence coefficients to incorporate the effect of the discontinuity in spar cap areas occurring at the fuselage side.

In the calculations of sht. 3-16, two spar areas at the fuselage side were used (per spar) for the determination of spar energy for the spar bays adjacent to the fuselage side. For the bay inboard of the fuselage side, the cap area inboard of the fuselage side was used. For the bay outboard of the fuselage side, the cap area outboard of the fuselage side was used. Since only one stress point exists on each spar at the fuselage side, the energy expressions should be written in terms of one only of the areas.

For the T_{ia} and K_{ip} calculations it was decided to write these stresses in terms of the smaller of the two areas, hence the energy expressions must be corrected to match these conditions.

To develop the required correction factors, consider the case illustrated in the sketch below.



From the basic equations (ref. report 7/0510/2 sht. 2-13)

$$2V = KV_6\,\sigma_{n_2}^2 + 2KV_7\,\sigma_{n_2}\sigma_{n_1} + KV_8\,\sigma_{n_1}^2$$

Centre Section — Spars.

For the above case the basic equation gives:

$$2V = KV_{6A}\sigma_{2A}^2 + 2KV_{7A}\sigma_{2A}\sigma_{1A} + KV_{8A}\sigma_{1A}^2 + KV_{6B}\sigma_{2B}^2 + 2KV_{7B}\sigma_{2B}\sigma_{1B} + KV_{8B}\sigma_{1B}^2$$

Now it is required to replace σ_{2A} by σ_{1B} or vice versa. Since the cap end load does not change across the boundary between A & B

$$\sigma_{2A} A_{S_{Ai}} = \sigma_{1B} A_{S_{Bo}}$$

hence $\sigma_{1B} = \sigma_{2A}\dfrac{A_{S_{Ai}}}{A_{S_{Bo}}}$ $\quad \sigma_{1B}^2 = \sigma_{2A}^2\left(\dfrac{A_{S_{Ai}}}{A_{S_{Bo}}}\right)^2$ $\quad \sigma_{1B}\sigma_{2B} = \sigma_{2A}\sigma_{2B}\dfrac{A_{S_{Ai}}}{A_{S_{Bo}}}$

or $\sigma_{2A} = \sigma_{1B}\dfrac{A_{S_{Bo}}}{A_{S_{Ai}}}$ $\quad \sigma_{2A}^2 = \sigma_{1B}^2\left(\dfrac{A_{S_{Bo}}}{A_{S_{Ai}}}\right)^2$ $\quad \sigma_{2A}\sigma_{1A} = \sigma_{1B}\sigma_{1A}\dfrac{A_{S_{Bo}}}{A_{S_{Ai}}}$

From these relations the following equations derive:

$$2V = KV_{8A}\sigma_{1A}^2 + 2KV_{7A}\sigma_{2A}\sigma_{1A} + \left(KV_{6A} + KV_{8B}\frac{A_{S_{Ai}}^2}{A_{S_{Bo}}^2}\right)\sigma_{2A}^2 + 2KV_{7B}\frac{A_{S_{Ai}}}{A_{S_{Bo}}}\sigma_{2A}\sigma_{2B} + KV_{6B}\sigma_{2B}^2$$

For the case where $A_{S_{Ai}}$ is smaller than $A_{S_{Bo}}$

$$2V = \left(KV_{6A}\frac{A_{S_{Bo}}^2}{A_{S_{Ai}}^2} + KV_{8B}\right)\sigma_{1B}^2 + 2KV_{7A}\frac{A_{S_{Bo}}}{A_{S_{Ai}}}\sigma_{1B}\sigma_{1A} + KV_{8A}\sigma_{1A}^2 + KV_{6B}\sigma_{2B}^2 + 2KV_{7B}\sigma_{2B}\sigma_{1B}$$

For the case where $A_{S_{Bo}}$ is smaller than $A_{S_{Ai}}$

Corrected spar coefficients are calculated on sht. 3-21 according to the above relationships.

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/7

SHEET NO 3-21

AIRCRAFT		PREPARED BY	DATE
C105	Cix	D L Turner	16 June '55
		CHECKED BY	DATE
		R N Glendy	June 20/55

Centre Section – Spars

Correction to spar elastic influence coefficients due to discontinuity in cap areas at fuselage side.

Spar Bay	1 $A_{S_{Ai}}$	2 $A_{S_{Bo}}$	3 $\frac{A_{S_{Ai}}}{A_{S_{Bo}}}$	4 $(3)^2$	5 * KV_{8B} $\times 10^6$	6 * KV_{7B} $\times 10^6$	7 Coefficients of σ_{2A}^2 $(5)\times(4)$	8 $\sigma_{2A}\sigma_{2B}$ $(6)\times(3)$
118-129	8.5560	8.9901	.95171	.90575	17.31624	9.69110	15.68418	9.22312
120-131	8.7727	9.6846	.90584	.82055	17.56775	9.83181	14.41522	8.90605
122-133	5.9348	6.4080	.92615	.85775	11.14632	6.18573	9.56076	5.72891

For $A_{S_{Ai}}$ less than $A_{S_{Bo}}$ * ref. sht.

Spar Bay	1 $A_{S_{Ai}}$	2 $A_{S_{Bo}}$	3 $\frac{A_{S_{Bo}}}{A_{S_{Ai}}}$	4 $(3)^2$	5 † KV_{6A} $\times 10^6$	6 † KV_{7A} $\times 10^6$	7 σ_{1B}^2 $(5)\times(4)$	8 $\sigma_{1A}\sigma_{1B}$ $(6)\times(3)$
107-117	5.9333	5.5293	.93191	.86846	15.2759	6.9257	13.26651	6.45413

For $A_{S_{Bo}}$ less than $A_{S_{Ai}}$ † ref. sht. 3-7

Final elements of the Cix matrix are determined on sht. 3-27

FORM 1119A

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/7	SHEET NO. 3-22	
AIRCRAFT: C105	PREPARED BY: D L Turner	DATE: 16 June '55
C_{ik}	CHECKED BY: R.N. Shearly	DATE: JUNE 20/55

Centre Section — Spar-Panel Coupling

Since the spar-panel coupling energy in this region has been calculated (sht 3-18) on the assumption that spar stresses are based on the "two spar areas at the fuselage side" concept (ref sht. 3-19) these terms will have to be corrected to read in terms of the stress based on the smaller area.

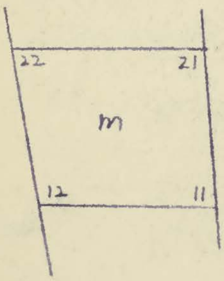
The correction is determined as follows:

From the basic equations (ref. report 7/0510/2 sht 2-13)

$$2V = 2KV_2 J_m(\sigma_{n_{22}} + \sigma_{n_{21}}) + 2KV_3 J_m(\sigma_{n_{12}} + \sigma_{n_{11}}) + 2KV_2 J_m(\overset{+}{\sigma}_{y_{22}} + \overset{+}{\sigma}_{y_{21}})$$

$$+ 2KV_3 J_m(\overset{+}{\sigma}_{y_{12}} + \overset{+}{\sigma}_{y_{11}})$$

where the subscripts are defined in the sketch below:



From this it can be seen that for the case where A_{s_o} is smaller than A_{s_i} the coupling terms 11 & 12 must be corrected in the panel inboard of the fuselage side:

i.e. term to be corrected is $2KV_3 J_{m_i}(\sigma_{n_{12}} + \sigma_{n_{11}})$

that for the case where A_{s_i} is smaller than A_{s_o} the coupling terms 22 & 21 of the panel outboard of the fuselage side must be corrected:

Centre Section – Spar-Panel Coupling

i.e. term to be corrected is $2KV_2 J_{m_o} (\sigma_{n_{22}} + \sigma_{n_{21}})$

and that for the mixed case (i.e. panels 114 & 124) one term from each panel must be corrected.

i.e. (in the case in question) terms $2KV_3 J_{m_i} \sigma_{n_{11}}$ (panel 124)

$2KV_2 J_{m_o} \sigma_{n_{22}}$ (panel 114)

The first & third cases only apply here.

From the work of sheet 3-20, for the first case (involving panels 125 & 126

$\sigma_{12_i} = \sigma_{12_o} \dfrac{A_{s_{12_o}}}{A_{s_{12_i}}}$ $\sigma_{11_i} = \sigma_{11_o} \dfrac{A_{s_{11_o}}}{A_{s_{11_i}}}$ (both panels)

For the third case (involving panels 114 & 124)

$\sigma_{n_{11_i}} = \sigma_{11_o} \dfrac{A_{s_{11_o}}}{A_{s_{11_i}}}$ panel 124

$\sigma_{n_{22_o}} = \sigma_{n_{22_i}} \dfrac{A_{s_{22_i}}}{A_{s_{22_o}}}$ panel 114

From this the table on sht. 3-24 may be set up to determine the corrected coupling elements.

AVRO AIRCRAFT LIMITED TECHNICAL DEPARTMENT (Aircraft)		REPORT NO. 7/0510/7 SHEET NO. 3-25	
AIRCRAFT: C105	Cix	PREPARED BY: D L Turner	DATE: 17 June 55
		CHECKED BY: R N Sheardy	DATE: June 20/55

Centre Section – Poisson's Ratio

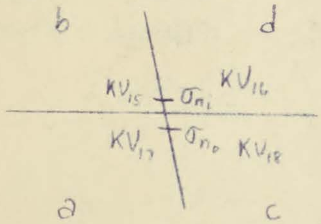
Since the Poisson's ratio energy terms in this region have been calculated (sht 3-18) on the assumption that spar stresses are based on the existence of two spar areas at the fuselage side (ref. sht. 3-19), these terms will have to be corrected to read in terms of stress based on the smaller area.

The correction is determined as follows:

From the basic equations:

$$2V = 2\sigma_n \overset{+}{\sigma}_y \left(KV_{17_a} + KV_{15_b} + KV_{18_c} + KV_{16_d} \right) \quad \text{ref. 7/0510/2 sht. 1-67}$$

a, b, c & d are panels



As determined on sht. 3-18

$$2V = 2\sigma_{n_o} \overset{+}{\sigma}_y \left(KV_{17_a} + KV_{18_c} \right) + 2\sigma_{n_i} \overset{+}{\sigma}_y \left(KV_{15_b} + KV_{16_d} \right)$$

From this, for A_{s_i} smaller than A_{s_o} (element 117-127)

$$2V = 2\sigma_{n_i} \overset{+}{\sigma}_y \frac{A_{s_i}}{A_{s_o}} \left(KV_{17_a} + KV_{18_c} \right) + 2\sigma_{n_i} \overset{+}{\sigma}_y \left(KV_{15_b} + KV_{16_d} \right) \quad \text{since } \sigma_{n_o} = \sigma_{n_i} \frac{A_{s_i}}{A_{s_o}}$$

and, for A_{s_o} smaller than A_{s_i} (elements 118-119, 120-121, 122-123)

$$2V = 2\sigma_{n_o} \overset{+}{\sigma}_y \left(KV_{17_a} + KV_{18_c} \right) + 2\sigma_{n_o} \overset{+}{\sigma}_y \frac{A_{s_o}}{A_{s_i}} \left(KV_{15_b} + KV_{16_d} \right) \quad \text{since } \sigma_{n_i} = \sigma_{n_o} \frac{A_{s_o}}{A_{s_i}}$$

These corrections are made and the elements are summarized on sht. 3-30

FORM 1719A

10

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO 7/0510/7
SHEET NO 3-26

AIRCRAFT		PREPARED BY	DATE
C105	C_{ik}	D L Turner	10 June 55
		CHECKED BY	DATE
		R D Clark	June 20/55

Inner Wing Poisson's Ratio Terms Required for Centre Section Calculations.

These figures are derived from report 7/0510/2 sht. 2-21 and are factored to correct for the change in innerwing skin thickness.

Correction factor is $\frac{\text{new } t}{\text{old } t} = \frac{.150}{.204} = .73529411$

ref. sht. 3-6

	Old Coeff.		New Coeff.	
Panel	KV_{17} $\times 10^6$	KV_{18} $\times 10^6$	KV_{17} $\times 10^6$	KV_{18} $\times 10^6$
114	-.76776	1.67964	-.56453	1.23503
115	-3.09600	-.76776	-2.27647	-.56453
116	-4.89864	-3.09600	-3.60194	-2.27647

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

REPORT NO. - 7/0-[illegible]
SHEET - - 3-27
DATE - - 10 [illegible]

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

PREPARED BY - [illegible]

7	8	9	10	11	12	13	14	15	16	17	18	19
$KV_{6\,3/4}$ ×10⁶	(6)+(7)	Cross Product Element ×10⁶	Element	$KV_{8\,3/4}$ ×10⁶	$KV_{6\,3/4}$ ×10⁶	(11)+(12)	Cross Product Element ×10⁶	Element	KV ×10⁶	KV ×10⁶	[illegible]	Cross Product Element ×10⁶
20.4993	36.15348	—	120-120	14.41522	19.7947	34.20952	—	122-122	9.56076	12.[illegible]	22.40166	
—	—	9.22312	120-131	—	—	—	8.96605	122-133	—	—		5.72891
21.44779	40.41664	—	131-131	19.24437	21.75932	41.00369	—	133-133	12.01388	13.58393	25.59781	
—	—	10.51755	131-146	—	—	—	10.67016	133-[illegible]				6.61304
23.10037	23.10037	—	146-146	—	23.43612	23.43612	—	149-149	—	14.42769	14.42769	

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/7
SHEET NO. 3-29

AIRCRAFT		PREPARED BY	DATE
C 105	Ci.x	D L Demers	13 June 55
		CHECKED BY	DATE
		R. N. Henry	June 20/55

Centre Section - Coupling

Summary of Coupling Elements.

	$KV \times 10^6$		$KV \times 10^6$
124-127	13.48598	136-128	14.68718
124-117	13.48598	136-141	15.88825
124-128	14.68712	136-140	15.88825
124-130	14.68712	136-142	15.88825
124-129	14.68712	136-143	15.88825
124-119	13.48598	136-130	14.68718
124-118	12.83474	136-129	14.68718
125-119	9.84724	137-130	10.72428
125-118	9.37172	137-129	10.72428
125-130	10.72428	137-142	11.60138
125-129	10.72428	137-143	11.60138
125-132	10.72428	137-145	11.60138
125-131	10.72428	137-146	11.60138
125-121	9.84724	137-132	10.72428
125-120	8.92002	137-131	10.72428
126-121	6.21412	138-132	6.76755
126-120	5.62900	138-131	6.76755
126-132	6.76754	138-145	7.32104
126-131	6.76754	138-146	7.32104
126-134	6.76754	138-148	7.32104
126-133	6.76754	138-149	7.32104
126-123	6.21412	138-134	6.76755
126-122	5.75521	138-133	6.76755
114-117	13.16665 ←		

FORM 1319A

Introduction

In this section of the report the redundant stress coefficients for the complete wing are calculated.

No redundancies occur in the wing tip analysis, hence there are no wing tip calculations here.

Since there were no section property changes in the outer wing, elements of the Kip matrix may be taken directly from report 7/0510/2. However it should be noted that with the removal of the outboard bay from the outer wing, redundancies 1-7, inclusive, of report 7/0510/2 no longer exist.

With the addition of some flexibility for the transport joint triangle (cell 83), a further redundancy has been added on the outer wing face of this triangle. This redundancy has been given the index 7, and coefficients for it are calculated in this section.

Redundant stresses at the transport joint are revised in accordance with the revised approach to this problem given in Section 7.

Elements of the inner wing and centre section wing (other than those mentioned above) are obtained by factoring the figures taken from report 7/0510/2 to account for the changes in skin thickness and rib & spar cap areas.

Note that redundant stresses at the fuselage centre line require special treatment.

Since the completion of the work of this section, several changes have been made in panel thicknesses and spar and rib cap areas. As a result, the matrix derived from these calculations must be factored by an appropriate set of correction factors before being used. These factors are calculated and tabulated in section 6.

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO 7/0510/7

SHEET NO 4-2

AIRCRAFT: C105	Kip	PREPARED BY: D L Turner	DATE: 17 June 55
		CHECKED BY	DATE

Redundant Stress Coefficients

It should be noted that the redefinition of the problem (to wing only) while work was in process cancelled some stress points and redundancies. Where possible work involving these was removed from the report, but this was not always possible. Hence reference to non existent indices may be found throughout this section. See fig. 8 for the final numbering code.

FORM 131PA

A. V ROE CANADA LIMITED MALTON - ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT No. 7/0510/7 SHEET NO 4-4	
		PREPARED BY	DATE
AIRCRAFT: C-105	Kip	R.N. SHEARLY	JAN. 21/55
		CHECKED BY	DATE
		[signature]	28 April '55

REDUNDANT GROUP MF_7 (CONT.)

$$\sigma_{70} = (K S'_2)_{70}\, MF_7$$

$$= .042317\, MF_7$$ REF REPORT 7/0510/2 SHT 1-52

$$\sigma_{72} = -\left(\frac{1}{h \cos\theta A_s}\right)_{72} MF_7 + \text{A TERM FOR } MF_{33}$$

$$= (K S'_2)_{72}\, MF_7 + \cdots$$

$$= -.018200\, MF_7 +$$ REF. REPORT 7/0510/2 SHT 1-52

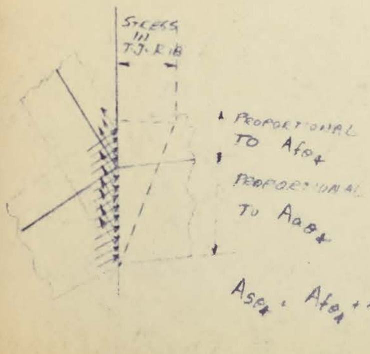
THE EFFECTS OF REDUNDANT GROUP MF_7 ON STRESS POINTS $i = 71$ & 74 WILL BE CONSIDERED BELOW

$$\sigma_{71} = \left(\frac{\cot\alpha - \tan\theta_4}{h A_r}\right)_{71} MF_7 + \text{A TERM FOR } MF_{32}$$

see sketch on previous sheet.

$$= \frac{\frac{1}{.824445} - .26699}{9.4312 \times 4.6468} MF_7 + \cdots$$

$$= .021585\, MF_7 + \cdots$$



DUE TO THE SPREADING OF THE LUMPED SPAR & RIB AREAS (OUTER WING) THE STRESS AT RIB POINT $i = 74$ IS AS FOLLOWS

$$\sigma_{74} = \left(1 - \frac{A_{f\theta_4}}{A_{s\theta_4}}\right) \frac{M_{74}^*}{(h A_r)_{74}} = .650739 \frac{M_{74}^*}{(h A_r)_{74}}$$

(SEE FIGURE AT LEFT)

M_{74}^* IS THE PORTION OF M_{74} DUE TO MF_7

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

AIRCRAFT:		REPORT NO. 7/0510/7	
C-105	Kip	SHEET NO. 4-6	
		PREPARED BY	DATE
		R.N. SHEARLY	JAN 21/55
		CHECKED BY	DATE
		[illegible] P.M.J.	28 April 55

REDUNDANT STRESSES AT THE T.J. RIB BETWEEN INNER & OUTER WINGS

WITH THE NUMBERING SYSTEM, USED IN THIS REPORT, OF THE REDUNDANT GROUPS AT THE T.J. THE STRESSES AT ALL STRESS POINTS OF THE OUTER WING COORDINATE SYSTEM (i = 14 TO 72) ARE FOUND IN THE REGULAR MANNER FOR THE OUTER WING (SEE SHEET 1-39 REPORT 7/0510/2) WITH THE EXCEPTION OF THE STRESSES PRODUCED BY THE REDUNDANT GROUP MF_7. THESE LATTER STRESSES WERE DEALT WITH EARLIER IN THIS SECTION. SEE SHEETS 4-3 & 4-4 & 4-4a.

TO CALCULATE THE STRESSES IN THE SPAR & RIB STRESS POINTS i = 73, 74, 75, 76, 78, 79, 81 & 82 THE FOLLOWING TRANSFORMATION MUST BE MADE.

$$MH = \frac{h_{11} b_2}{h_{11} b_2 + h_{22} b_1} \frac{MF_{x_2}}{\cos\theta_2} \cos(\alpha+\theta_2) + \frac{h_{22} b_1}{h_{11} b_2 + h_{22} b_1} \frac{MF_{x_1}}{\cos\theta_1} \cos(\alpha+\theta_1) +$$

$$+ \frac{h_{11} b_2}{h_{11} b_2 + h_{22} b_1} \frac{MF_{\theta_2}}{\cos\theta_2} \sin\alpha + \frac{h_{22} b_1}{h_{11} b_2 + h_{22} b_1} \frac{MF_{\theta_1}}{\cos\theta_1} \sin\alpha$$

$$MR_2 = \frac{A_{F\theta_2}}{A_{S\theta_2}} \left\{ - \frac{MF_{x_2}}{\cos\theta_2} \sin(\alpha+\theta_2) + \frac{MF_{\theta_2}}{\cos\theta_2} \cos\alpha \right\}$$

$$MR_1 = \frac{A_{F\theta_1}}{A_{S\theta_1}} \left\{ - \frac{MF_{x_1}}{\cos\theta_1} \sin(\alpha+\theta_1) + \frac{MF_{\theta_1}}{\cos\theta_1} \cos\alpha \right\}$$

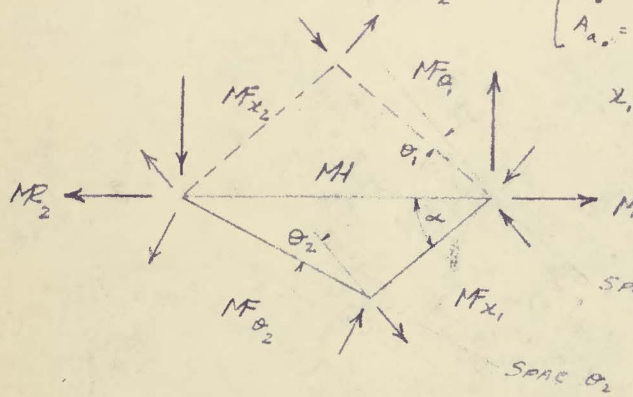
REF.

SEE FIGURE ON NEXT SHEET FOR DEFINITIONS

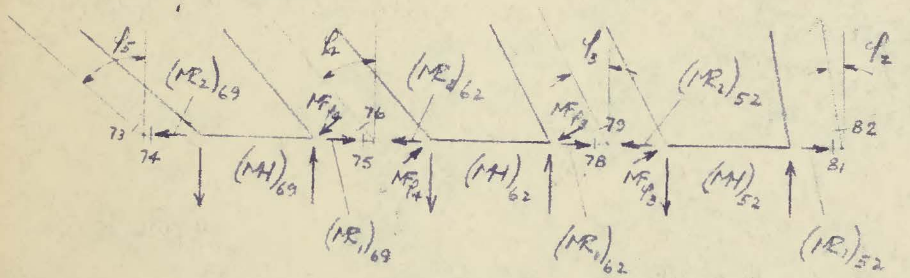
A. V. ROE CANADA LIMITED
MALTON - ONTARIO
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO.	7/0510/7
SHEET NO.	4-7
AIRCRAFT: C-105	Kip
PREPARED BY: R.N. SHEARLY	DATE: JAN. 21/55
CHECKED BY: [signature]	DATE: 28 April '55

AT OBLIQUE RIB
$A_{s_\bullet}$ = SPAR AREA OF SPAR •
$A_{f_\bullet}$ = SPAR AREA FWD OF WEB OF SPAR •
$A_{a_\bullet}$ = SPAR AREA AFT OF WEB OF SPAR •



THESE REDUNDANT GROUPS MH, MR_2 & MR_1 PRODUCE STRESSES AS OUTLINED BELOW



STRESS IN A TYPICAL SPAR STRESS POINT

$$\sigma_m = \frac{1}{h \cos\varphi\, A_s}\left\{ (MH)_{\varphi_-} - (MH)_{\varphi_+} \right\}$$

STRESS IN A TYPICAL RIB STRESS POINT

$$\overset{+}{\sigma}_y = \frac{1}{h\, A_r}\left[(MR_2)_{\varphi_-} + (MR_1)_{\varphi_+} + \frac{\tan\varphi}{A_s}\left\{ A_f (MH)_{\varphi_-} + A_a (MH)_{\varphi_+} \right\} - \frac{1}{\cos\varphi} MF_\varphi \right]$$

WHERE THE EFFECT OF THE MH GROUPS ARE ACCORDING TO THE FOLLOWING TYPICAL INTERPRETION

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

REPORT NO. 7/0510/7		
SHEET NO. 4-9		
AIRCRAFT: C-105	KIP	
PREPARED BY	DATE	
R.N. SHEARLY	JAN. 21/55	
CHECKED BY	DATE	
D. Dunn R.N.S.	28 April '55	

STRESS IN A TYPICAL PANEL STRESS POINT OF INNER WING IS FOUND IN THE REGULAR MANNER FOR THE INNER WING USING THE APPROPRIATE MH GROUP FOR THE M_{X_1} GROUP AND CONSIDERING THAT THE MR GROUP AFFECTS THE RIB ONLY.

A. V. ROE

MA[illegible]

TECHNI[illegible]

AIRCRAFT -

WEIGHT -

C. G. POSITION

CALCULATIONS OF COEFFICIENTS RELATING THE REDUNDANT GROUPS M_1, MR_2 & MR_1 WITH MF_{12}, MF_{41}, MF_{θ_2} & MF_{θ_1}.

$\alpha = 39.5037°$
$\sin\alpha = .636128$
$\cos\alpha = .771584$
} *

See Sheet 4-10 For Equations

PANEL	1	2	3	4	5	6	7	8	9	10	11
	h_{11} *	h_{22} *	b_1 *	b_2 *	$h_{11}b_2$ ① × ④	$h_{22}b_1$ ② × ③	$h_{11}b_2+h_{22}b_1$ ⑤ + ⑥	$\frac{h_{11}b_2}{h_{11}b_2+h_{22}b_1}$ ⑤/⑦	$\frac{h_{22}b_1}{h_{11}b_2-h_{22}b_1}$ ⑥/⑦	θ_2 *	θ_1 *
52	6.4011	8.3415	25.2174	27.9143	178.682	210.351	389.033	.459298	.540702	4.32°	-2.09°
62	8.3415	9.3858	22.9984	25.2204	210.376	215.858	426.234	.493569	.506431	10.41°	4.82°
69	9.3858	9.4312	21.1328	22.9946	215.823	199.308	415.131	.519891	.480109	14.95°	10.41°

PANEL	20	21	22	23	24	25	26	27	28	29	30
	$\frac{h_{11}b_2 \frac{1}{\cos\theta_2}}{h_{11}b_2-h_{22}b_1}$	$\frac{h_{22}b_1 \frac{1}{\cos\theta_1}}{h_{11}b_2+h_{22}b_1}$	$\frac{A_{f_2}}{A_{s_2}\cos\theta_2}$	$\frac{A_{s_1}}{A_{s_1}\cos\theta_1}$	KJ_1	KJ_2	KJ_3	KJ_4	KJ_5	KJ_6	KJ_7
	⑧/⑭	⑨/⑮	$\frac{A_{f_2}}{A_{s_2}⑭}$	$\frac{A_{s_1}}{A_{s_1}⑮}$	⑳ × ⑯	㉑ × ⑰	⑳ × .636128	㉑ × .636128	-㉒ × ⑱	㉒ × .771584	-㉓ × ⑲
52	.460928	.541062	.501775	.211282	.329752	.429751	.293209	.344185	-.350574	.387162	-.128367
62	.501829	.508229	.508368	.501775	.323151	.363591	.319227	.323299	-.388938	.392249	-.350574
69	.538105	.488144	.361497	.508368	.312836	.314338	.342304	.310522	-.294129	.278925	-.388938

* REF. REPORT 7/0510/2 SHT 1-53

THE RECURRENT ...
..., M_{θ_2} & M_{θ_1}.

... 4-10 FOR EQUATIONS

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

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

REPORT NO. 7/0...
SHEET 4-12
DATE JAN 13/55
PREPARED BY - R N S...

7	8	9	10	11	12	13	14	15	16	17	18	19
$h_{11}b_2 + h_{22}b_1$	$\frac{h_{11}b_2}{h_{11}b_2+h_{22}b_1}$	$\frac{h_{22}b_1}{h_{11}b_2+h_{22}b_1}$	θ_2	θ_1	$\alpha+\theta_2$	$\alpha+\theta_1$	$\cos\theta_2$	$\cos\theta_1$	$\cos(\alpha+\theta_2)$	$\cos(\alpha+\theta_1)$	$\sin(\alpha+\theta_2)$	$\sin(\alpha+\theta_1)$
5 + 6	5/7	6/7	*	*	*	*	*	*	*	*	*	*
387.033	.459298	.540702	4.32°	-2.03°	44.32°	37.41°	.996463	.999325	.715402	.794373	.698707	.607561
426.234	.493569	.506431	10.41°	4.32°	42.91°	44.32°	.983540	.996463	.643946	.715402	.765...	.698707
415.131	.519891	.480109	14.95°	10.41°	54.45°	42.91°	.966151	.983540	.581366	.643946	.83642	.765071

26	27	28	29	30	31	32	33	34	35	36	37	38
KJ_3	KJ_4	KJ_5	KJ_6	KJ_7	KJ_8							
20 × .636128	21 × .636128	− 22 × 18	22 × .711584	− 23 × 19	23 × .711584							
.393207	.344185	−.350574	.387162	−.128367	.163022							
.319227	.323299	−.388938	.392249	−.350574	.387162							
.342304	.310522	−.294129	.278925	−.388938	.392249							

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

REPORT No. 7/0510/7		
SHEET No. 4-13		
AIRCRAFT: C-105	PREPARED BY: R.N. SHEARLY	DATE: JAN. 20/55
Kip	CHECKED BY: [illegible]	DATE: 29 April 55

REDUNDANT GROUPS MH, MR₂ & MR₁ IN TERMS OF
REDUNDANT GROUPS MFx & MFB

$(MH)_{52} = .329752\ MF_{37} + .429751\ MF_{21} + .293209\ MF_{24} + .344185\ MF_{38}$

$(MH)_{62} = .323151\ MF_{35} + .363591\ MF_{27} + .319227\ MF_{29} + .323299\ MF_{36}$

$(MH)_{69} = .312836\ MF_{33} + .314338\ MF_{31} + .342304\ MF_{32} + .310522\ MF_{34}$

$(MR_2)_{52} = -.350594\ MF_{37} + .387162\ MF_{24}$

$(MR_1)_{52} = -.128367\ MF_{21} + .163022\ MF_{38}$

$(MR_2)_{62} = -.388938\ MF_{35} + .392249\ MF_{29}$

$(MR_1)_{62} = -.350594\ MF_{27} + .387162\ MF_{36}$

$(MR_2)_{69} = -.294129\ MF_{33} + .278925\ MF_{32}$

$(MR_1)_{69} = -.388938\ MF_{31} + .392249\ MF_{34}$

COEFFICIENTS FROM SHT. 4-12

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

REPORT NO.	7/0510/7
SHEET NO.	4-14
PREPARED BY	DATE
R.N. SHEARLY	JAN. 19/55
CHECKED BY	DATE
[illegible] R.N.S.	29 April '55

AIRCRAFT: C-105

Kip

CALCULATIONS OF THE COEFFICIENTS RELATING SPAR STRESSES AT THE OBLIQUE RIB (AT SPAR SPOTS i = 73, 76, 79, & 82) IN TERMS OF THE REDUNDANT GROUPS $(MH)_{ii}$ & $(MH)_{i,i+1}$

SEE SHT 4-10 FOR EQUATIONS

	1	2	3	4	5	6
SPAR SPOT	h	$\cos\phi$	$h\cos\phi$ ① × ②	A_s	$h\cos\phi A_s$ ③ × ④	KK_1 1/⑤
73	9.4312	.824438	7.7754	6.0210	46.8157	.0213604
76	9.3858	.885834	8.3152	5.7035	47.4257	.0210856
79	8.3414	.940707	7.8468	5.8480	45.8881	.0217921
82	6.4011	.981019	6.2796	4.4141	27.7188	.0360766

ref. Fig. 6

NOTE $KK_2 = -KK_1$

Calculation Of The Coefficients Relating Rib Stresses Of The Oblique Rib (At Rib Spots $i = 74, 75, 78$ & 81) In Terms Of The Redundant Groups $(MR)_{\phi-}$, $(MR)_{\phi+}$, $(MH)_{\phi-}$, $(MH)_{\phi+}$ & $MF\phi x_{i+}$.

A. V. ROE CA
MALTO
TECHNICAL

AIRCRAFT -
WEIGHT -
C. G. POSITION -

	1	2	3	4	5	6	7	8	9	10	11	1
Rib Spot	h	A_r	ϕ	$\cos\phi$	$\sin\phi$	A_s	A_f	A_R	A_f/A_s ⑦/⑥	A_R/A_s ⑧/⑥	$h A_r$ ① × ②	$\frac{A_f}{A_s}$ ③
74	9.4312	10.6039	34°-28.11′	.824438	.686472	6.0210	2.1029 *	3.9181 *	.34926	.65074	100.0066	.23
75	9.3858	7.7888	27°-38.02′	.885934	.523536	5.7035	2.85175	2.85175	.5	.5	73.1041	.26
78	8.3414	6.7288	19°-40.76′	.940707	.360600	5.8480	2.9240	2.9240	.5	.5	56.1276	.18
81	6.4011	6.2638	11°-10.87′	.981019	.197664	4.4141	3.4821 *	.9320 *	.78886	.21114	40.0952	.15
	* Reference Sht. 4-11											

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

REPORT NO. 7/05 0/2	SHEET NO. 4-16	
PREPARED BY: R.N. SHERRIN	DATE: JAN 19/55	
CHECKED BY: [illegible]	DATE: 29 April 55	
AIRCRAFT: C-105	K_{ILP}	

STRESSES AT SPAR SPOTS i = 73, 76, 77 & 82 IN TERMS OF REDUNDANT GROUPS M

$\sigma_{73} = .0213604\ (M)_{69}$

$\sigma_{76} = .0210956\ (M)_{62} - .0210956\ (M)_{69}$

$\sigma_{77} = .0217921\ (M)_{52} - .0217921\ (M)_{62}$

$\sigma_{82} = \quad -.0360766\ (M)_{52}$

REF. SH. 4-14

STRESSES AT RIB SPOTS i = 74, 75, 78 & 81 IN TERMS OF REDUNDANT GROUPS M, MR & MF

$\sigma_{74} = .00999934\ (MR_2)_{69} + .00239741 \times (M)_{69}$

$\sigma_{75} = .0136791 \{(MR_2)_{62} + (MR_1)_{69}\} + .00358075 \{(M)_{62} + (M)_{69}\} - .0154403\ MF_{33}$

$\sigma_{78} = .0178165 \{(MR_2)_{52} + (MR_1)_{62}\} + .00321231 \{(M)_{52} + (M)_{62}\} - .01893951\ MF_{40}$

$\sigma_{81} = .0249406\ (MR_1)_{52} + .0054090\ (M)_{52}$

REF. SHT. 4-15

STRESSES AT PANEL SPOTS i = 84, 85 & 86 IN TERMS OF REDUNDANT GROUPS M

* REF. REPORT 7/0510/2 SHT. 2-16

$\tau_{84} = -\left(\frac{b_2}{a t u'}\right)_{84} (M)_{69} + = -.0113583^* \times \frac{.204}{.150} (M)_{69} + = -.0154473\ (M)_{69} +$

$\tau_{85} = -\left(\frac{b_2}{a t u'}\right)_{85} (M)_{62} + = -.0123078^* \times \frac{.204}{.150} (M)_{62} + = -.0167413\ (M)_{62} +$

$\tau_{86} = -\left(\frac{b_2}{a t u'}\right)_{86} (M)_{52} + = -.0150559^* \times \frac{.204}{.150} (M)_{52} + = -.0204760\ (M)_{52} +$

FORM 1319A

COMBINING THE INFORMATION ON SHTS 4-13 & 4-16 YIELDS STRESSES AT SPAR SPOTS $i = 73, 76, 79$ & 82 IN TERMS OF REDUNDANT GROUPS MF_{x_0}'s & MF_{θ}'s

$$\sigma_{73} = .0213604(.312836\,MF_{33} + .314338\,MF_{31} + .342304\,MF_{32} + .310522\,MF_{34})$$

$$\begin{aligned}\sigma_{76} = {} & .0210856(.323151\,MF_{35} + .363591\,MF_{27} + .319227\,MF_{29} + .323299\,MF_{36}) - \\ & -.0210856(.312836\,MF_{33} + .314338\,MF_{31} + .342304\,MF_{32} + .310522\,MF_{34})\end{aligned}$$

$$\begin{aligned}\sigma_{79} = {} & .0217921(.329752\,MF_{37} + .429751\,MF_{21} + .293209\,MF_{24} + .344185\,MF_{38}) - \\ & -.0217921(.323151\,MF_{35} + .363591\,MF_{27} + .319227\,MF_{29} + .323299\,MF_{36})\end{aligned}$$

$$\sigma_{82} = -.0360766(.329752\,MF_{37} + .429751\,MF_{21} + .293209\,MF_{24} + .344185\,MF_{38})$$

STRESSES AT RIB SPOTS $i = 74, 75, 78$ & 81 IN TERMS OF REDUNDANT GROUPS MF_{x_0}'s, MF_{θ}'s & MF_{ϕ}'s

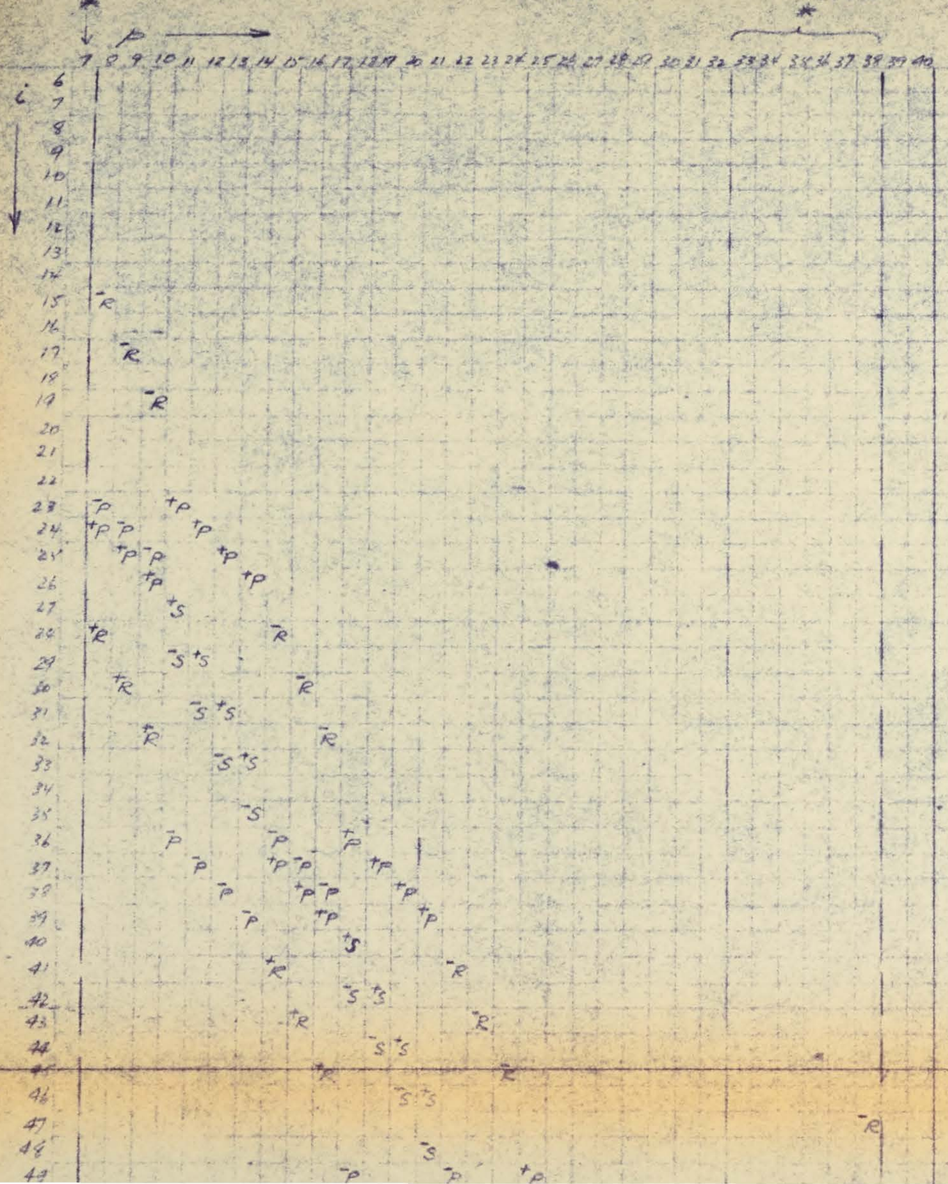
$$\begin{aligned}\sigma_{74} = {} & .00999934(-.294129\,MF_{33} + .278925\,MF_{32}) + \\ & + .00239741 \times (.312836\,MF_{33} + .314338\,MF_{31} + .342304\,MF_{32} + .310522\,MF_{34})\end{aligned}$$

$$\begin{aligned}\sigma_{75} = {} & .0136791(-.388938\,MF_{35} + .392249\,MF_{29} - .388938\,MF_{31} + .392249\,MF_{34}) + \\ & + .00358075(.323151\,MF_{35} + .363591\,MF_{27} + .319227\,MF_{29} + .323299\,MF_{36}) + \\ & + .00358075(.312836\,MF_{33} + .314338\,MF_{31} + .342304\,MF_{32} + .310522\,MF_{34}) - \\ & - .0154408\,MF_{39}\end{aligned}$$

$$\begin{aligned}\sigma_{78} = {} & .0178165(-.350594\,MF_{37} + .387162\,MF_{24} - .350594\,MF_{27} + .387162\,MF_{36}) + \\ & + .00321231(.329752\,MF_{37} + .429751\,MF_{21} + .293209\,MF_{24} + .344185\,MF_{38}) + \\ & + .00321231(.323151\,MF_{35} + .363591\,MF_{27} + .319227\,MF_{29} + .323299\,MF_{36}) - \\ & - .0189395\,MF_{40}\end{aligned}$$

$$\begin{aligned}\sigma_{81} = {} & .0249406(-.128367\,MF_{21} + .163022\,MF_{38}) + \\ & + .00104090(.329752\,MF_{37} + .429751\,MF_{21} + .293209\,MF_{24} + .344185\,MF_{38})\end{aligned}$$

DUMMY OF PORTION OF K_{ip} MATRIX



CODE

S SPAR TERMS } SIGNS NOTED
R RIB TERMS
P PANEL TERMS

• NEW TERMS FOR THIS REPORT

* THESE VALUES OF p ARE NEW FOR THIS REPORT

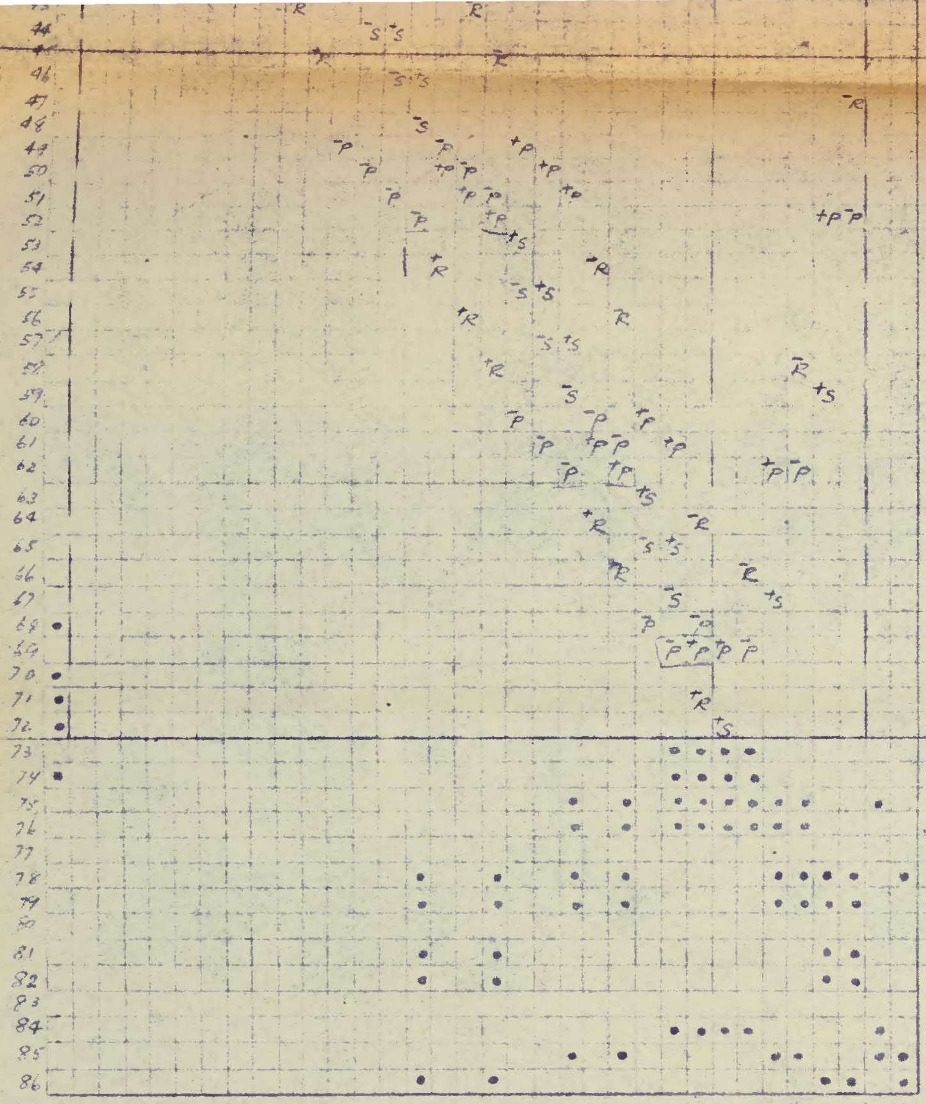
SOURCE OF COEFFICIENTS

(1) THE COEFFICIENTS IN THE PORTION $i = 6$ TO 72 & $p = 8$ TO 32 ARE TAKEN FROM THE K_{ip} MATRIX IN REPORT 7/0510/2
NOTE! THE FOLLOWING COEFFICIENTS ALTHOUGH APPEARING IN THE MATRIX OF REPORT 7/0510/2 DO NOT APPEAR AT LEFT DUE TO THE REVISED NUMBERING OF THE REDUNDANT GROUPS AT THE T.J. RIB.

	K_{ip} FOR i =	& p =
	47	24
	58	29
	59	21, 24
{SEE NOTE (4)} •	66	32
	67	27, 29
	72	31, 32

(2) THE COEFFICIENTS IN THE PORTION $i = 6$ TO 72 & $p = 33$ TO 38 ARE TAKEN FROM TABLES ON ...

R.N. SHERRY
JAN 20/55



67 27, 29
72 31, 32

(2) THE COEFFICIENTS IN THE PORTION, l = 6 TO 72 & p = 33 TO 38 ARE TAKEN FROM TABLES ON SHEETS 1-51 & 1-52 OF REPORT 7/0510/2 IN ACCORDANCE WITH THE EQUATIONS OF SHEET 1-39 OF THE SAME REPORT.

(3) THE COEFFICIENTS SHOWN THUS • ARE COMPUTED IN THE SHEETS (4-2 et seq.)

(4) K_{lp} FOR l = 52 & p = 21, 24
62 27, 29
69 31, 32
MUST BE FOUND AS THE COEFFICIENTS DESCRIBED IN NOTE (2).

A.V.ROE CANADA LIMITED MALTON – ONTARIO TECHNICAL DEPARTMENT (Aircraft)		REPORT NO. 7/0510/7 SHEET NO. 4-22	
AIRCRAFT C.105	INNER WING	PREPARED BY: V.F. GARDENER	DATE: FEB 11 '55
		CHECKED BY: R.N. SHEARLY R.N.S.	DATE: APR. 27/55

REDUNDANT STRESS COEFFS.

SINCE THE ELEMENTS OF THE K_{ip} (STRESS – REDUNDANT MOMENT) MATRIX ARE A FUNCTION OF WING GEOMETRY, WHICH IS UNCHANGED FROM PREVIOUS ONE, AND RIB, SPAR AREAS & PANEL THICKNESS, THE NEW VALUES CAN BE OBTAINED SIMPLY BY MULTIPLYING THE OLD VALUES BY THE RATIO OF OLD TO NEW AREA (FOR SPARS AND RIBS) AND THICKNESS (FOR PANELS)

THESE RATIOS FOR EACH STRESS POINT ARE GIVEN IN p. 4-23

Inner Wing — Kip

Multiplying Factors for Converting Report 7/0510/2 Values

A. V. ROE C

MALT

TECHNICA

AIRCRAFT -

WEIGHT -

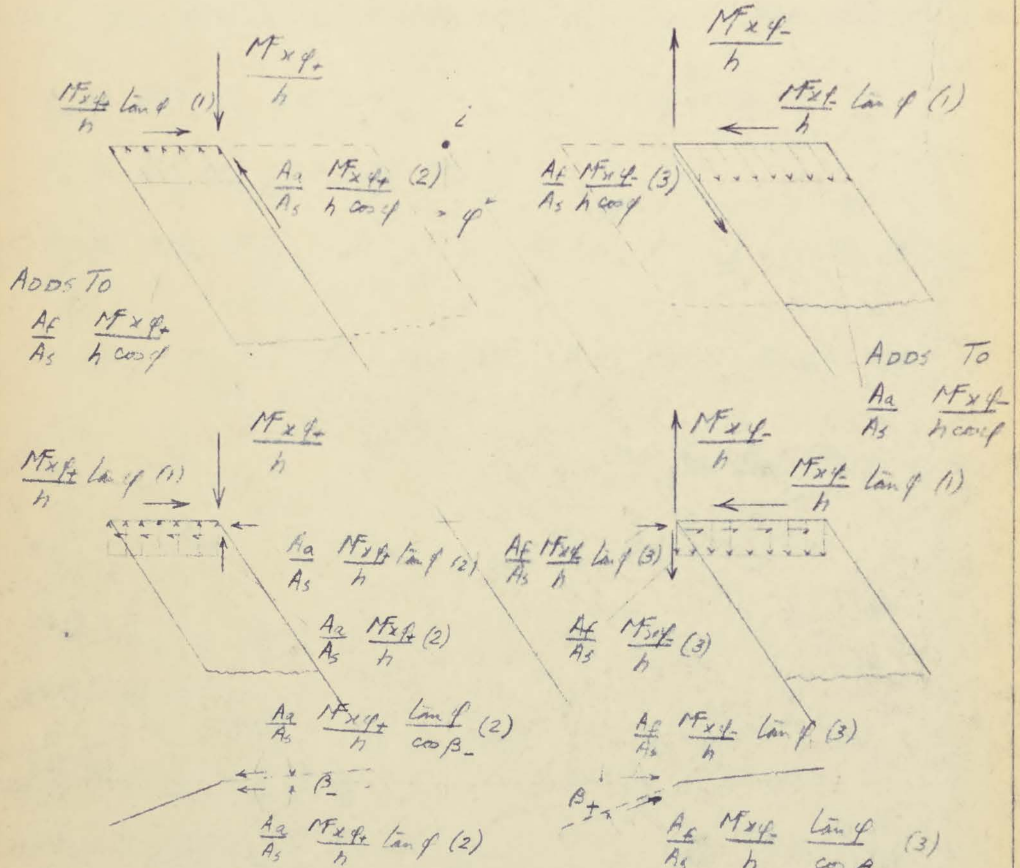
C. G. POSITION -

	1	2	3	4	5	6	7	8	9	10	11	
STRESS POINT	OLD AREA	NEW AREA	OLD A / NEW A		STRESS POINT	OLD AREA	NEW AREA	OLD A / NEW A		STRESS POINT	OLD AREA	N
73	5.037	6.0010	.84054		91	37069	27256	1.3650		106.	·204	·1
74	4.000	10.6038	.37722		92	6.443	47385	1.35971		107	6.177	5.6
75	4.2266	7.7888	.54265		93	37069	27256	1.3600		108	11.376	7.7
76	9.066	5.7035	1.58955		94	·204	·150	1.3600		109	4.5401	337
78	2.8936	6.7288	.43003		95	·204	·150	1.3600		110	11.594	1.3
79	9.17	5.8480	1.56806		96	·204	·150	1.3600		111	4.5401	3.3
81	17536	6.2638	.28092		97	5.779	5.6839	1.01673		112	7.377	5.
82	6.118	4.4141	1.38601		98	10.538	6.9850	1.50853		113	4.5401	33
84	·204	·150	1.3600		99	4.1024	3.0165	1.3600		114	·204	·15
85	·204	150	1.3600		100	10.708	7.1626	1.49499		115	·204	1
86	·204	·150	1.3600		101	4.1024	3.0165	1.3600		116	·204	·1
87	5.422	6.6091	.82038		102	6.838	5.0976	1.34142		117	6.608	5
88	9766	6.3121	1.54719		103	4.1024	3.0165	1.3600		118	12.307	8.5
89	37069	27256	1.3600		104	·204	·150	1.3600		119	6.1176	4
90	9.91	6.4720	1.53121		105	·204	·150	1.3600		120	12.571	8.7

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/7			
SHEET NO 4-26			
AIRCRAFT: C-105	Kip	PREPARED BY: R.N. SHEARLY	DATE: MAR 8/55
		CHECKED BY:	DATE:



NOTES (1) FROM RIB FLANGE (2) FROM SPAR FLANGE AFT OF SPOT i (3) FROM SPAR FLANGE FORWARD OF SPOT i

RIB FLANGE AREA $= A_R$ (ONE SIDE OF ℄)

STRESS IN RIB FLANGE

FORWARD OF i

$$-\left(\frac{A_a}{A_s}\frac{M_{Fx\varphi_+}}{h}\frac{\tan\varphi}{\cos\beta_-}\right)\frac{1}{A_R}$$

AFT OF i

$$-\left(\frac{A_f}{A_s}\frac{M_{Fx\varphi_-}}{h}\frac{\tan\varphi}{\cos\beta_+}\right)\frac{1}{A_R}$$

TOTAL STRESS IN RIB

SEE SHEET 7.3.5

$$-\frac{\tan\varphi}{h A_R A_s}\left(\frac{A_a M_{Fx\varphi_+}}{\cos\beta_-}+\frac{A_f M_{Fx\varphi_-}}{\cos\beta_+}\right)$$

FORM 1194

A. V. RO
TEC
AIRCRAFT
WEIGHT
C. G. POSITI

REDUNDANT STRESS COEFFS — PANEL

$$\tau_m = KP_1 \, MF_{x_2} + KP_2 \, MF_{x_1} + KP_3 \, MF_{\phi_2} + KP_4 \, MF_{\phi_1}$$

$$\frac{\cdot 204}{\cdot 150} = 1{\cdot}36$$

	1	2	3	4	5	6	7	8	9	10	11
	VALUES FROM 7/0510/2 p 342				KP_1	KP_2	KP_3	KP_4			
PANEL	KP_1	KP_2	KP_3	KP_4	1·36 × ①	1·36 × ②	1·36 × ③	1·36 × ④			
124	·0056400	-·0061510	·0031448	-·0024549	·0076813	-·0083654	·0050927	-·0047394			
125	·0071162	-·0073600	·0045006	-·0041350	·00967[illegible]	-·010054	·0059712	-·0056236			
126	[illegible]	-·0103817	·0055595	-·0054187	·0137140	-·0149354	·0079689	-·0076414			
136	·0052372	-·0056657	·0031886	-·0029676	·00712[illegible]	-·0077054	·0043364	-·0040354			
137	·0067789	-·0073333	·0038407	-·0036171	·0092193	-·0099733	·0052234	-·0049193			
138	·0099137	-·0107245	·0052895	-·0050721	·0134826	-·0145853	·0071937	-·0068981			

FORM 1543

REPORT 7/0510/7 SHT 4-31

R.N. SHERLY MAR 8/55

SOURCE SHTS 4-28, 4-29, 4-30

68 69 70 71 72 73 74

114 115 116 117 118 119 120 121 122 123 124 125 126 127 128 129 130 131 132 133 134 135 136 137

.0076414

.0071228

.0092173

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/7	SHEET NO. 5-0	
AIRCRAFT: C105	Tia	
PREPARED BY: D.L. Turner	DATE: 17 June 55	
CHECKED BY	DATE	

Introduction

In this section of the report, calculations are made to determine the elements of the Tia matrix for all wing components except the wing tip, which is covered in section 2.

Except for Tia elements due to the interaction loads between the wing (as defined for this report) and other components of the aircraft, all elements calculated in this section are determined through operations carried out on the I.B.M. automatic computer. The method used is outlined in shts. 5-1 et seq.*

The calculations for the interaction loads between the wing and other components of the aircraft appear on shts. 5-98 et seq.

Due to last minute changes in some panel thicknesses, rib & spar areas, the matrix made up from the results of the calculations of this section must be factored by an appropriate set of correction factors before being used. These factors are calculated & tabulated in section 6.

* See report 7/0510/7 Vol II for instructions to the computation centre and results of this computation.

FORM 719A

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C105	Tia	REPORT NO. 7/0510/7	
		SHEET NO. 5-1	
		PREPARED BY: D L Turner	DATE: 8 June '55
		CHECKED BY: DLT	DATE

Statically Determinate Stresses

Due to the large amount of routine calculation involved in developing the Tia matrix for the structure, (as exemplified in report 7/0510/2), it has been decided to set up a system, making use of the I.B.M. automatic computer, which will eliminate as much of this routine work as possible.

The method is outlined below:

(1) Construct matrices T_1, T_2, T_3 & T_4,

where T_1 is the matrix made up of the coefficients of the equations relating stresses in the wing tip to the unit loads applied to the wing tip.

T_2 relates stresses in the outer wing to the unit loads applied to the outer wing.

T_3 relates stresses in the inner wing to the unit loads applied to the inner wing.

T_4 relates stresses in the centre section wing to the unit loads applied to the centre section wing.

These matrices are all portions of the required Tia matrix (see sketch on sht. 5-5)

(2) Construct matrices G_1, G_2 & G_3,

where G_1 is the matrix made up of the coefficients of the equations relating unit loads on the wing tip to interaction loads (designated P_1) between the wing tip and the outer wing.

G_2 relates unit loads applied to the outer wing to interaction loads (designated P_2) between the outer & inner wing.

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C105	T_{ia}	REPORT NO. 7/0510/7	
		SHEET NO. 5-2	
		PREPARED BY	DATE
		D.L. Turner	27 April 55
		CHECKED BY	DATE
		R.N. Shearly	May /55

Statically Determinate Stresses.

G_3 relates unit loads applied to the inner wing to interaction loads between the inner wing and centre section

(3) Construct matrices H_1 & H_2.

where H_1 is the matrix made up of the coefficients of the equations relating the interaction loads P_1 to interaction loads P_2.

H_2 relates interaction loads P_2 to interaction loads P_3.

(4) From this data, values of P_1, P_2 & P_3 may be determined in terms of the actual applied loads (Q_a) as follows:

$$P_1 = G_1 Q_1$$

$$P_2 = G_2 Q_2 + H_1 P_1$$

$$P_3 = G_3 Q_3 + H_2 P_2$$

where Q_1, Q_2, Q_3 and Q_4 are those parts of Q_a applied to the wing tip, outer wing, inner wing and centre section respectively (ref. sht. 5-5)

(5) Construct matrices T_1', T_2' & T_3'.

where T_1' is the matrix made up of the coefficients of the equations relating stresses in the outer wing to interaction loads P_1

T_2' relates stresses in the inner wing to interaction loads P_2.

T_3' relates stresses in the centre section wing to interaction loads P_3.

FORM 1318A

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0510/7	SHEET No. 5-3	
AIRCRAFT: C105	T_{ia}	
PREPARED BY: D L Turner	DATE: 8 June '55	
CHECKED BY: DLJ	DATE:	

Statically Determinate Stresses.

(6) From this data, values of $\mathring{S}_1, \mathring{S}_2, \mathring{S}_3$ & $\mathring{S}_4$ may be determined in terms of the actual applied loads (Q_a) as follows:

$$\mathring{S}_1 = T_1 Q_1$$

$$\mathring{S}_2 = T_2 Q_2 + T_1' P_1$$ where P_1, P_2 & P_3 are determined from (4)

$$\mathring{S}_3 = T_3 Q_3 + T_2' P_2$$

$$\mathring{S}_4 = T_4 Q_4 + T_3' P_3$$

$\mathring{S}_1, \mathring{S}_2, \mathring{S}_3$ & $\mathring{S}_4$ are the stresses in the wing tip, outer wing, inner wing and centre section, respectively, (see sketch on sht. 5-4) due to all loads Q_a.

It follows from the above that the overall T_{ia} matrix (where $\mathring{S}_i = T_i = T_{ia} Q_a$ defines the T_{ia} matrix) may be constructed according to the chart on sht. 5-5.

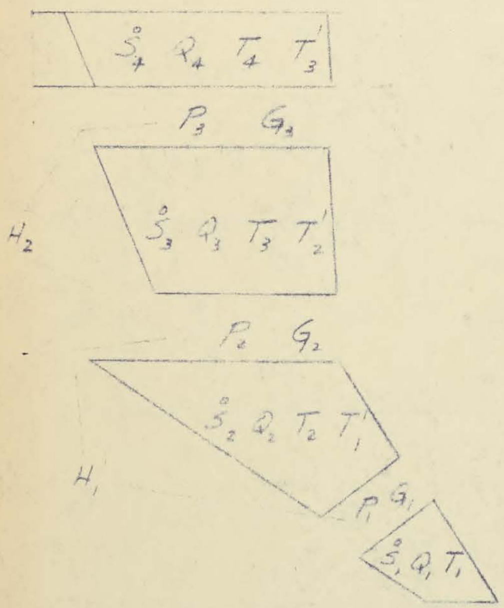
In this section of the report, calculations are made to determine the elements of matrices $G_1, T_2, T_1', H_1, G_2, T_2', T_3, G_3, H_2, T_3'$ & T_4. Matrix T_1 is covered in section 2 (wing tip)

For the sake of completeness, the instructions issued to the computer section on "Computation of Preliminary Matrices," which covers the computations outlined above, (report 7/0510/7 Vol II) should be referred to.

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

AIRCRAFT: C-105	T_{1a}	REPORT NO. 7/0510/7	SHEET NO. 5-4
		PREPARED BY: R.N. SHEARLY	DATE: FEB 6/55
		CHECKED BY: R.N. SHEARLY DD	DATE: MAY 5/55

THE FOLLOWING IS A COPY OF SOME PRELIMINARY WORK BY MR. A. GRZEDZIELSKI



Q — LOADS

P_1 P_2 P_3 — INTERACTION LOADS

$\overset{\circ}{S}$ — STRESSES.

$$\overset{\circ}{S}_1 = T_1 Q_1$$

$$\overset{\circ}{S}_2 = T_2 Q_2 + T_1' P_1$$

$$\overset{\circ}{S}_3 = T_3 Q_3 + T_2' P_2$$

$$\overset{\circ}{S}_4 = T_4 Q_4 + T_3' P_3$$

$$P_1 = G_1 Q_1$$

$$P_2 = G_2 Q_2 + H_1 P_1$$

$$P_3 = G_3 Q_3 + H_2 P_2$$

FORM 1219A

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C105	T_{1a}	REPORT NO. 7/0510/7	SHEET NO 5-5
		PREPARED BY: D L Turner	DATE: 27 April '55
		CHECKED BY: R.N. SHEARLY	DATE: MAY '55

Statically Determinate Stresses

Chart for constructing the T_{1a} matrix, given T_1', T_2', T_3', G_1, G_2, G_3, H_1, H_2, T_1, T_2, T_3 & T_4.

The area outlined in heavy lines is that area built up from the computation of preliminary matrices.

The given matrices and the final T_{1a} matrix appear in the set of matrices accompanying this report.

$i \rightarrow$ / $a \rightarrow$	Q_1	Q_2	Q_3	Q_4
S_1	T_1	0	0	0
S_2	$T_1' \times G_1$	T_2	0	0
S_3	$T_2' \times H_1 \times G_1$	$T_2' \times G_2$	T_3	0
S_4	$T_3' \times H_2 \times H_1 \times G_1$	$T_3' \times H_2 \times G_2$	$T_3' \times G_3$	T_4

AVRO AIRCRAFT LIMITED TECHNICAL DEPARTMENT (Aircraft)		REPORT NO. 7/0510/7 SHEET NO. 5-6 & 7 *	
		PREPARED BY	DATE
AIRCRAFT: C105	T_{ia}	D L Turner	8 June 55
		CHECKED BY	DATE
		[illegible]	

Statically Determinate Stresses

The stress points "i" and the load points "a" associated with each portion of the structure are tabulated below.

	Stress Points "i"	Load Points "a"
Tip	6 - 13 incl.	1-12 incl
Outer Wing	14-72 incl & $\overset{*}{74}$, $\overset{*}{75}$, $\overset{*}{78}$, $\overset{*}{81}$ & 83.	13, 14, 20-33 incl., 35, 36 37, 39, 40 & 42
Inner Wing	73, $\overset{*}{74}$, $\overset{*}{75}$, 76, $\overset{*}{78}$, $\overset{*}{79}$, $\overset{*}{81}$, 82, 84-118 incl, $\overset{*}{119}$, 120 $\overset{*}{121}$, 122 & $\overset{*}{123}$.	15-19 incl., 34, 38, 41, 43-55 incl.
Centre Section	$\overset{*}{119}$, $\overset{*}{121}$ & $\overset{*}{123}$, 124-149 incl.	56-67 incl. except 58 & 63 also 81 & 82.

In the above table, the stress points marked $\overset{*}{i}$ are stress points which, in the preliminary calculations, are associated with two portions of the structure; e.g. outer wing & inner wing. In each case the two resulting parts of the T_{ia} element must be combined to obtain the final T_{ia} element.

Note that the interaction loads, a = 101 - 114 incl., do not appear in the preliminary calculations. Stresses due to these loads are calculated separately.

Note also that; stress points 77, 80, 127 & 147 appear only in the interaction load calculations; stress point 83 serves only as a panel identification; and load points 58, 63 and 68 - 80 do not exist.

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT C105	Wing Tip	REPORT NO 7/0510/7	
		SHEET NO 5-8 *	
		PREPARED BY	DATE
		[signature]	17 May 55
		CHECKED BY	DATE
		[signature]	May 24/55

Statically Determinate Stress Coefficients

Calculations for determining the elements of the wing tip - outer wing interaction (G_1) matrix for use in calculating the stresses in the remainder of the structure due to wing tip loads (see shts. 5-1 et seq.)

On sht. 2-15, equations are given relating the shear, moment and torque at the shear centre on rib 18 (the wing tip - outer wing boundary) to the wing tip applied loads.

The interaction loads between the wing tip and the outer wing consist of bending moments (M_{θ_1}, M_{θ_2}, M_{θ_3}, M_{θ_4} & M_{θ_5}, +ve for tension in top surface) acting on each spar (vectors perp. to spar); a torque (nose up +ve), T_{TIP}, acting around rib 18 between the front and rear spars; and vertical shears (+ve down) (P_{θ_1}, P_{θ_2}, P_{θ_3}, P_{θ_4} & P_{θ_5}) acting at each spar at rib 18.

Values of these interaction loads may be found by distributing the moment and shear given on sht. 2-15 according to the equations of . Interaction torque is already given by the torque equation of sht. 2-15.

On sht. 5-9, values of the moment & shear distribution factors are determined (according to the equations of and on sht. 5-10 the matrix elements are calculated and tabulated.

The matrix G_1 may now be defined, using the notation of sht. 5-1 et seq., by means of the equation

$$P_1 = G_1 Q_1$$

where P_1 denotes the matrix vector of the interaction loads, G_1 the interaction matrix, and Q_1 the matrix vector of applied loads (wing tip.)

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO 7/0510/7	SHEET NO 5-9	*
AIRCRAFT C105	Wing Tip	
PREPARED BY	DATE	
D L Turner	17 May 55	
CHECKED BY	DATE	
R. M. Ahearn	MAY 24/55	

Statically Determinate Stresses.

G_1 Matrix

From the proportions of the total moment and shear at rib 18 carried by each spar are:

$$\frac{M_i}{M} = \frac{I_i \cos^3\theta_i}{\Sigma I_i \cos^3\theta_i} \qquad \frac{Q_i}{Q} = \frac{I_i \cos^3\theta_i}{\Sigma I_i \cos^3\theta_i}$$

(1) i	(2) $I_i \cos^3\theta_i$	(3) $\frac{M_i}{M}$ .00396926(2)	(4) $I_i \cos^3\theta_i$	(5) $\frac{Q_i}{Q}$ .00396926(4)	
θ_1	37.90389	.150474	37.88487	.150375	
θ_2	78.77599	.312682	78.49712	.311575	ref. sht. 1-3
θ_3	69.38028	.275388	68.23828	.270855	
θ_4	55.84000	.221643	53.94960	.214140	
θ_5	14.24606	.056546	13.36637	.053055	

$\Sigma I_i \cos^3\theta_i = 251.93624$ sht. 1-3

$\frac{1}{\Sigma I_i \cos^3\theta_i} = .00396926$

Torque is not distributed in this calculation

Elements of the G_1 matrix are calculated from this data, and tabulated on sht. 5-10.

FORM 1104

…ents of Wing Tip – Outer Wing Interaction (G_1) Matrix.

6	7	8	9	10	11	12	13	14	15	16	17	18	19	2…
Q_5	Q_6	Q_7	Q_8	Q_9	Q_{10}	Q_{11}	Q_{12}			Q_1	Q_2	Q_3	Q_4	Q…
62.9670	62.9670	30.9670	30.9670	34.7227	94.9670	62.9670	30.9670			17.48[illegible]	20.27226	14.29006	14.29006	947…
										36.32702	42.12536	29.63497	29.63497	19.…
										31.39425	37.10101	26.15277	26.15277	17.…
										25.75022	29.86[illegible]	21.04877	21.04877	13.…
62.9670	62.9670	30.9670	30.9670	34.7227	94.9670	62.9670	30.9670			6.56945	7.61803	5.37000	5.37000	3.5…
$\overline{24}.4552$	20.2826	$\overline{31}.5442$	26.1625	22.0966	29.4026	35.2826	41.1625			$\overline{12}.6661$	7.0975	$\overline{17}.3656$	14.4026	$\overline{24}$…
1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000			.150375	.150375	.150375	.150375	.15…
										.311575	.311575	.311575	.311575	.31…
										.270855	.270855	.270855	.270855	.27…
										.214140	.214140	.214140	.214140	.21…
1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000	1.0000			.053055	.053055	.053055	.053055	.05…

A. V. ROE CANADA LIMITED
MALTON, ONTARIO
ENGINEERING DIVISION -- AIRCRAFT

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

REPORT NO. - 7/0510/2
SHEET - - 5-10 *
DATE - - 17 May 55
PREPARED BY - D.L. [illegible]

Elements of G. Matrix

	17	18	19	20	21	22	23	24	25	26	27	28	29
	Q_2	Q_3	Q_4	Q_5	Q_6	Q_7	Q_8	Q_9	Q_{10}	Q_{11}	Q_{12}		
..2	20.27226	14.29006	14.29006	9.47490	9.47490	4.65973	4.65973	20.27226	14.29006	9.47490	4.65973		
02	42.12536	29.69447	29.69447	19.68865	19.68865	9.68282	9.68282	42.12536	29.69447	19.68865	9.68282		
25	37.10101	26.15277	26.15277	17.34036	17.34036	8.52794	8.52794	37.10101	26.15277	17.34036	8.52794		
22	29.86034	21.04877	21.04877	13.95619	13.95619	6.86362	6.86362	29.86034	21.04877	13.95619	6.86362		
5	7.61803	5.37000	5.37000	3.56053	3.56053	1.75106	1.75106	7.61803	5.37000	3.56053	1.75106		
61	7.0975	$\overline{1}7.3656$	14.4026	$\overline{2}4.4552$	20.2826	$\overline{3}1.5449$	26.1625	22.0366	29.4026	35.2826	41.1625		
5	.150375	.150375	.150375	.150375	.150375	.150375	.150375	.150375	.150375	.150375	.150375		
5	.311575	.311575	.311575	.311575	.311575	.311575	.311575	.311575	.311575	.311575	.311575		
55	.270855	.270855	.270855	.270855	.270855	.270855	.270855	.270855	.270855	.270855	.270855		
0	.214140	.214140	.214140	.214140	.214140	.214140	.214140	.214140	.214140	.214140	.214140		
5	.053055	.053055	.053055	.053055	.053055	.053055	.053055	.053055	.053055	.053055	.053055		

A.V.ROE CANADA LIMITED MALTON – ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7/0510/7 SHEET NO. 5-11	
		PREPARED BY:	DATE:
AIRCRAFT C105	Outer Wing	P.J. Turner	4 Feb '55
		CHECKED BY: R.N. SHEARLY	DATE: MAY /55

Statically Determinate Stresses.

Determination of the elements of the T_2 (outer wing stress – outer wing load) matrix. (ref. shts 5-1 et seq.)

Since the geometry and section properties of the outer wing have not changed from those used in the analysis of report 7/0510/2, the elements of the T_2 matrix for these loads may be taken directly from the T_{10} matrix of report 7/0510/2. These loads are at $j = 13, 14, 20-33, 35-37, 39, 40$ & 42 and the unchanged stresses due to these loads occur at $i = 14-67, 69, 72$.

With the addition of redundancy 7, and the inclusion of the cell forward and inboard of this redundancy in this analysis, loads applied to the front spar ($j = 20, 25, 30, 35$ & 39) will be carried through to load point 42 rather than 39. These loads will then be transferred to load point 43, through the structure added forward & inboard of redundancy 7, in the manner detailed in

Thus in this analysis, no stress will be generated in panels 68 & 73, and stresses will be generated at stress point 70 of;

$$\sigma_{70} = \frac{MQ_{42}}{h_{70} A_{70}}$$

where MQ_{42} represents the moment at load point 42 due to loads 20, 25, 30, 35 & 39.

Hence

$$\sigma_{70} = .039704(103.3146\, Q_{20} + 91.7639\, Q_{25} + 71.4332\, Q_{30} + 47.6219\, Q_{35}$$

$$+ 23.8108\, Q_{39})$$

$$\sigma_{70} = 4.340227\, Q_{20} + 3.693394\, Q_{25} + 2.836164\, Q_{30} + 1.890780\, Q_{35} + .945384\, Q_{39}$$

The value of $\frac{1}{h_{70} A_{70}}$ derives from rep. 7/0510/2 sht 4-41, moment arms from sht 4-45.

A.V.ROE CANADA LIMITED MALTON — ONTARIO TECHNICAL DEPARTMENT (Aircraft)		REPORT NO. 7/0510/7 SHEET NO. 5-12	
		PREPARED BY:	DATE:
AIRCRAFT C105	Outer Wing	[signature]	4 Feb. 55
		CHECKED BY: E.N. SHEARLY	DATE: MAY /55

Statically Determinate Stresses

Expressions for the stresses developed at stress points 71, 74, 75, 78 & 81 due to loads 20, 25, 30, 35, 39 & 42 (i.e. stresses developed as a result of the transfer of these loads from load point 42 to load point 43) are developed in

According to this work:

$$M_{74} = -MQ_{42}\frac{c}{a} + \left[Q_{42}\, b\cos\alpha + MQ_{42}\frac{b}{a}\cos\alpha\right]$$

$$M_{71} = MQ_{42}\frac{b}{a} + Q_{42}\, b.$$

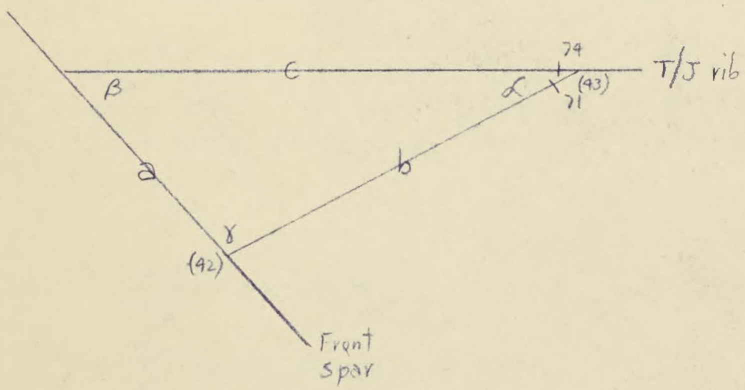
where M_{74} represents the moment in the transport joint rib at stress point 74, M_{71} represents the moment in the outer wing rib at stress point 71. Here again MQ_{42} & Q_{42} represent the moment and load at load point 42 due to loads at $a = 20, 25, 30, 35, 39$ & 42. The meaning of the constants in these expressions, together with their numerical values are given on sht. 5-13

The term bracketted in the expression for M_{74} represents the contribution to M_{74} from the outer wing rib. The remainder represents the moment transferred through the transport joint rib.

Now, from the work of sht. 7-2 et seq. all of M_{74} will not cause a rib stress at stress point 74. In this work, it is shown that all of the portion of the moment transferred through the transport joint rib causes stress at stress point 74, but that only .349276 of the moment transferred through the outer wing rib will cause stress.

Statically Determinate Stresses

Structure added forward & inboard of redundancy 7.



$a = 35.4527$ $\alpha = 39.5°$ $\cos\alpha = .77162$
$b = 28.0825$
$c = 52.2944$

$\frac{c}{a} = 1.475047$ $\frac{b}{a} = .792112$

$b\cos\alpha = 21.669019$ $\frac{b}{a}\cos\alpha = .611209$

Ref. Figs. 5 & 6
Shts. 1-62 & 1-63

A.V.ROE CANADA LIMITED MALTON – ONTARIO TECHNICAL DEPARTMENT (Aircraft)		REPORT NO. 7/0510/7 SHEET NO. 5-14	
		PREPARED BY: D.L. Turner	DATE: 4 Feb. 55
AIRCRAFT C105	Outer Wing	CHECKED BY: E.N. SHEARLY	DATE: MAY /55

Statically Determinate Stresses

Hence :

$$\sigma_{74} = \frac{1}{h_{74}A_{74}}\left\{-MQ_{a2}\frac{c}{a} + .349276\left[Q_{a2}\, b\cos\alpha + MQ_{a2}\frac{b}{a}\cos\alpha\right]\right\}$$

From this same work, the proportions of each of these two parts of M_{74} causing stress at stress points 75, 78 & 81 are obtained. (Note that these factors are the same for each of these two parts)

Hence :

$$\sigma_{75} = \frac{1}{h_{75}A_{75}} \times .633101\ M_{74} \qquad \sigma_{81} = 0$$

$$\sigma_{78} = \frac{1}{h_{78}A_{78}} \times .287453\ M_{74}$$

A table of these proportions is reproduced on sht. 5-17 for reference.

Values of the coefficients of the applied loads for these stresses are determined in the table on sht. 5-15.

Stress at stress point 71 derives directly from the moment equation, hence :

$$\sigma_{71} = \frac{M_{71}}{h_{71}A_{71}} = \frac{1}{h_{71}A_{71}}\left(MQ_{a2}\frac{b}{a} + Q_{a2}\, b\right)$$

where the value of $\frac{1}{h_{71}A_{71}}$ is obtained from report 7/0510/2 sht 4.41.

Coefficients of the applied load for this stress are determined on sht. 5-16.

Statically Determinate Stress Coefficients at Stress Points 71, 74, 75, 78 & 81 due to loads at load points 20, 25, 30, 35, 39 & 42.

A. V. ROE CA
MALTO
TECHNICAL

AIRCRAFT -
WEIGHT -
C. G. POSITION -

	1	2	3	4	5	6	7	8	9	10	11	1
	Q_{20}	Q_{25}	Q_{30}	Q_{35}	Q_{39}	Q_{42}						
(1) Q_{42}	1	1	1	1	1	1						
(2) MQ_{42}	109.3146	91.7639	71.4327	47.6219	23.8108	0			Ref Report 7/05?0/2 sht			
$-M' = -MQ_{42}\frac{c}{d}$ (3) $-1.4?047\times(2)$	$\overline{161.2442}$	$\overline{135.3561}$	$\overline{105.3666}$	$\overline{70.2445}$	$\overline{35.1220}$	0			These values of M' are			
$Q_{42}\,b\cos\alpha =$ (4) $21.669019\times(1)$	21.669019	21.669019	21.669019	21.669019	21.669019	21.669019						
$MQ_{42}\frac{h}{d}\cos\alpha =$ (5) $.611209\times(2)$	66.8141	56.0869	43.6603	29.1069	14.5534	0						
(6) = (4) + (5)	88.4831	77.7559	65.3293	50.7759	36.2224	21.6690						
$M_{74} =$ (7) (3) + (6)	$\overline{72.7611}$	$\overline{57.6002}$	$\overline{40.0373}$	$\overline{19.4686}$	1.1004	21.6690						
(8) $1.000\times(3)$	$\overline{161.2442}$	$\overline{135.3561}$	$\overline{105.3666}$	$\overline{70.2445}$	$\overline{35.1220}$	0						
(9) $.349276\times(6)$	30.9050	27.1583	22.8180	17.7348	12.6516	7.5685						
(10) (8) + (9)	$\overline{130.3392}$	$\overline{108.1978}$	$\overline{82.5486}$	$\overline{52.5097}$	$\overline{22.4704}$	7.5685						
σ_{74} (11) $.009999\times(10)$	$\overline{1.303262}$	$\overline{1.081870}$	$\overline{.825403}$	$\overline{.525044}$	$\overline{.224682}$	.075677			At Stress Point 74	h is 9.4312	A is 10.6038	hA 100.0
σ_{75} (12) $.008660\times(7)$	$\overline{.630111}$	$\overline{.498818}$	$\overline{.346723}$	$\overline{.168598}$	.009529	.187654			75	9.3858	7.7888	73.10
σ_{78} (13) $.005122\times(7)$	$\overline{.372682}$	$\overline{.295028}$	$\overline{.205071}$	$\overline{.099718}$	.005636	.110989			78	8.3414	6.7288	56.12
σ_{81} (14) $0\times(7)$	0	0	0	0	0	0			81	6.4011	6.2638	40.09

A.V. ROE CANADA LIMITED MALTON – ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7/0510/7 SHEET NO. 5 - 16	
		PREPARED BY: D L Turner	DATE: 4 Feb. '55
AIRCRAFT C105	Outer Wing	CHECKED BY: R. N. SHEARLY	DATE: MAY /55

Statically Determinate Stresses

	Q_{20}	Q_{25}	Q_{30}	Q_{35}	Q_{39}	Q_{42}
(1) MQ_{42}	109.3146	91.7639	71.4327	47.6219	23.8108	0
(2) Q_{42}	1	1	1	1	1	1
(3) $MQ_{42}\frac{h}{I}$ .792112 (1)	86.5894	72.6873	56.5827	37.7219	18.8608	0
(4) $Q_{42}b$ 28.0825 (2)	28.0825	28.0825	28.0825	28.0825	28.0825	28.0825
(5) M_{31} * (3)+(4)	114.6719	100.7698	84.6652	65.8044	46.9433	28.0825
(6) σ_{31} .023518 (5)	2.61[illegible]	2.299365	1.991831	1.501525	1.071152	.640786

Coefficients of Applied Loads Q_{20}, Q_{25}, Q_{30}, Q_{35}, Q_{39} & Q_{42} for stress σ_{31}.

---

* These values are used in calculating elements of the G_2 matrix

A. V. ROE CANADA LIMITED MALTON - ONTARIO TECHNICAL DEPARTMENT (Aircraft)		REPORT NO. 7/0510/17 SHEET NO. 5-18	
		PREPARED BY	DATE
AIRCRAFT: C105	Outer Wing	D.T. Jenner	17 Feb 55
		CHECKED BY	DATE
		R.N. SHEARLY	11/1/55

Statically Determinate Stresses

In resolving the moments in the M/S, F/C/S, R/C/S & R/S, due to outer wing loads applied to these spars, into components perpendicular and parallel to the transport joint rib, stresses are generated in the transport joint rib at stress points 74, 75, 78 & 81.

These stresses are due to the redistribution of the torque components $M \cos \gamma$ resulting from this resolution (γ is the angle between the transport joint rib and the spar carrying M)

The determination of these moments, and their resolution into torque and moment components is done in determining the elements of the G_3 matrix. (ref. sht. 5-48)

The torques are tabulated below (vectors perp to T/J rib)

$$T_{M/S} = 86.36771 Q_{21} + 72.50451 Q_{26} + 56.44095 Q_{31} + 37.62702 Q_{36} + 18.81343 Q_{41}$$

$$T_{F/S} = 62.40105 Q_{22} + 49.59246 Q_{27} + 34.75470 Q_{32} + 17.37743 Q_{37}$$

$$T_{R/C/S} = 40.58497 Q_{23} + 29.03910 Q_{28} + 15.66411 Q_{33}$$

$$T_{R/S} = 21.60718 Q_{30} + 11.52658 Q_{29}$$

From the work of shts. 7-10 & 7-11 the proportions of these torques causing stresses at each of stress points 74, 75, 78 & 81 are as in the upper table of sht. 5-17.

Stresses at these points are calculated using the equation

$$\sigma = \frac{kT}{hA}$$ where k is the proportion factor obtained as above.

Values of $\frac{1}{hA}$ are given on sht. 5-15

The computations as outlined above are carried out in the table on sht. 5-19

FORM 1319A

Elements of T_1 Matrix at $i = 24, 25, 28$ & 31 due to

$a = 21, 22, 23, 24, 26, 27, 28, 29, 31, 32, 33, 36$, [illegible] & 40

A. V. ROE C[illegible]
MALTO[illegible]
TECHNICAL [illegible]

AIRCRAFT -
WEIGHT -
C. G. POSITION

	1	2	3	4	5	6	7	8	9	10	11	1[illegible]
		$T_{R/S}$			$T_{FR/S}$			$T_{[illegible]/S}$			$T_{R/S}$	
	$\frac{1}{hA}$	factor	$\frac{k}{hA}$ (1)×(2)		factor	$\frac{k}{hA}$ (1)×[illegible]		factor	$\frac{k}{hA}$ (1)×[illegible]		factor	[illegible] (1)×
σ_{24}	.00[illegible]	-.349276	-.00[illegible]		0	0		0	0		0	0
σ_{25}	.013679	-.633101	-.008660		-.133101	-.001821		.366853	.005019		.3[illegible]	.005[illegible]
σ_{28}	.017817	-.287453	-.005122		-.287453	-.005124		.212[illegible]	.003[illegible]		.712547	.012[illegible]
σ_{31}	.024341	0	0		0	0		0	0		.211[illegible]	.005[illegible]

... # 81 due to

... 75.3 ...

A. V. ROE CANADA LIMITED MALTON, ONTARIO TECHNICAL DEPT. (AIRFRAME)	REPORT NO. 7/[illegible] SHEET [illegible]
AIRCRAFT C105 WEIGHT C. G. POSITION	DATE 16 [illegible] PREPARED BY [illegible]

6	7	8	9	10	11	12	13	14	15	16	17	18	19
$\frac{k}{[illegible]}$		$T_{R/s}$ factor	$\frac{k}{TA}$		$T_{R/s}$ factor	$\frac{k}{TA}$		a	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
									$T_{M/s}$	[illegible]	[illegible].8660	[illegible]122	
0		0	0		0	0		21	86.36721	$\overline{.301520}$	$\overline{.742[illegible]}$	$\overline{.442305}$	0
.001821		.366853	.005019		.366853	.005019		22	72.[illegible]	$\overline{.253[illegible]}$	$\overline{.628830}$	$\overline{.371362}$	
.00512		.212[illegible]	.00372[illegible]		.712547	.012695		21	56.4409	$\overline{.1[illegible]}$	$\overline{.482274}$	$\overline{.255088}$	0
0		0	0		.211137	.015266		[illegible]	37.62702	$\overline{.13394}$	$\overline{.325850}$	$\overline{.192726}$	
								40	18.[illegible]	$\overline{.063[illegible]}$	$\overline{.162924}$	$\overline{.094[illegible]2}$	0
									$T_{F/c/s}$		[illegible]1621	[illegible]152	
								22	62.4[illegible]15	0	$\overline{.113[illegible]32}$	$\overline{.313[illegible]18}$	0
								27	49.[illegible]	0	$\overline{.090308}$	$\overline{.254013}$	0
								31	34.[illegible]	0	$\overline{.063288}$	$\overline{.178014}$	0
								3[illegible]	17.[illegible]	0	$\overline{.031699}$	$\overline{.083007}$	
									$T_{R/c/s}$		[illegible]	[illegible]	
								2[illegible]	40.[illegible]	0	.203[illegible]	.153[illegible]	0
								28	2[illegible]	0	.145747	.109[illegible]71	0
								3[illegible]	15.[illegible]	0	.078615	.05[illegible]	
									$T_{R/s}$		[illegible](15)	[illegible]2.625(15)	[illegible]5[illegible](15)
								24	21.[illegible]	0	.10[illegible]498	.27[illegible]303	.113783
								2[illegible]	11.52[illegible]	0	.05[illegible]3	.147213	.[illegible]

A. V. ROE CANADA LIMITED MALTON - ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7/0510/7 SHEET NO. 5-20	
AIRCRAFT: C105	Outer Wing	PREPARED BY: [illegible]	DATE: 17 Feb '55
		CHECKED BY: E. N. SHEARLY	DATE: 1 May '55

Statically Determinate Stresses

In resolving the torque in the rear cell and the moment in the rear spar of the outer wing (due to loads 13 & 19) into components perpendicular and parallel to the transport joint rib, stresses are generated at stress points 74, 75, 78 & 81 in the transport joint rib. These stresses are due to the redistribution (within the transport joint rib) of the torque component resulting from this resolution.

The components in question are determined in the calculations for the G_2 matrix (shts. 5-48 & 50) and are quoted below.

$$T_1 = 21.60718\, Q_{13} + 11.59658\, Q_{14} \quad \text{(torque comp. at rear spar)}$$

$$T' \cos 32.5° = 11.57495\, (Q_{13} + Q_{14}) \quad \text{(torque comp. at rear cell)}$$

T_1 is identical to $T_{R/S}$ of the work of sht. 5-18 except that Q_{13} & Q_{14} replace Q_{24} & Q_{29} respectively. Hence:

$$\sigma_{74} = 0$$

$$\sigma_{75} = .108446\, Q_{13} + .058203\, Q_{14}$$

$$\sigma_{78} = .274303\, Q_{13} + .147219\, Q_{14}$$

$$\sigma_{81} = .113783\, Q_{13} + .061068\, Q_{14}$$

To be combined with values on sht. 5-22.

FORM 1318A

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

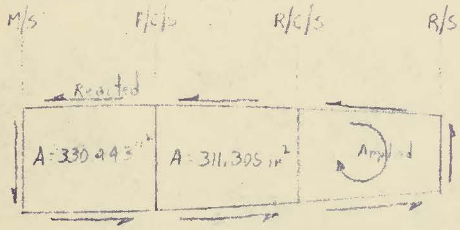
REPORT NO. 7/0510/2
SHEET NO. 5-21

AIRCRAFT: C105	Outer Wing	PREPARED BY	DATE
		D. Turner	17 Feb 55
		CHECKED BY	DATE
		E.N. Shearly	11 A/ /55

Statically Determinate Stresses

To determine transport joint rib stresses due to $T' \cos 39.50°$

This torque applied to the transport joint rib as a torque in the rear cell is redistributed over the complete wing cell between the main spar and the rear spar as indicated below.



$A_c = 900.639 \text{ in}^2$
For areas see sht 7-9

Torque is $T' \cos 39.50° = 11.57445 (Q_{13} + Q_{14})$

From this it can be seen that bending moments in the rib are:

at M/S $\quad B.M. = 0$

at F/C/S $\quad B.M. = \frac{330.443}{900.639} \times 11.57445 (Q_{13} + Q_{14})$

$= 4.246647 (Q_{13} + Q_{14})$

at R/C/S $\quad B.M. = \frac{330.443 + 311.305}{900.639} \times 11.57445 (Q_{13} + Q_{14})$

$= 8.247345 (Q_{13} + Q_{14})$

at R/S $\quad B.M. = 0$

FORM 1319A

Report No. 7/0510/7
Sht No 5-23
8 Feb. '55

D L Turner

Dummy T_2 Matrix

$a \rightarrow$

	13	14	20	21	22	23	24	25	26	27	28	29	30	31	32	33	35	36	37	39	40	42
14																						
15																						
16																						
17																						
18																						
19																						
20																						
21	✓																					
22																						
23																						
24																						
25																						
26	✓																					
27			✓																			
29				✓																		
31					✓																	
33						✓																
34		✓																				
35	✓						✓															
36																						
37																						
38																						
39	✓	✓																				
40			✓					✓														
42				✓					✓													
44					✓					✓												
46						✓					✓											
48	✓	✓					✓					✓										
49																						
50																						
51																						
52	✓	✓																				
53			✓					✓					✓									
55				✓					✓					✓								
57					✓					✓					✓							
59						✓					✓					✓						
60																						
61																						
62																						
63			✓					✓					✓				✓					
65				✓					✓					✓				✓				
67					✓					✓					✓				✓			
68																						
69																						
70			✓					✓					✓				✓			✓		
71			✓					✓					✓				✓			✓		✓
72				✓					✓					✓				✓			✓	
74			✓	✓				✓	✓				✓	✓			✓	✓		✓	✓	✓
75	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓
78	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓
81	✓	✓					✓					✓										
	13	14	20	21	22	23	24	25	26	27	28	29	30	31	32	33	35	36	37	39	40	42

Statically Determinate Stresses.

Determination of the elements of the T_1 (Outer Wing Stress – Wing Tip Load) matrix (ref. sht. S-1 et seq.)

The load set P_1 gives the interaction loads between the wing tip and the outer wing. These loads take the form of a set of spar moments and vertical shears acting at the spar-rib 18 intersections, and a torque acting on the complete wing cell at rib 18. (See sht S-8)

To trace these loads through the outer wing structure the first step is to resolve the torque at rib 18 into three separate torques acting in the three rear cells of the outer wing. This resolution will result in the generation of stresses at the stress points of rib 18.

Resolving the torque T_{Tip} of the set P_1 into three torques in the three rear cells of the outer wing at rib 18 gives:

$$T_R = \frac{121.5312}{320.8958} T_{Tip} = .378724\ T_{Tip} \qquad T_{F-} = \frac{89.6177}{320.8958} T_{Tip} = .279274\ T_{Tip}$$

$$T_{R+} = \frac{109.7469}{320.8958} T_{Tip} = .342002\ T_{Tip}$$

where torque is distributed according to cell area and:

$$A_R\ 121.5312\ in^2 \quad A_{R+} = 109.7469\ in^2 \quad A_{F-} = 89.6177\ in^2$$

$$A_R + A_{R+} + A_{F-} = 320.8958 \qquad \text{(ref. sht. 1-18)}$$

where T_R = Torque in rear cell
T_{R+} = Torque in cell f'w'd of rear cell
T_{F-} = Torque in cell a'ft of front cell

} Corresponding Cell Areas are { A_R, A_{R+}, A_{F-}

No Torque in front cell

A.V.ROE CANADA LIMITED MALTON - ONTARIO TECHNICAL DEPARTMENT (Aircraft)		REPORT NO. 7/0510/7 SHEET NO. 5-25	
		PREPARED BY: D L Turner	DATE: 21 Jan. 55
AIRCRAFT C105	Outer Wing	CHECKED BY: K N Shepley DLT	DATE: May '55

Statically Determinate Stresses.

Since the torque T_{Tip} applied to rib 18 (around the complete wing cell) is taken off into the outer wing through the rear three cells only, bending moments will arise in rib 18, ranging from zero at the front spar to a maximum at the M/S and dropping to zero again at the rear spar.

Rib 18 moment at the M/S will be:

$$M_{R_{M/S}} = -\frac{A_F}{A_C} T_{Tip} = -\frac{85.2331}{406.1289} T_{Tip} = -.209867\, T_{Tip}$$

$A_F = 85.2331$, sht 1-18 = Area of front cell

$A_C = 406.1289$ = Area of complete wing cell

At the F/C/S

$$M_{R_{F/C/S}} = -\frac{A_F + A_{F-}}{A_C} T_{Tip} + T_{F-} = -\left(\frac{174.8608}{406.1289} - .279274\right) T_{Tip} = -.151256\, T_{Tip}$$

At the R/C/S

$$M_{R_{R/C/S}} = -\frac{A_F + A_{F-} + A_{R+}}{A_C} T_{Tip} + T_{F-} + T_{R+}$$

$$= -\left(\frac{284.5937}{406.1289} - .279274 - .342002\right) T_{Tip} = -.079481\, T_{Tip}$$

where M_R is moment in rib at spar denoted by further subscript.

Stresses at the corresponding stress points will be:

$$\overset{+}{\sigma}_{Y_{15}} = \frac{M_{R_{M/S}}}{h_{M/S} A_{r_{M/S}}} = \frac{-.209867}{5.9584 \times 2.6606} T_{Tip} = -.013204\, T_{Tip}$$

$$\overset{+}{\sigma}_{Y_{17}} = \frac{M_{R_{F/C/S}}}{h_{F/C/S} A_{r_{F/C/S}}} = \frac{-.151256}{6.4481 \times 2.6512} T_{Tip} = -.008848\, T_{Tip}$$

$$\overset{+}{\sigma}_{Y_{19}} = \frac{M_{R_{R/C/S}}}{h_{R/C/S} A_{r_{R/C/S}}} = \frac{-.079481}{6.2826 \times 2.6182} T_{Tip} = -.004832\, T_{Tip}$$

$$\overset{+}{\sigma}_{Y_{21}} = 0$$

A. V. ROE CANADA LIMITED MALTON - ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7/0510/7 SHEET NO. 5-27	
		PREPARED BY	DATE
AIRCRAFT: C105	Outer Wing	[illegible]	9 Feb 55
		CHECKED BY	DATE
		R.N. SHEERLY	[illegible] May /55

Statically Determinate Stresses

Consider now the moments (2) of sht. 5-26.

These moments ; M_{θ_5}, M_{θ_4}, M_{θ_3}, M_{θ_2} & M_{θ_1} ; are applied at load points 20, 21, 22, 23 & 24 respectively. They are transmitted through their respective spars to the transport joint as constant moments, except for M_{θ_5}, which is transferred through the front spar only as far as stress point 70. From here it is transferred to the transport joint in the same manner as MQ_{42} of

Consider now M_{θ_5} inboard of load point 42.

This moment is carried through panel 83 to generate moments at stress points 71 & 74 as outlined in (and as was done for MQ_{42} in calculating the elements of the T_2 matrix).

From sht. 5-12.

$$M_{71} = MQ_{42}\frac{b}{a}$$

see sketch on sht 5-28.

since $MQ_{42} = M_{\theta_5}$ here, and $\frac{b}{a} = .792112$ (sht 5-13.)

$$M_{71} = .792112\ M_{\theta_5}$$

now $\sigma_{71} = \dfrac{M_{71}}{h_{71}A_{71}}$ $\qquad \dfrac{1}{h_{71}A_{71}} = .022818$ (sht 5-16)

then $\sigma_{71} = .792112 \times .022818\ M_{\theta_5}$

$$= .018075\ M_{\theta_5}$$

The stress at stress point 74 will be considered when stresses in the transport joint rib are calculated.

FORM 1319A

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

REPORT NO. 7/0510/7
SHEET NO. 5-28

PREPARED BY	DATE
D L Turner	23 Feb. '55
CHECKED BY	DATE
R.N. SHEARLY	MAY /55

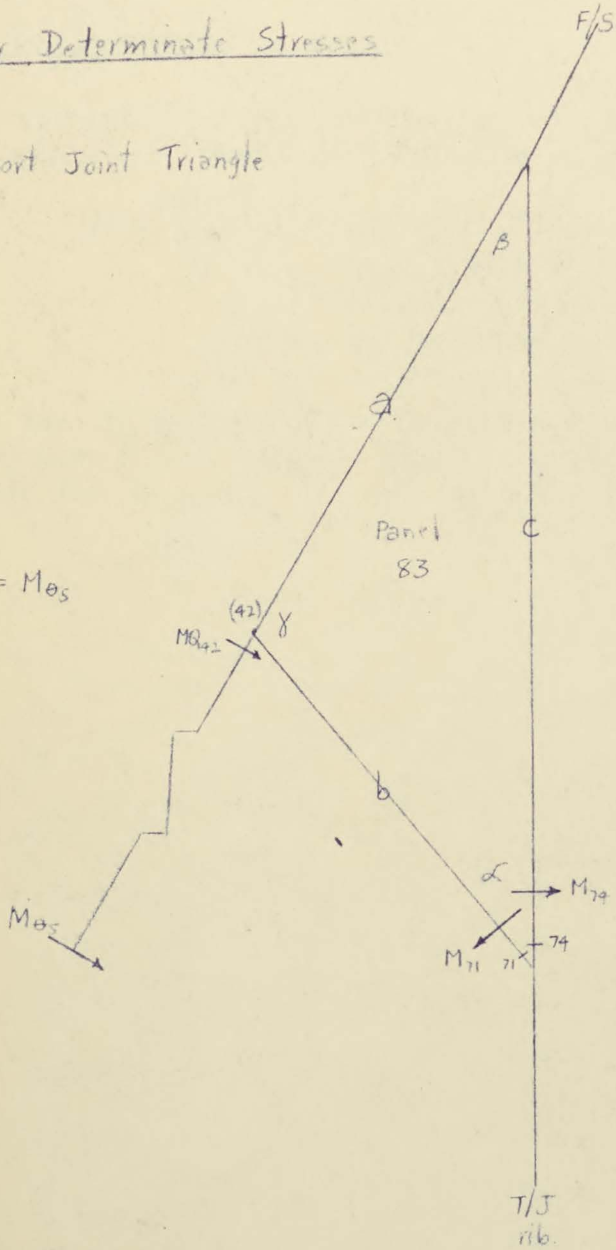
AIRCRAFT: C105

Outer Wing

Statically Determinate Stresses

Transport Joint Triangle

$MQ_{42} = M_{\theta s}$



FORM 1319A

Statically Determinate Stresses

These moments (sht. 5-30) are now resolved into components acting parallel and perpendicular to the transport joint rib as was done for the moments due to the vertical shears (sht. 5-26)

From (1) * $T'_{M/s} = -.792112\ M_{\theta_5} \sin 50.5^\circ = -.792112 \times .77162\ M_{\theta_5}$
$= -.611209\ M_{\theta_5}$
$\bar{M}'_{M/s} = .792112\ M_{\theta_5} \cos 50.5^\circ = .792112 \times .63608\ M_{\theta_5}$
$= .503847\ M_{\theta_5}$

(2) † $T_{M/s} = M_{\theta_4} \sin 54.45^\circ = .81361\ M_{\theta_4}$
† $\bar{M}_{M/s} = M_{\theta_4} \cos 54.45^\circ = .58141\ M_{\theta_4}$

Nose up T +ve
Tension in top of wing $\bar{M}$ +ve

$T_{F/c/s} = M_{\theta_3} \sin 49.91^\circ = .76503\ M_{\theta_3}$
$\bar{M}_{F/c/s} = M_{\theta_3} \cos 49.91^\circ = .643[illegible]\ M_{\theta_3}$

$T_{R/c/s} = M_{\theta_2} \sin 44.32^\circ = .69867\ M_{\theta_2}$
$\bar{M}_{R/c/s} = M_{\theta_2} \cos 44.32^\circ = .715996\ M_{\theta_2}$

$T_{R/s} = M_{\theta_1} \sin 37.41^\circ = .60752\ M_{\theta_1}$
$\bar{M}_{R/s} = M_{\theta_1} \cos 37.41^\circ = .79430\ M_{\theta_1}$

(3) ‡ $T''_{M/s} = 1.475097\ M_{\theta_5}$

Stresses will arise in the transport joint rib due to the redistribution of the concentrated torques through the rear three cells of the wing.

These stresses are determined in the same manner as was done for the T_2 matrix (ref shts. 5-18 et seq

Calculations are done on sht. 5-32

‡ T'' is torque due to portion of M_{θ_5} transferred to load pt. 43 through T/J rib

* $\left.\begin{matrix} T'_{M/s} \\ \bar{M}'_{M/s} \end{matrix}\right\}$ are $\begin{matrix}\text{torque}\\ \text{moment}\end{matrix}$ components, vectors acting $\begin{matrix}\text{perp}\\ \text{parallel}\end{matrix}$ to T/J rib, acting at M/S due to moment at stress point 71

† $\left.\begin{matrix} T_{M/s} \\ \bar{M}_{M/s} \end{matrix}\right\}$ are $\begin{matrix}\text{torque}\\ \text{moment}\end{matrix}$ components, vectors $\begin{matrix}\text{perp}\\ \text{parallel}\end{matrix}$ to T/J rib, acting at M/S due to moment at stress pt 72

FORM 1319A

Determination of Transport Joint Rib Stresses Due to Moment Components of Set P_1

A. V. ROE C[...]
MALT[...]
TECHNICA[...]

AIRCRAFT -
WEIGHT -
C. G. POSITION -

Stress Pt.	1	2	3	4	5	6	7	8	9	10	11	12
	Stress Due to Unit				Stress due to							
	$T'_{M/s}$ or $T_{M/s}$	$T_{F/c/s}$	$T_{R/c/s}$	$T_{R/s}$	$T'_{M/s}$ $\overline{.611209}\times Mo_x(1)$	$T_{M/s}$ $.81361\times Mo_x(1)$	$T_{F/c/s}$ [illegible] $\times Mo_x(2)$	$T_{R/c/s}$ [illegible] $\times Mo_x(3)$	$T_{R/s}$ [illegible] $\times Mo_x(4)$	$T''_{M/s}$ Factor	$\frac{1}{hA}$	(10)[...]
74	$\overline{.003492}$	0	0	0	.002134	$\overline{.002891}$	0	0	0	$\overline{1}.0000$	.00[illegible]	$\overline{.00}$[...]
75	$\overline{.008660}$	$\overline{.001821}$	.005013	.005013	.005293	$\overline{.00[illegible]}$	$\overline{.001393}$	.003[illegible]	.003[illegible]	-.633[illegible]	.013[illegible]	$\overline{.00}$[...]
78	$\overline{.005122}$	$\overline{.005122}$	.003727	.012695	.003131	$\overline{.004[illegible]}$	$\overline{.003318}$	.002696	.002712	-.287453	[illegible]	$\overline{.0}$[...]
81	0	0	0	.005266	0	0	0	0	.003193	0	[illegible]	

* [illegible] 5-17

For data in cols 1-4 see sht 5.1[illegible]

FORM 1543

A. V. ROE CANADA LIMITED MALTON - ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7/0510/2 SHEET NO. 5-33	
		PREPARED BY	DATE
AIRCRAFT: C105	Outer Wing	D. L. [illegible]	9 Feb. 5[illegible]
		CHECKED BY	DATE
		R. N. Sheeley	3rd May 1955

Statically Determinate Stresses

This results in a set of loads at the transport joint consisting of:

(1) A torque of $1.475097 M_{\theta_6} + .611209 M_{\theta_5} + .81361 M_{\theta_4} + .76503 M_{\theta_3}$ $+ .69867 M_{\theta_2} + .60742 M_{\theta_1}$ (nose up +ve).

(2) Moments of $.563597 M_{\theta_6} + .5514 M_{\theta_4}$, $.6739[illegible] M_{\theta_3}$, $.715946 M_{\theta_2}$ and $.79430 M_{\theta_1}$ acting at the M/S, [illegible], R/C/S & R/S respectively. (vectors parallel to 7/3 rib, tension in top +ve)

Consider now the torques (3) of sht. 5-26.

These torques, consisting of:

(1) $.378724 \ T_{Tip}$ in rear cell
(2) $.342062 \ T_{Tip}$ in rear centre cell
(3) $.279274 \ T_{Tip}$ in forward centre cell

will remain constant during transfer inboard to the transport joint. During this transfer they will generate stresses in the outer wing panels as follows:

(1) panels 26, [illegible] & [illegible]
(2) panels 25, 38, 51 & 62
(3) panels 24, 37, 50, [illegible] & 69

The equation used to determine the stresses in these panels is found in report 7/0510/2 Sht 4-5, and is:

$$T_{ns} = \frac{1}{t(h_{n}b_1 + h_1 b_n)} T$$

where T is the appropriate torque as defined above; t is the panel skin thickness, and h & b are cell heights & widths as defined on sht. 5-35 & fig 5

FORM 1319A

A.V.ROE CANADA LIMITED MALTON – ONTARIO TECHNICAL DEPARTMENT (Aircraft)		REPORT NO. 7/0510/7 SHEET NO. 5-34	
		PREPARED BY: D.J. Turner	DATE: 25 Jan 55
AIRCRAFT C105	Outer Wing	CHECKED BY: E.D. SHARPEY	DATE: MAY, /55

Statically Determinate Stresses

For the rear cells, values of the constants $\frac{1}{t(h_{s}b_{s}+h_{u}b_{u})}$ are obtained from report 7/0510/2 sht 4-42. These values, when multiplied by the factor .378724, (the proportion of T_{Tip} carried by these cells), may be entered in the T_1' matrix.

(1) Panel	(2) $\frac{1}{t(h_{s}b_{s}+h_{u}b_{u})}$	(3) .378724 (2)
26	.0232132	.008791
39	.0183889	.006964
52	.0143602	.005439

Since, in the calculations of report 7/0510/2, no torque was carried through the rear centre and forward centre cells, values of the constants for these cells must be calculated here. This is done, together with the final calculation of the elements of the T_1' matrix, in the table on sht. 5-35.

From here these figures are entered directly into the T_1' matrix.

This leaves a torque (vector perp to T/J rib)

$$T_{Tip}\cos\alpha_0 = T_{Tip}\cos 39.50^\circ$$
$$= .77163\,T_{Tip} \qquad \text{(nose up)}$$

and a moment of (vector parallel to T/J rib)

$$-T_{Tip}\sin\alpha_0 = -.63614\,T_{Tip}$$ acting at the transport joint.

Note: 39.50° is the angle between the o/w ribs & the T/J rib.

Determination of the Elements of the T_1' Matrix Due to Rear Centre and Forward Centre Cell Torques.

Outer Wing

$$T_m = \frac{1}{t(h_{22} b_1 + h_{11} b_2)} T$$

A. V. ROE C
MALT
TECHNICA

AIRCRAFT -
WEIGHT -
C. G. POSITION

	1	2	3	4	5	6	7	8	9	10	11
Cell	t	b_1	b_2	h_{11}	h_{22}	$h_{22} b_1$ (2)×(5)	$h_{11} b_2$ (3)×(4)	(6)+(7)	(1)×(8)	1/(9)	
25	.164	17.2913	18.8291	6.2826	7.0511	121.570130	118.6098	240.1799	39.3895	.025387	
38	.172	18.8291	20.7769	6.8683	7.7476	146.30923	142.6985	289.0090	49.7087	.020117	
51	.182	20.7769	22.9989	7.5468	8.5677	178.0060	173.5643	351.[illegible]	63.9855	.015628	
62	.192	22.9989	25.2204	8.3415	[illegible]	215.8589	210.3760	426.2344	81.8320	.012219	
24	.166	14.4469	15.8192	6.4491	[illegible]	94.1360	102.0038	196.1398	32.5592	.030713	
37	.174	15.8192	17.4020	7.0511	7.1619	113.2955	122.7526	236.0481	41.0729	.024347	
50	.184	17.4020	19.2709	7.7496	7.9183	137.8497	149.3418	287.1915	52.8432	.018924	
61	.195	19.2709	21.1328	8.5677	8.6748	167.1712	181.0595	348.2307	67.9050	.014726	
69	.204	21.1328	22.9946	9.3618	9.4312	199.3077	215.8227	415.1304	84.6866	.011808	
23	.167	17.6435	19.3195	5.9594	4.0513	71.4791	115.1133	186.5924	31.1609	.032091	
	Data from fig 5										

FORM 1543

A. V. ROE CANADA LIMITED MALTON - ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**	REPORT NO. 7/0510/7 SHEET NO. 5-36	
AIRCRAFT: C105	PREPARED BY	DATE
Outer Wing	[signature]	10 Feb '55
	CHECKED BY	DATE
	R.N. SHEAREY 2/13	MAY '55

Statically Determinate Stresses

The moment is broken down into three parts, acting on the three cells as follows:

(1) $-.378724 \times .63614\, T_{tip} = -.240922\, T_{tip}$ rear cell

(2) $-.342002 \times .63614\, T_{tip} = -.217561\, T_{tip}$ r/c cell

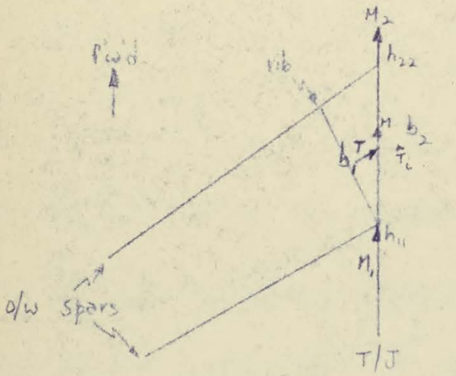
(3) $-.279279 \times .63614\, T_{tip} = -.177657\, T_{tip}$ f/c cell

These three moments are distributed between the spars bounding their respective cells according to the following:
(ref.

$$M_2 = \frac{h_{22} b_1}{h_{22} b_1 + h_{11} b_2} M$$

$$M_1 = \frac{h_{11} b_2}{h_{22} b_1 + h_{11} b_2} M$$

where M_1 is the proportion in the after spar, M_2 is the proportion in the forward spar, and other data is as defined in the sketch below.



h are heights

b are lengths

T is appropriate proportion of T_{tip}.

T_1 is component of T giving torque inboard of T/J rib.

M is component of T giving bending moment on inner wing

$M_1 + M_2 = M$

A. V. ROE CANADA LIMITED MALTON - ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7/0510/7	
		SHEET NO. 5-39	
AIRCRAFT: C105	Outer Wing	PREPARED BY	DATE
		D.L. Turner	11 Feb. 55
		CHECKED BY	DATE
		R.N. SHERRY	23 MAY /55

Statically Determinate Stresses

Summary of T_1' elements calculated in this section

Rib 18 stresses due to T_{TIP}

$\sigma_{15} = -.013204\ T_{TIP}$

$\sigma_{17} = -.008848\ T_{TIP}$ ref sht. 5-25

$\sigma_{19} = -.004832\ T_{TIP}$

From Moments ref sht. 5-32.

$\sigma_{74} = -.012615\ M_{\theta_5} - .002841\ M_{\theta_4}$

$\sigma_{75} = -.007481\ M_{\theta_5} - .007046\ M_{\theta_4} - .001335\ M_{\theta_3} + .003507\ M_{\theta_2} + .003049\ M_{\theta_1}$

$\sigma_{78} = -.004429\ M_{\theta_5} - .004167\ M_{\theta_4} - .003918\ M_{\theta_3} + .002646\ M_{\theta_2} + .007712\ M_{\theta_1}$

$\sigma_{81} = .003199\ M_{\theta_1}$

$\sigma_{71} = .018075\ M_{\theta_5}$

From Torques ref sht 5-35

$\tau_{25} = .008682\ T_{TIP}$ $\tau_{24} = .008577\ T_{TIP}$

$\tau_{38} = .006880\ T_{TIP}$ $\tau_{37} = .006799\ T_{TIP}$

$\tau_{51} = .005345\ T_{TIP}$ $\tau_{50} = .005285\ T_{TIP}$

$\tau_{62} = .004179\ T_{TIP}$ $\tau_{61} = .004113\ T_{TIP}$

$\tau_{63} = .003298\ T_{TIP}$

FORM 1319A

Report No 7/0510/7
Sht. 5-40
11 Feb. '55
D.L. Turner

Dummy "T₁" Matrix

	Mox	Mox	Mox	Mox	Mx	Tm	Pos	Pos	Pos	Pos	Pos
14	✓										
15						✓					
16		✓									
17						✓					
18			✓								
19						✓					
20				✓							
21											
22					✓						
23											
24						✓					
25						✓					
26						✓					
27	✓						✓				
29		✓						✓			
31									✓		
33				✓						✓	
34											
35					✓						✓
36											
37						✓					
38						✓					
39						✓					
40	✓						✓				
42		✓						✓			
44			✓						✓		
46				✓						✓	
48					✓						✓
49											
50						✓					
51						✓					
52						✓					
53	✓						✓				
55		✓						✓			
57			✓						✓		
59				✓						✓	
60											
61						✓					
62						✓					
63	✓						✓				
65		✓						✓			
67			✓						✓		
68											
69						✓					
70	✓						✓				
71	✓						✓				
72		✓						✓			
74	✓	✓					✓	✓			
75	✓	✓	✓	✓	✓		✓	✓	✓	✓	✓
78	✓	✓	✓	✓	✓		✓	✓	✓	✓	✓
81	✓				✓						✓

Statically Determinate Stresses

Summary of Loads at Transport Joint

From Vertical Shears

$$T = {}^{\times}161.2442\,P_{\theta_5} - {}^{\times}88.4831\,P_{\theta_5} + {}^{\dagger}86.3677\,P_{\theta_4} + {}^{\dagger}62.4010\,P_{\theta_3}$$
$$+ {}^{\dagger}40.5850\,P_{\theta_2} + {}^{\dagger}21.6072\,P_{\theta_1}$$ (rear three bays, nose up +ve)

$M_{43} = 72.9405\,P_{\theta_5} + {}^{\ddagger}61.7217\,P_{\theta_4}$ at M/s

$M_{41} = {}^{\ddagger}52.5252\,P_{\theta_3}$ at F/c/s

$M_{38} = {}^{\dagger}41.5595\,P_{\theta_2}$ at R/c/s

$M_{34} = {}^{\dagger}28.2502\,P_{\theta_1}$ at R/s

tension in top +ve

$V_{43} = P_{\theta_5} + P_{\theta_4}$

$V_{41} = P_{\theta_3}$

$V_{38} = P_{\theta_2}$

$V_{34} = P_{\theta_1}$

$\times$ ref. sht 5-15
$\ddagger$ ref sht. 5-48 via sht. 5-18
$\dagger$ ref sht. 5-18
$**$ ref sht. 5-33

From Moments

$$T = {}^{**}1.475097\,M_{\theta_5} + {}^{**}.611203\,M_{\theta_5} + {}^{**}.81361\,M_{\theta_4} + {}^{**}.76503\,M_{\theta_3}$$
$$+ {}^{**}.6[illegible]867\,M_{\theta_2} + {}^{**}.60752\,M_{\theta_1}$$ (rear three bays, nose up +ve)

$M_{43} = {}^{**}.503847\,M_{\theta_5} + {}^{**}.58191\,M_{\theta_4}$ at M/s

$M_{41} = {}^{**}.64399\,M_{\theta_3}$ at F/c/s

$M_{38} = {}^{**}.715446\,M_{\theta_2}$ at R/c/s

tension in top +ve

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

AIRCRAFT: C105	Outer Wing	REPORT NO. 7/0510/7	SHEET NO. 5-43
		PREPARED BY	DATE
		D.L. Turner	11 Feb. '55
		CHECKED BY	DATE
		E.N. Shearer	20 May /55

Statically Determinate Stresses

Determination of Matrix H_1

The elements of the H_1 matrix consist of the coefficients of the set P_1, for P_2 written in terms of P_1.

The set P_2 consists of a torque T_0 at the transport joint acting on the inner wing (vector perpendic. to T/J rib, nose up +ve), moments $M\phi_{50}$, $M\phi_{40}$, $M\phi_{30}$ & $M\phi_{20}$ (vectors parallel to T/J rib, tension in top surface +ve) acting at the M/S, F/c/s, R/c/s & R/S respectively, and vertical shears $P\phi_{50}$, $P\phi_{40}$, $P\phi_{30}$ & $P\phi_{20}$ (down +ve) acting at the M/S, F/c/S, R/c/S & R/S respectively.

From the equations of shts 5-41 & 42 it can be seen that T represents T_0; M_{43}, M_{41}, M_{38} & M_{34} represent $M\phi_{50}$, $M\phi_{40}$, $M\phi_{30}$ & $M\phi_{20}$ respectively; and that V_{43}, V_{41}, V_{38} & V_{34} represent $P\phi_{50}$, $P\phi_{40}$, $P\phi_{30}$ & $P\phi_{20}$ respectively.

Thus we may write:

$$T_0 = 72.7611\,P_{\theta_5} + 86.3677\,P_{\theta_4} + 62.4010\,P_{\theta_3} + 40.5850\,P_{\theta_2} + 21.6072\,P_{\theta_1} + .863838\,M_{\theta_5} + .81361\,M_{\theta_4} + .76503\,M_{\theta_3} + .69867\,M_{\theta_2} + .60752\,M_{\theta_1} + .77163\,T_{tip}.$$

$$M\phi_{50} = 72.9405\,P_{\theta_5} + 61.7217\,P_{\theta_4} + .503897\,M_{\theta_5} + .58191\,M_{\theta_4} - .085295\,T_{tip}.$$

$$M\phi_{40} = 52.5282\,P_{\theta_3} + .64399\,M_{\theta_3} - .202542\,T_{tip}$$

$$M\phi_{30} = 41.5595\,P_{\theta_2} + .715446\,M_{\theta_2} - .237648\,T_{tip}$$

$$M\phi_{20} = 28.2502\,P_{\theta_1} + .79430\,M_{\theta_1} - .110655\,T_{tip}$$

$$P\phi_{50} = P_{\theta_5} + P_{\theta_4} \qquad P\phi_{20} = P_{\theta_1}$$

$$P\phi_{40} = P_{\theta_3}$$

$$P\phi_{30} = P_{\theta_2}$$

The elements of this matrix are tabulated on sht 5-44.

FORM 1319A

A. V. ROE CA
MALTON
TECHNICAL D

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

Elements of H_1 Matrix

$P_1 \rightarrow$ / $P_2 \downarrow$	1	2	3	4	5	6	7	8	9	10	11	12
	M_{θ_5}	M_{θ_4}	M_{θ_3}	M_{θ_2}	M_{θ_1}	T_{θ}	P_{θ_5}	P_{θ_4}	P_{θ_3}	P_{θ_2}	P_{θ_1}	
$M_{\phi_{5_0}}$	.503847	.58141	—	—	—	$\overline{.083295}$	72.[illegible]	61.7217	—	—	—	
$M_{\phi_{4_0}}$	—	—	.64399	—	—	$\overline{.202542}$	—	—	52.5282	—	—	
$M_{\phi_{3_0}}$	—	—	—	.715446	—	$\overline{.237648}$	—	—	—	41.5595	—	
$M_{\phi_{2_0}}$	—	—	—	—	.72430	$\overline{.110625}$	—	—	—	—	28.2502	
T_0	.863838	.81361	.76503	.69867	.60752	.7763	72.7611	86.3677	62.4010	40.[illegible]	21.6072	
$P_{\phi_{5_0}}$	—	—	—	—	—	—	1.00000	1.00000	—	—	—	
$P_{\phi_{4_0}}$	—	—	—	—	—	—	—	—	1.00000	—	—	
$P_{\phi_{3_0}}$	—	—	—	—	—	—	—	—		1.00000	—	
$P_{\phi_{2_0}}$	—	—	—	—	—	—	—	—	—	—	1.00000	
Check row	1.367628	1.390223	1.[illegible]	[illegible]116	1.40[illegible]	.135430	146.7016	14[illegible].0894	[illegible]	[illegible].144	50.8574	

FORM 154

Statically Determinate Stresses

Determination of the G_2 Matrix.

The G_2 matrix consists of the coefficients of the loads on the outer wing ($Q_{o/w}$), for P_2 written in terms of $Q_{o/w}$.

The set P_2 consists of a torque T_0 at the transport joint acting on the inner wing (vector perp. to T/J rib, nose up +ve), moments $M\phi_{50}$, $M\phi_{41}$, $M\phi_{30}$ & $M\phi_{20}$ (vectors parallel to T/J rib, tension in top surface +ve) acting at the M/S, F/c/s, R/c/s & R/s respectively and vertical shears $P\phi_{50}$, $P\phi_{40}$, $P\phi_{30}$ & $P\phi_{20}$ (down +ve) acting at the M/S, F/c/s, R/c/s & R/s respectively.

For $Q_{o/w}$ applied at load points on the M/S, F/c/s, R/c/s & R/s, elements of the G_2 matrix may be calculated as follows:

(1) Determine the spar moment at the T/J due to each unit load Q. (These moment arms may be obtained from report 7/0510/2 sht 4-45)

(2) Resolve these moments into components perpendicular & parallel to the T/J rib. This gives the values of moments & torques at the T/J due to these loads.

This calculation is done in the table on sht. 5-48.

Note: The appropriate angles and their sin & cos values are given in fig. 5

$Q_{o/w}$ applied to the front spar are carried along the front spar to load point 42, from whence they are transferred across the forward triangular panel of the outer wing to load point 43. The moments at load point 43 due to this transfer were calculated during the computation of the elements of the T_2 matrix, the results being tabulated on shts. 5-48 & 49. These results give a torque M' (vector perp to T/J) and a moment M_{71} (vector perp to f'wd i/b rib of o/w)

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

REPORT NO. 7/0510/7	SHEET NO. 5-46	
AIRCRAFT: C105	Outer Wing	PREPARED BY: [illegible] — DATE: 14 Feb 55
		CHECKED BY: R. H. Shearly — DATE: 20 MAY /55

Statically Determinate Stresses.

The required matrix elements may be obtained by resolving M_{71} into components parallel & perpendicular to the T/J rib, and combining the latter (torque) component with M'.
This calculation is done in the table on sht. 5-49.

For $Q_{o/w}$ applied aft of the rear spar (load points 13 & 14).
These loads are applied 15 in. aft of the rear spar. From their load points they are transferred forward along their respective ribs to the rear spar, giving rise to torques $T'_{13} = 15.000\,Q_{13}$ & $T'_{14} = 15.000\,Q_{14}$ at load points 24 & 29 resp.
These torques travel to the T/J through the rear wing cells (cells 26, 39 & 52) to the T/J, where they are resolved into components perpendicular & parallel to the T/J rib. The torque components ($T' \cos 39.50°$) of this resolution are carried directly to the inner wing. The wing bending components ($T' \sin 39.50°$) are distributed between load points 39 & 38 according to the distributions given on sht. 5-36. et seq. where it is shown that.

$$\left.\begin{aligned} M_{38} &= -.5407018\ M \\ M_{39} &= -.4592982\ M \end{aligned}\right\} \text{sht. 5-37}$$

where $M = T' \sin 39.50°$
& subscript on M indicates moment at load point denoted by subscript.

The loads, after transfer to the rear spar, travel inboard along the rear spar to the T/J, resulting in a vertical shear at the R/S – T/J intersection of $Q_{13} + Q_{14}$, and a bending moment at the R/S (vector perp. to R/S, tension in top flange) of:

$$M = 35.5662\ Q_{13} + 19.0884\ Q_{14}$$

see load 24 & 29 moment arms, sht. 5-48

Resolving this into components perpendicular and parallel to the T/J rib gives:

FORM 1319A

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

REPORT NO. 7/0510/2
SHEET NO. 5-47

AIRCRAFT: C105

Outer Wing

PREPARED BY	DATE
[illegible]	14 Feb 55
CHECKED BY	DATE
[illegible]	May 1955

Statically Determinate Stresses

$M_1 = 28.25023\ Q_{13} + 15.16192\ Q_{14}$

$T_1 = 21.60718\ Q_{13} + 11.59658\ Q_{14}$

Note: This resolution was previously carried out for loads 24 & 29. Results here are taken from sht. 5.48.

Thus, at load point 34

$M_{34} = -.4592982\ T' \sin 39.50 + 28.25023\ Q_{13} + 15.16192\ Q_{14}$

$= -.4592982 \times 15.0000\ Q_{13} \sin 39.50^\circ + 28.25023\ Q_{13}$
$-.4592982 \times 15.0000\ Q_{14} \sin 39.50^\circ + 15.16192\ Q_{14}$

& at load point 38

$M_{38} = -.5407018\ T' \sin 39.50$

$= -.5407018 \times 15.0000\ Q_{13} - .5407018 \times 15.0000\ Q_{14}$

$T = T' \cos 39.50$

$= 15.0000\ Q_{13} \cos 39.50^\circ + 15.0000\ Q_{14} \cos 39.50^\circ$

M_{34} & M_{38} +ve produce tension in top skin (vector parallel to T/J rib.)

T +ve denotes nose up torque (vector perp. to T/J)

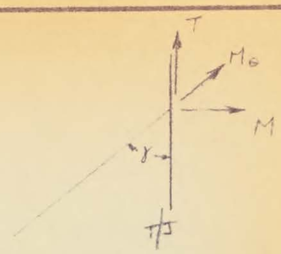
These calculations are carried out on sht. 5-50.

The G_2 matrix resulting is tabulated on sht. 5-51.

FORM 1319A

G_2 MATRIX

FORCES & MOMENTS AT T/J
DUE TO UNIT LOADS ON OUTSIDE WINGS



A. V. ROE
M
TECH

AIRCRAFT
WEIGHT
C. G. POSITIO

	1	2	3	4	5	6	7	8	9	10	11
LOAD POINT	FORCE AT POINT	MOMENT ARM	COS γ	SIN γ	M ①·④	T ②·③					
21	43 (ϕ_5)	106.1587	.81361	.58141	61.72173	86.36771					COS
26	"	89.1147	"	"	51.81213	72.50461					
31		69.3704		"	40.33264	56.44045		M/S			
36	"	46.2470			26.88847	37.62702					
40		23.1234			13.44418	18.81343					
22	41 (ϕ_4)	81.5668	.76503	.64393	52.52820	62.40105					
27	"	64.8242	"		41.74614	49.59246		F/c/s			
32	"	45.4292			29.25535	34.75470					
37		22.7147			14.62804	17.37743					
23	38 (ϕ_3)	58.0889	.69867	.715446	41.55947	40.58497					
28	"	41.5634			29.73637	29.03910		R/c/s			
33	"	22.4199			16.04023	15.66411					
24	34 (ϕ_2)	35.5662	.60752	.79430	28.25023	21.60715		R/S			
29	"	19.0884	"		15.16192	11.59658					

FORM 1543

A. V. ROE CA

MALTON

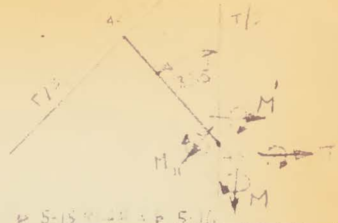
TECHNICAL

AIRCRAFT -

WEIGHT -

C. G. POSITION -

[illegible] [illegible] [illegible] [illegible]
[illegible] [illegible] [illegible]
[illegible] on [illegible] [illegible]



Values of M_{11} & M' [illegible] From [illegible] p 5-15 [illegible] p 5-16

	1	2	3	4	5	6	7	8	9	10	11	12
Load Point	M_{11}	M'	M_{11} [illegible] .7762 (1)	M [illegible]	T (2) - (3)							
20	114.6[illegible]	[illegible]	88.4[illegible]	72.3405	72.7611							
25	100.7638	135.3561	77.7560	64.0976	57.6001		M = M_{11} sin 32[illegible]					
30	84.6652	105.3666	65.3294	53.8638	40.0372							
35	65.8044	70.2446	50.7760	41.8569	19.4685							
[illegible]	46.9433	35.1220	36.2224	29.8597	-1.1004							
42	28.0825	0	21.6690	17.8627	-21.6690							

FORM 1543

A. V. R

EN

AIRCRAF

WEIGHT

C. G. PO

	7	8	9	10	11	12	13	14	15	16	17	18	19	20	
	24	25	26	27	28	29	30	31	32	33	35	36	37	39	
		64.0976	51.81[illegible]				53.8538	40.33264			41.8569	26.85597		29.8597	13.[illegible]
				41.79614					29.25595				14.62809		
47					29.73637					16.04023					
	28.25023					15.16192									
[illegible]	21.60718	57.6001	72.50461	49.59246	29.03910	11.59658	40.0372	56.4404[illegible]	34.75970	15.66911	19.4685	37.62702	17.37793	1.1004	18.[illegible]
		1.0000	1.0000				1.0000	1.0000			1.0000	1.0000		1.0000	1.0[illegible]
				1.0000					1.0000				1.0000		
					1.0000					1.0000					
	1.0000					1.0000									
4	50.85741	122.6977	125.31679	92.33860	59.77547	27.75850	94.8910	97.77309	65.01065	32.70939	62.3254	65.51549	33.00547	29.7523	33.[illegible]

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

REPORT NO. 7/0510/7
SHEET NO. 5.33

AIRCRAFT: C-105	Inner Wing	PREPARED BY: R.N. SHEARLY	DATE: FEB 7/55
		CHECKED BY: R.N. SHEARLY	DATE:

DETERMINATION OF T_2' MATRIX (CONT)

DUE TO SPAR KINK MOMENT AT RK/S

M/S	−.360600	×	0	=	0
F/C/S	−.360600	×	.366899	=	−.132304
R/C/S	−.360600	×	.212547	=	−.076644
R/S	−.360600	×	0	=	0

DUE TO SPAR KINK MOMENT AT R/S

M/S	−.197664	×	0	=	0
F/C/S	−.197664	×	.366899	=	−.072523
R/C/S	−.197664	×	.712547	=	−.140845
R/S	−.197664	×	.211137	=	−.041734

THE TORSIONAL MOMENT ACTING ON THE TRANSPORT JOINT RIB IS REACTED BY THE TORSIONAL MOMENTS IN THE CELLS BETWEEN THE M/S & R/S. CALLING THE CELL BETWEEN
M/S & F/C/S CELL 84
F/C/S & R/C/S CELL 85
R/C/S & R/S CELL 86

THEN SINCE THE SHEAR FLOW AROUND THE CELLS IS TO BE CONSTANT AT THE T.J.*

$$\frac{T_{84}}{A_{84}} = \frac{T_{85}}{A_{85}} = \frac{T_{86}}{A_{86}} = \frac{(T)_0}{A_{84}+A_{85}+A_{86}}$$

WHERE A_{84}, A_{85}, A_{86} = AREA OF CELL 84, 85, 86

T_{84}, T_{85}, T_{86} = TORQUE CARRIED BY CELL 84, 85, 86

$(T)_0$ = TOTAL TORQUE

$A_{84} = 330.443$ $A_{85} = 311.305$ $A_{86} = 258.891$ REF SHT 7-9

$A_{84} + A_{85} + A_{86} = 900.639$

* THIS PRODUCES STRESSES IN T.J. RIB 1=75 & 79 SEE SHT 5-120, 121 & 122

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

REPORT NO. 7/0510/7
SHEET NO. 5-54

AIRCRAFT:		PREPARED BY	DATE
C-105	Inner Wing	R.N. SHEARLY	FEB. 7/55
		CHECKED BY	DATE
		R.N. SHEARLY DIS	MAY /55

DETERMINATION OF T_2' MATRIX (CONT)

$$T_{84} = \frac{330.443}{900.639}(T)_0 = .366898\,(T)_0$$

$$T_{85} = \frac{311.305}{900.639}(T)_0 = .345649\,(T)_0$$

$$T_{86} = \frac{258.891}{900.639}(T)_0 = .287453\,(T)_0$$

THE SPAR KINK MOMENTS (SEE SHEET 5-52) PRODUCE THE FOLLOWING TORSIONAL MOMENTS IN CELLS 84, 85 & 86

DUE TO SPAR KINK MOMENT AT M/S

CELL 84 = −.686472 × .366898 = −.251865
85 = −.686472 × .345649 = −.237278
86 = −.686472 × .287453 = −.197329

DUE TO SPAR KINK MOMENT AT F/C/S

CELL 84 = −.523536 × .366898 = −.192084
85 = −.523536 × .345649 = −.180960
86 = −.523536 × .287453 = −.150492

DUE TO SPAR KINK MOMENT AT F/C/S

CELL 84 = −.360600 × .366898 = −.132303
85 = −.360600 × .345649 = −.124641
86 = −.360600 × .287453 = −.103656

DUE TO SPAR KINK MOMENT AT R/S

CELL 84 = −.197664 × .366898 = −.072523
85 = −.197664 × .345649 = −.068322
86 = −.197664 × .287453 = −.056819

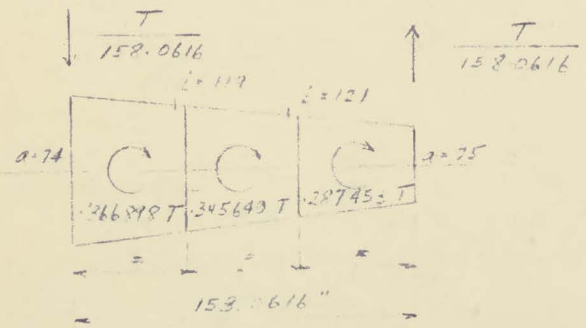
FORM 1319A

A. V. ROE CANADA LIMITED MALTON - ONTARIO TECHNICAL DEPARTMENT (Aircraft)		REPORT No. 7/0510/7 SHEET No. 5.55	
AIRCRAFT: C-105	Inner Wing	PREPARED BY R.N. SHEARLY	DATE FEB 8/55
		CHECKED BY R.N. SHEARLY	DATE MAY /55

DETERMINATION OF T_2' MATRIX (CONT)

A UNIT TORQUE, T APPLIED TO THE INNER WING AT THE TRANSPORT JOINT RIB DIVIDES INTO THE THREE TORQUES

$$\left.\begin{array}{l} .366898\ T \\ .345649\ T \\ .287453\ T \end{array}\right\} \text{ACTING IN THE CELL BETWEEN} \left\{\begin{array}{l} M/S \ \& \ F/C/S \\ F/C/S \ \& \ R/C/S \\ R/C/S \ \& \ R/S \end{array}\right.$$



AT THE FUSELAGE SIDE BEAM THE UNIT TORQUE IS REACTED BY A COUPLE ACTING AT LOAD POINTS a = 74 & 75. THE RESULTING RIB MOMENTS ARE

$$M_{119} = \frac{158.0616}{3} \cdot \frac{T}{158.0616} - .366898\,T = -.033565\,T$$

$$M_{121} = \frac{2}{3} \cdot 158.0616 \cdot \frac{T}{158.0616} - .366898\,T - .345649\,T =$$

$$= -.045880\,T$$

	1	2	3
SOURCE	APPLIED TORQUE, T	M_{119} $-.033565 \times$ 1	M_{121} $-.045880 \times$ 1
$(T)_0$	1.000000	$-.033565$	$-.045880$
$(M_{45})_0$	$-.686472$	.023041	.031495
$(M_{44})_0$	$-.523536$	.017572	.024020
$(M_{43})_0$	$-.360620$	.012104	.016544
$(M_{42})_0$	$-.197664$	.006634	.009069

FORM 1319A

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C-105	Inner Wing	REPORT NO. 7/0510/7	SHEET NO 5-56
		PREPARED BY: R.N. SHEARLY	DATE: FEB 23/55
		CHECKED BY: R.N. SHEARLY	DATE: MAY /55

DETERMINATION OF T_2' MATRIX (CONT.)

THE GENERALIZED UNIT FORCE $(P_{45})_0$ CAUSES A BENDING MOMENT AT STRESS POINTS, i = 87, 97, 107 & 117 EQUAL TO THE MOMENT ARM BETWEEN THE LOAD POINT a = 43 & THE RESPECTIVE STRESS POINT (SEE REPORT 7/0510/2 SHT 4-46)

THE GENERALIZED UNIT FORCE $(P_{46})_0$ CAUSES A BENDING MOMENT AT STRESS POINTS, i = 88, 91, 108 & 118 EQUAL TO THE MOMENT ARM BETWEEN THE LOAD POINT a = 41 & THE RESPECTIVE STRESS POINT (SEE REPORT 7/0510/2 SHT 4-46)

THE GENERALIZED UNIT FORCE $(P_{47})_0$ CAUSES A BENDING MOMENT AT STRESS POINTS, i = 90, 100, 110 & 120 EQUAL TO THE MOMENT ARM BETWEEN LOAD POINT a = 38 & THE RESPECTIVE STRESS POINT (SEE REPORT 7/0510/2 SHT 4-46)

THE GENERALIZED UNIT FORCE $(P_{48})_0$ CAUSES A BENDING MOMENT AT STRESS POINTS, i = 92, 102, 112 & 122 EQUAL TO THE MOMENT ARM BETWEEN LOAD POINT a = 34 & THE RESPECTIVE STRESS POINT (SEE REPORT 7/0510/2 SHT 4-46)

FORM 1310A

…NG

…ACTORS

…MATI…

5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20
	$\frac{1}{h A_R}$	$\frac{1}{.150 \psi}$		STRESS POINT	h	A_S	A_R	$\frac{1}{\psi'}$	$\frac{1}{h A_S}$	$\frac{1}{h A_R}$	$\frac{1}{.150 \psi'}$		STRESS POINT	h	
	.0099993			93	6.4011		27256			.05731637			112	6.4011	5
	.0118791			94				.0011355			.00750333		113	6.4011	
	.0178165			95				.0012712			.00847467		114		
	.0244…			96				.0016121			.01074733		115		
…020				97	11.3242	5.6839			.01475449				116		
…3044				98	10.8764	6.9856			.01316167				117	15.0010	5
…3999				100	8.9040	7.1626			.01567993				118	12.7162	8
…3184				102	6.4011	5.0976			.03064641				119	12.7162	1
		.00914067		103	6.4011		3.0165			.05178354			120	9.5984	8
		.00390667		104				.00032340			.00615600		121	9.5384	
		.01211667		105				.0010846			.00723067		122	6.4011	5
…456				106				.0014275			.00351667		123	6.4011	
…5598				107	13.3734	5.6050			.01334082						
…895				108	11.7430	7.7310			.01101501						
…830				110	9.8310	7.9269			.01366620						

A. V. ROE CANADA LIMITED
MALTON, ONTARIO
ENGINEERING DIVISION — AIRCRAFT

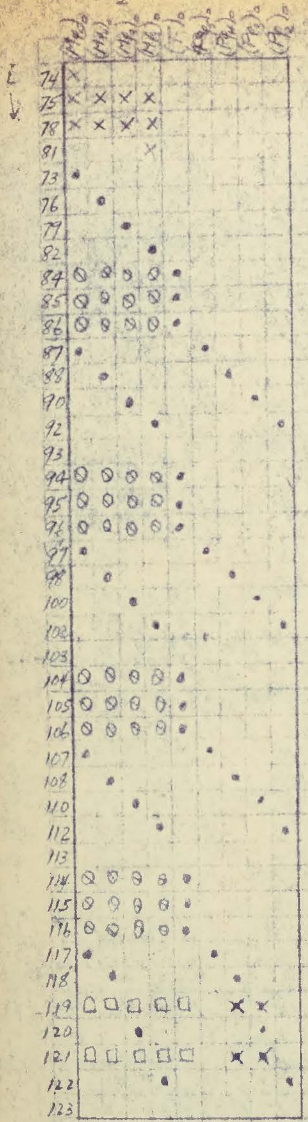
AIRCRAFT - - C105	REPORT NO. - 7/0510/7
WEIGHT - -	SHEET - - 5-59
C. G. POSITION -	DATE - - FEB. 9 '55
	PREPARED BY - V F GARDENER

	17	18	19	20	21	22	23	24	25	26	27	28	29
ψ'		STRESS POINT	h	A_s	A_R	$\frac{1}{\psi'}$	$\frac{1}{hA_s}$	$\frac{1}{hA_R}$	$\frac{1}{150\psi'}$				
		112	6 4011	5.4950			.02843005						
.33		113	6 4011		3.3383			.04673721					
67		114				.00075674			00504493				
33		115				.00032274			00615160				
		116				.0012616			.00841067				
		117	15.0010	5.9333			.01123526						
		118	12.7162	8.5560			.00919119						
		119	12.7162		4.4982			01748251					
		120	9 5984	8.7727			.01187593						
00		121	9.5984		4.4982			.02316127					
67		122	6 4011	5.9348			02632323						
67		123	6.4011										

DUMMY OF T_2' MATRIX

REPORT 7105107 SHT 5-60

R.N. SHEARER FEB/55



PRELIMINARY T_2^{L} MATRIX

i	$(M_{4/5})_0$	$(M_{4/4})_0$	$(M_{4/3})_0$	$(M_{4/2})_0$	$(T)_0$	$(P_{4/5})_0$	$(P_{4/4})_0$	$(P_{4/3})_0$	$(P_{4/2})_0$	STRESS FACTOR REF SNP 5-59
74	.239768									.009999
75	.434606	.069683	-.132304	-.072523						.013679
78	.197328	.150492	-.076644	-.140845						.017816
81				-.041734						.029941
73	1.21295									.017610
76		1.12875								.018680
79			1.06303							.020500
82				1.01935						.035392
84	-.251865	-.192084	-.132303	-.072523	.366898					.009141
85	-.237278	-.180960	-.124641	-.068322	.345649					.009307
86	-.197329	-.150492	-.103656	-.056819	.287453					.012117
87	1.21295					27.8986				.014255
88		1.12875					25.9619			.015696
90			1.06303					24.4503		.017949
92				1.01935					23.4456	.032969
93										.057317
94	-.251865	-.192084	-.132303	-.072523	.366898					.007503
95	-.237278	-.180960	-.124641	-.068322	.345649					.008475
96	-.197329	-.150492	-.103656	-.056819	.287453					.010747
97	1.21295					58.7740				.014754
98		1.12875					54.6940			.013162
100			1.06303					51.5095		.015680
102				1.01935					49.3929	.030646
103										.051790
104	-.251865	-.192084	-.132303	-.072523	.366898					.006156
105	-.237278	-.180960	-.124641	-.068322	.345649					.007231
106	-.197329	-.150492	-.103656	-.056819	.287453					.009517
107	1.21295					92.9439				.013341
108		1.12875					86.4919			.011015
110			1.06303					81.4560		.013666
112				1.01935					78.1089	.028430

										[illegible]	[illegible]	[illegible]	
85	$\overline{.237278}$	$\overline{.180960}$	$\overline{.124641}$	$\overline{.068322}$	.345649					.009907	$\overline{.002351}$	$\overline{.001793}$	
86	$\overline{.197329}$	$\overline{.150492}$	$\overline{.103656}$	$\overline{.056819}$	.287453					.012117	$\overline{.002391}$	$\overline{.001824}$	
87	1.21295					27.8986				.014255	.017251		
88		1.12875					25.9619			.015696		.017717	
90			1.06303					24.4503		.017949			
92				1.01935					23.4456	.032969			
93										.057317			
94	$\overline{.251865}$	$\overline{.192084}$	$\overline{.132303}$	$\overline{.072523}$	.366898					.007503	$\overline{.001830}$	$\overline{.001441}$	[illegible]
95	$\overline{.237278}$	$\overline{.180960}$	$\overline{.124641}$	$\overline{.068322}$	.345649					.008475	$\overline{.002011}$	$\overline{.001534}$	[illegible]
96	$\overline{.197329}$	$\overline{.150492}$	$\overline{.103656}$	$\overline{.056819}$	.287453					.010747	$\overline{.002121}$	$\overline{.001617}$	[illegible]
97	1.21295					58.7740				.014754	.017896		
98		1.12875					54.6940			.013162		.014857	
100			1.06303					51.5095		.015680			[illegible]
102				1.01935					49.3929	.030646			
103										.051790			
104	$\overline{.251865}$	$\overline{.192084}$	$\overline{.132303}$	$\overline{.072523}$	.366898					.006156	$\overline{.001550}$	$\overline{.001182}$	[illegible]
105	$\overline{.237278}$	$\overline{.180960}$	$\overline{.124641}$	$\overline{.068322}$	.345649					.007231	$\overline{.001716}$	$\overline{.001309}$	[illegible]
106	$\overline{.197329}$	$\overline{.150492}$	$\overline{.103656}$	$\overline{.056819}$	.287453					.009517	$\overline{.001878}$	$\overline{.001432}$	[illegible]
107	1.21295					92.9439				.013341	.016182		
108		1.12875					86.4919			.011015		.012433	
110			1.06303					81.4560		.013666			[illegible]
112				1.01935					78.1089	.028430			
113										.046797			
114	$\overline{.251865}$	$\overline{.192084}$	$\overline{.132303}$	$\overline{.072523}$	.366898					.005045	$\overline{.001271}$	$\overline{.000969}$	[illegible]
115	$\overline{.237278}$	$\overline{.180960}$	$\overline{.124641}$	$\overline{.068322}$	.345649					.006152	$\overline{.001460}$	$\overline{.001113}$	[illegible]
116	$\overline{.197329}$	$\overline{.150492}$	$\overline{.103656}$	$\overline{.056819}$	.287453					.008411	$\overline{.001660}$	$\overline{.001266}$	[illegible]
117	1.21295					130.7596				.011235	.013627		
118		1.12875					121.6826			.009191		.010374	
119	.023041	.017572	.012104	.006634	$\overline{.033565}$		$\overline{35.1248}$	$\overline{17.5624}$		.017483	.000403	.000307	[illegible]
120			1.06303					114.5978		.011876			[illegible]
121	.031495	.024020	.016544	.009069	$\overline{.045880}$		$\overline{17.5624}$	$\overline{35.1248}$		.023161	.000729	.000556	[illegible]
122				1.01935						.026323			
123									109.8888				

* N.B. THE C
$(P_{4})_0$, $(P_{8})_0$ & $(P_{R})_0$ OF
THE COEFFICIENTS TA
38 & 34 RESP.
CONSISTENT & ELIM

T_2^{-1} MATRIX

T_2' MATRIX

	$(M/_2)_0$	$(T)_0$	$(P/_5)_0$	$(P/_4)_0$	$(P/_3)_0$	$(P/_2)_0$	STRESS FACTOR REV SHT 5-59	$(M/\phi_5)_0$	$(M/\phi_4)_0$	$(M/\phi_3)_0$	$(M/\phi_2)_0$	$(T)_0$	$(P/\phi_5)_0$ *	
							.009999	.002397						
4	$\overline{.072523}$						.013679	.005945	.000953	$\overline{.001810}$	$\overline{.000992}$			
4	$\overline{.140845}$						.017816	.003516	.002681	$\overline{.001365}$	$\overline{.002509}$			
	$\overline{.041734}$						.024941				$\overline{.001041}$			
							.017610	.021360						
							.018680		.021085					
3							.020500			.021792				
	1.01935						.035392				.036077			
3	$\overline{.072523}$	.366898					.009141	$\overline{.002302}$	$\overline{.001756}$	$\overline{.001209}$	$\overline{.000663}$	.003354		
1	$\overline{.068322}$	.345649					.009307	$\overline{.002351}$	$\overline{.001793}$	$\overline{.001235}$	$\overline{.000677}$	.003424		
6	$\overline{.056819}$	.287453					.012117	$\overline{.002391}$	$\overline{.001824}$	$\overline{.001256}$	$\overline{.000688}$	.003483		
			27.8986				.014255	.017251					.397695	
				25.9619			.015696		.017717					40
3					24.4503		.017949			.019080				
	1.01935					23.4456	.032969				.033607			
							.057317							
3	$\overline{.072523}$	.366898					.007503	$\overline{.001890}$	$\overline{.001441}$	$\overline{.000993}$	$\overline{.000544}$	.002753		
1	$\overline{.068322}$	.345649					.008475	$\overline{.002011}$	$\overline{.001534}$	$\overline{.001056}$	$\overline{.000579}$	.002929		
	$\overline{.056819}$	.287453					.010747	$\overline{.002121}$	$\overline{.001617}$	$\overline{.001114}$	$\overline{.000611}$	.003089		
			58.7740				.014754	.017896					.867152	
				54.6940			.013162		.014857					.71
					51.5095		.015680			.016668				
	1.01935					49.3929	.030646				.031239			
							.051790							
	$\overline{.072523}$	.366898					.006156	$\overline{.001550}$	$\overline{.001182}$	$\overline{.000814}$	$\overline{.000446}$	.002259		
	$\overline{.068322}$	.345649					.007231	$\overline{.001716}$	$\overline{.001309}$	$\overline{.000901}$	$\overline{.000494}$	.002499		
	$\overline{.056819}$	.287453					.009517	$\overline{.001878}$	$\overline{.001432}$	$\overline{.000986}$	$\overline{.000541}$	.002736		
			92.9439				.013341	.016182					1.239965	
				86.4919			.011015		.012433					.9
					81.4560		.013666			.014527				
	1.01935					78.1089	.028430				.028960			

	.068322	.345649					.009307	.002351	.001793	.001235	.000677	.003424		
	.056819	.287453					.012117	.002391	.001824	.001256	.000688	.003483		
			27.8986				.014255	.017251					.397695	
				25.9619			.015696		.017717					.40
					24.4503		.017949			.019080				
	1.01935					23.4456	.032969				.033607			
							.057317							
	.072523	.366898					.007503	.001890	.001441	.000993	.000544	.002753		
	.068322	.345649					.008475	.002011	.001534	.001056	.000579	.002929		
	.056819	.287453					.010747	.002121	.001617	.001114	.000611	.003089		
			58.7740				.014754	.017896					.867152	
				54.6940			.013162		.014857					.719
					51.5095		.015680			.016668				
	1.01935					49.3929	.030646				.031239			
							.051790							
	.072523	.366898					.006156	.001550	.001182	.000814	.000446	.002259		
	.068322	.345649					.007231	.001716	.001309	.000901	.000494	.002499		
	.056819	.287453					.009517	.001878	.001432	.000986	.000541	.002736		
			92.9439				.013341	.016182					1.23936	
				86.4919			.011015		.012433					.95
					81.4560		.013666			.014527				
	1.01935					78.1089	.028430				.028960			
							.046797							
3	.072523	.366898					.005045	.001271	.000969	.000667	.000366	.001851		
1	.068322	.345649					.006152	.001460	.001113	.000767	.000420	.002126		
6	.056819	.287453					.008411	.001660	.001266	.000872	.000478	.002418		
			130.7596				.011235	.013627					1.469084	
				121.6826			.009191		.010374					1.118
4	.006634	.033565		35.1248	17.5624		.017483	.000403	.000307	.000212	.000116	.000587		.614
3					114.5978		.011876			.012625				
4	.009069	.045880		17.5624	35.1248		.023161	.000728	.000556	.000383	.000210	.001063		.40
	1.01935					109.8888	.026323				.026832			

* N.B. THE COEFFICIENTS TABULATED
$(P_{4u})_o$, $(P_{8})_o$ & $(P_{8})_o$ OF MATRIX T_2' ARE NO
THE COEFFICIENTS TABULATED IN T_3 MATRIX
38 & 34 RESP. THIS WAS DONE TO KE
CONSISTENT & ELIMINATES ROUNDING ERRO

REPORT 7/0510/7
SHEET 5-61
R.N.L. MAY/55

T_2' MATRIX

$(P_{\varphi_3})_0$	$(P_{\varphi_2})_0$	STRESS FACTOR REF SHT 5-59	$(M_{\varphi_5})_0$	$(M_{\varphi_4})_0$	$(M_{\varphi_3})_0$	$(M_{\varphi_2})_0$	$(T)_0$	$(P_{\varphi_5})_0$ *	$(P_{\varphi_4})_0$ *	$(P_{\varphi_3})_0$ *	$(P_{\varphi_2})_0$ *	
		.009999	.002397									74
		.013679	.005945	.000953	$\overline{.001810}$	$\overline{.000992}$						75
		.017816	.003516	.002681	$\overline{.001365}$	$\overline{.002509}$						78
		.024941				$\overline{.001041}$						81
		.017610	.021360									73
		.018680		.021085								76
		.020500			.021792							79
		.035392				.036097						82
		.009141	$\overline{.002302}$	$\overline{.001756}$	$\overline{.001209}$	$\overline{.000663}$	.003354					84
		.009907	$\overline{.002351}$	$\overline{.001793}$	$\overline{.001235}$	$\overline{.000677}$	.003424					85
		.012117	$\overline{.002391}$	$\overline{.001824}$	$\overline{.001256}$	$\overline{.000688}$	.003483					86
		.014255	.017251					.397695				87
		.015696		.017717					.407498			88
1.4503		.017949			.019080					.438858		90
	23.4456	.032969				.033607					.772378	92
		.057317										93
		.007503	$\overline{.001830}$	$\overline{.001441}$	$\overline{.000993}$	$\overline{.000544}$	.002753					94
		.008475	$\overline{.002011}$	$\overline{.001534}$	$\overline{.001056}$	$\overline{.000579}$	.002929					95
		.010747	$\overline{.002121}$	$\overline{.001617}$	$\overline{.001114}$	$\overline{.000611}$	.003089					96
		.014754	.017896					.867152				97
		.013162		.014857					.719882			98
.5095		.015680			.016668					.807669		100
	49.3929	.030646				.031239					1.513695	102
		.051790										103
		.006156	$\overline{.001550}$	$\overline{.001182}$	$\overline{.000814}$	$\overline{.000446}$	.002259					104
		.007231	$\overline{.001716}$	$\overline{.001309}$	$\overline{.000901}$	$\overline{.000494}$	.002499					105
		.009517	$\overline{.001878}$	$\overline{.001432}$	$\overline{.000986}$	$\overline{.000541}$	.002736					106
		.013341	.016182					1.239965				107
4560		.011015		.012433					.952708			108
		.013666			.014527					1.113178		110
	78.1089	.023430				.028980					2.220636	112
		.046797										113

		.009907	.002351	$\overline{.001793}$	$\overline{.001235}$	$\overline{.000677}$	.003424					85
		.012117	$\overline{.002391}$	$\overline{.001824}$	$\overline{.001256}$	$\overline{.000688}$	.003483					86
		.014255	.017251					.397695				87
		.015696		.017717					.407498			88
4593		.017949			.019080					.438858		90
	23.4456	.032969				.033607					.772378	92
		.057317										93
		.007503	$\overline{.001890}$	.001441	$\overline{.000993}$	$\overline{.000544}$	.002753					94
		.008475	$\overline{.002011}$	.001534	$\overline{.001056}$	$\overline{.000579}$	.002929					95
		.010747	$\overline{.002121}$	$\overline{.001617}$	$\overline{.001114}$	$\overline{.000611}$	.003089					96
		.014754	.017896					.867152				97
		.013162		.014857					.719882			98
5095		.015680			.016668					.807669		100
	49.3929	.030646				.031239					1.513695	102
		.051790										103
		.006156	$\overline{.001550}$	$\overline{.001182}$	$\overline{.000814}$	$\overline{.000446}$	.002259					104
		.007231	$\overline{.001716}$	$\overline{.001309}$	$\overline{.000901}$	$\overline{.000494}$	.002499					105
		.009517	$\overline{.001878}$	$\overline{.001432}$	$\overline{.000986}$	$\overline{.000541}$	.002736					106
		.013341	.016182					1.239365				107
		.011015		.012433					.352708			108
4560		.013666			.014527					1.113178		110
	78.1089	.028430				.028980					2.220636	112
		.046797										113
		.005045	$\overline{.001271}$	$\overline{.000969}$	$\overline{.000667}$	$\overline{.000366}$	.001851					114
		.006152	$\overline{.001460}$	$\overline{.001113}$	$\overline{.000767}$	$\overline{.000420}$	.002126					115
		.008411	$\overline{.001660}$	$\overline{.001266}$	$\overline{.000872}$	$\overline{.000478}$	.002415					116
		.011235	.013627					1.469084				117
		.009191		.010374					1.118385			118
5624		.017483	.000403	.000307	.000212	.000116	$\overline{.000587}$		$\overline{.614087}$	$\overline{.307043}$		119
.5778		.011876			.012625					1.360963		120
.1248		.023161	.000729	.000556	.000383	.000210	$\overline{.001063}$		$\overline{.406763}$	$\overline{.813525}$		121
	107.8888	.026323				.026832					2.892603	122
												123

* N.B. THE COEFFICIENTS TABULATED IN COLUMNS $(P_{15})_0$, $(P_{14})_0$, $(P_{13})_0$ & $(P_{12})_0$ OF MATRIX T_2' ARE NOT THESE BUT ARE THE COEFFICIENTS TABULATED IN T_3 MATRIX COLUMNS 0: 43, 41, 38 & 34 RESP. THIS WAS DONE TO KEEP THE COEFFICIENTS CONSISTENT & ELIMINATES ROUNDING ERRORS.

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/7

SHEET NO. 5-62 *

AIRCRAFT		PREPARED BY	DATE
C105	Inner Wing – T_3	D.L. Turner	8 June '55
		CHECKED BY	DATE
		DLT	

Determination of the T_3 Matrix

The coefficients in the T_3 matrix are obtained by factoring the coefficients of the appropriate part of the T_{1a} matrix of report 7/0510/2 to account for the changes in panel thickness and spar & rib areas (ref. sht. 5-63), except for:

(a) The coefficients in rows $i = 119$ & 121, which are calculated on sheets 5-63 & 5-64.

(b) The coefficients in the columns $a = 15, 19$ & 73, which are calculated on sheets 5-65, 5-66 & 5-67, and tabulated in the summary on sheet 5-68.

A.V. ROE CANADA LIMITED
MALTON – ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/7	
SHEET NO. 5-63	
PREPARED BY: V.F. GARDENER	DATE: FEB. 11 '55
CHECKED BY: R.N. SHEARLY	DATE: MAY /55

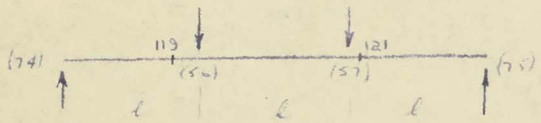
AIRCRAFT: C 105 — INNER WING.

STATICALLY DETERMINATE STRESS T_3 MATRIX

THE COEFFICIENTS OF THIS (T_3) MATRIX ARE OBTAINED BY FACTORIZING THE OLD VALUES BY THE RATIO OF THE OLD TO NEW AREAS.

THESE RATIOS ARE THE SAME AS THOSE USED FOR THE K_{ij} MATRIX, GIVEN IN p. 4-40

RIB AT FUSE. SIDE



$\ell = 52.6872$

UNIT LOAD AT (56)

$$\text{STRESS } 119 = \frac{2}{3} \times \frac{52.6872}{12.7162 \times 4.4982} = .61407$$

$$\text{"} \quad 121 = \frac{1}{3} \times \frac{52.6872}{9.5984 \times 4.4982} = .406767$$

UNIT LOAD AT (57)

$$\text{STRESS } 119 = \frac{1}{3} \times \frac{52.6872}{12.7162 \times 4.4982} \quad .3070349$$

$$\text{"} \quad 121 = \frac{2}{3} \times \frac{52.6872}{9.5984 \times 4.4982} \quad .81353496$$

DETERMINATION OF THE T_3 MATRIX (CONT.)

LOAD Q_{15} GIVES RISE TO AN EXTERNAL RIB MOMENT AT LOAD POINT (a=34) = 30 Q_{15} (NOSE UP POSITIVE)

THE RESULTING RIB MOMENTS = EXTERNAL RIB MOMENT × APPROPRIATE DISTRIBUTION FACTOR

AT M/S	$30\,Q_{15}$ × 0	=	0
AT F/C/S	$30\,Q_{15}$ × .366899	=	11.0070 Q_{15}
AT R/C/S	$30\,Q_{15}$ × .712547	=	21.3764 Q_{15}
AT R/S	$30\,Q_{15}$ × 1.000000	=	30.0000 Q_{15}

THE RESULTING RIB STRESSES = $M_R \times \frac{1}{h \cdot A_R}$

AT M/S	0 × .0099993	=	0
AT F/C/S	11.0070 Q_{15} × .0136791	=	.150566 Q_{15}
AT R/C/S	21.3764 Q_{15} × .0178165	=	.380853 Q_{15}
AT R/S	30.0000 Q_{15} × .0249406	=	.748218 Q_{15}

THE LOAD, Q_{15} PRODUCES

$$(T)_0 = 30\,Q_{15}$$

$$(P_{f_1}) = 1.000000\,Q_{15}$$

WHICH WHEN USED IN CONJUNCTION WITH THE T_2^1 MATRIX YIELD ELEMENTS IN THE T_3 MATRIX. SEE SUMMARY FOR RESULTS SUMMARY ON SHEET 5-68.

Load Q_{19} will give rise to an EXTERNAL RIB MOMENT AT LOAD POINT (a=43)

= − 52.2944 Q_{19} (SEE SHT. 5-67) (NOSE UP POSITIVE)

THE RESULTING RIB MOMENT = EXT. RIB MOMENT × APPROPRIATE DISTRIBUTION FACTOR

AT M/S	−52.2944 Q_{19}	× −1.000000.	= 52.2944 Q_{19}
AT F/C/S	"	× −.633101	= 33.1076 Q_{19}
AT R/C/S	"	× −.287453	= 15.0322 Q_{19}
AT R/S	"	× 0	= 0

AVRO AIRCRAFT LIMITED **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7/0510/7 SHEET NO 5-68 *	
AIRCRAFT: C-105	Inner Wing	PREPARED BY R.N. SHEARLY	DATE FEB 17/55
		CHECKED BY R.N. SHEARLY	DATE MAY /55

DETERMINATION OF T_3 MATRIX (CONT.)

SUMMARY OF TERMS DUE TO LOADS Q_{15} & Q_{19}

$i \downarrow$ \ $a \rightarrow$	15	19	73
74	—	.522907	.261454
75	.150566	.452882	.226441
78	.380853	.267821	.133910
81	.748218	—	—
84	.100620	-.175395	-.087698
85	.102720	-.179056	-.089528
86	.104490	-.182141	-.091071
87	—	.397695*	.198848*
92	.772978	—	—
94	.082590	-.143966	-.071983
95	.087870	-.153170	-.076585
96	.092670	-.161537	-.080769
97	—	.867152*	.433576*
102	1.513695	—	—
104	.067770	-.118133	-.059066
105	.074970	-.130684	-.065342
106	.082080	-.143077	-.071539
107	—	1.239965*	.619982*
112	2.220636*	—	—
114	.055530	-.096797	-.048398
115	.063780	-.111179	-.055589
116	.072540	-.126448	-.063224
117	—	1.469084*	.734542*
119	-.017610	.030697	.015348
121	-.031890	.055589	.027794
122	2.892603*	—	—

$Q_{73} = 0$
since redefinition of problem

* N.B. THE COEFFICIENTS TABULATED IN COLUMNS a=15, & 19 MARKED THUS * ARE NOT THE VALUES TABULATED IN MATRIX T_3 INSTEAD THE COEFFICIENTS OF COLUMNS a=34 & 43 respectively ARE TABULATED. THIS WAS DONE TO BE CONSISTENT & TO ELIMINATE ROUNDING ERRORS.

A. V. ROE CANADA LIMITED MALTON - ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7/0510/7 SHEET NO. 5-69*	
AIRCRAFT:		PREPARED BY	DATE
C-105	G_3	R.N. SHEARLY	FEB 10/55
		CHECKED BY	DATE
		R.N. SHEARLY	MAY /55

DETERMINATION OF THE G_3 MATRIX

THE G_3 MATRIX EXPRESSES THE INTERACTION GENERALIZED FORCES BETWEEN THE INNER WING AND THE CENTER SECTION IN TERMS OF THE UNIT LOADS APPLIED TO THE INNER WING. (THIS INCLUDES Q_a FOR $a = 15$ TO 19 inclusive) AT THIS PARTICULAR PARTITION (I/W TO C/S) THE SHEAR FORCES AND TORSIONAL MOMENT TRANSFER THRU THE FUSELAGE SIDE BEAM RIB TO THE REACTION POINTS ($a = 81$ & 82) AND HENCE ONLY THE SPAR BENDING MOMENTS ARE TRANSMITTED FROM THE INNER WING TO THE CENTER SECTION. THE ELEMENTS OF THE MATRIX ARE SIMPLY THE APPROPRIATE MOMENT ARMS.

TO OBTAIN THE CORRECT MOMENT ARMS

(1) ALL LOADS ON THE M/S ARE ASSUMED TO CAUSE B.M. AT $i = 117$ (M_5)
(2) ALL LOADS ON THE F/C/S ARE ASSUMED TO CAUSE B.M. AT $i = 118$ (M_6)
(3) ALL LOADS ON THE R/C/S ARE ASSUMED TO CAUSE B.M. AT $i = 120$ (M_7)
(4) ALL LOADS ON THE E/S ARE ASSUMED TO CAUSE B.M. AT $i = 122$ (M_8)

(5) LOAD AT $a = 19$ TRANSFERS TO LOAD POINT $a = 43$ AS SHEAR FORCE AND TORSIONAL MOMENT. THE SHEAR FORCE IS DEALT WITH THE SAME AS LOAD AT $a = 43$. THE TORSIONAL MOMENT HAS NO EFFECT ON THE INTERACTION GENERALIZED FORCES BETWEEN INNER WING AND CENTER SECTION.

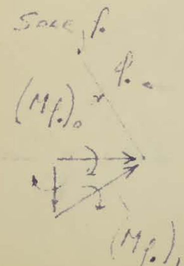
(6) LOADS AT $a = 15, 16, 17$ & 18 TRANSFER TO LOAD POINT $a = 39, 47, 51$ & 55 RESP. AS SHEAR FORCE AND TORSIONAL MOMENT. THE SHEAR FORCE IS DEALT WITH THE SAME AS LOAD AT $a = 39, 47, 51$ & 55 RESP., THE TORSIONAL MOMENT HAS NO EFFECT ON THE INTERACTION GENERALIZED FORCES BETWEEN INNER WING AND CENTER SECTION.

FORM 1319A

A. V. ROE CANADA LIMITED MALTON - ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7/0510/7 SHEET NO. 5-71	
AIRCRAFT:		PREPARED BY	DATE
C-105	H_2	R.N. SHERRELL	FEB 10/55
		CHECKED BY	DATE
		R.N. SHERRELL	MAY '55

DETERMINATION OF THE H_2 MATRIX

THE H_2 MATRIX EXPRESSES THE INTERACTION GENERALIZED FORCES ACTING BETWEEN THE INNER WING AND THE CENTER SECTION, SUBSEQUENTLY IN THIS PARAGRAPH CALLED THE INNER INTERACTION FORCES IN TERMS OF THE UNIT INTERACTION GENERALIZED FORCES ACTING BETWEEN THE OUTER WING AND THE INNER WING, SUBSEQUENTLY IN THIS PARAGRAPH CALLED THE OUTER INTERACTION FORCES. THE UNIT MOMENTS OF THE OUTER INTERACTION FORCES $\{(M_{f_5})_0, (M_{f_4})_0, (M_{f_3})_0 \; \& \; (M_{f_2})_0\}$ MUST BE RESOLVED INTO A SPAR MOMENT AND A TORSIONAL MOMENT. THE SPAR MOMENT FOR ANY PARTICULAR SPAR OF THE INNER INTERACTION FORCES WILL SIMPLY BE THE PART THAT RESOLVES INTO THAT SPAR. THE PART THAT RESOLVES INTO THE TORSIONAL MOMENT DOES NOT AFFECT THE INNER INTERACTION FORCES. (SEE PREVIOUS WORK ON q_3 MATRIX) THE UNIT TORSIONAL MOMENT OF THE OUTER INTERACTION FORCES $\{(T)_0\}$ DOES NOT AFFECT THE INNER INTERACTION FORCES. (SEE PREVIOUS WORK ON q_3 MATRIX) THE UNIT SHEAR FORCES OF THE OUTER INTERACTION FORCES $\{(P_{f_5})_0, (P_{f_4})_0, (P_{f_3})_0 \; \& \; (P_{f_2})_0\}$ CAUSE A SPAR MOMENT THE SAME AS LOADS AT α 43, 41, 38 & 34 RESP. (SEE PREVIOUS WORK ON q_3 MATRIX)



77.80

$$(M_{f.})_1 = \frac{1}{\cos f.}(M_{f.})_0$$

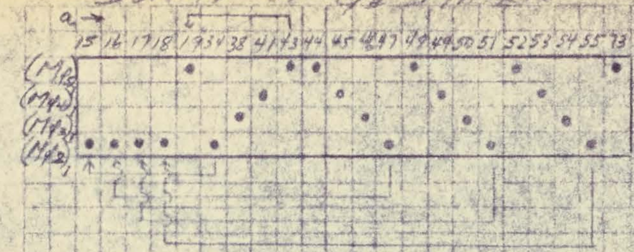
FORM 1318A

REPORT 7/0510/7 SHEET S-32

R.N. SHERRLY FEB 10/55

RN. 2U

DUMMY OF G_2 MATRIX



DUMMY OF H_2 MATRIX

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

AIRCRAFT:		REPORT NO. 7/0510/2	
C-105	Centre Section	SHEET NO. 5-79	
		PREPARED BY	DATE
		E.M. SHEARLY	FEB 11/55
		CHECKED BY	DATE
		BSO	

DETERMINATION OF T_3' MATRIX (CONT.)

HAVING OBTAINED THE BENDING MOMENTS IN THE SPARS & RIBS, M_{nia} & M_{yia}, AND THE TORSIONAL MOMENTS IN THE CELLS T_m, (SEE PRELIMINARY T_3' MATRIX) THE AXIAL STRESSES IN THE SPARS & RIBS AND THE SHEARING STRESS IN THE PANELS ARE FOUND BY MULTIPLYING BY THE APPROPRIATE STRESS FACTOR

$$\sigma_{nia} = \frac{1}{h_i A_{si}} M_{nia}$$

$$\sigma_{yia} = \frac{1}{h_i A_{ri}} M_{yia}$$

$$\tau_m = \frac{1}{4't} T_m$$

REF. REPORT 7/0510/2 SHEET 4.5.

FORM 1319A

... SECTION

... FACTORS

* THE VALUE IS ... SEE SHEET 5-110 FOR CORRECTIVE TREATMENT.

A. V...

AIRC...

WEIG...

C. G...

6	7	8	9	10	11	12	13	14	15	16	17	18	19	20
$\frac{1}{h H_2}$	$\frac{1}{\psi'}$			STRESS P	n	A_3	A_{1c}	$\frac{1}{\psi'}$	$\frac{1}{h A_3}$	$\frac{1}{h A_{1c}}$	$\frac{1}{\psi'}$			STRESS
...748251				136				·00053626			·00357506			151
...316127				137				·00060413			·00462753			152
				138				·00101507			·00676713			153
	* ·00418124			139										154
	·00520020			140	17·9644	7·3318			·00759235					155
	·00149640			141	17·9644		2·1600			·02577113				
				142	4·4880		2·1600			·03195492				
				143	14·4880	10·0801			·00684741					
				144										
				145	10·2672		2·1600			·04503146				
				146	10·2672	10·3354			·00942368					
				147										
				148	6·4011		2·1600			·07232553				
				149	6·4011	7·4231			·02104554					
				150										

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

REPORT NO. 7/05/0/7	
SHEET NO. 5-83	
PREPARED BY	DATE
R.N. SHEARLY	FEB 15/55
CHECKED BY	DATE
DD	

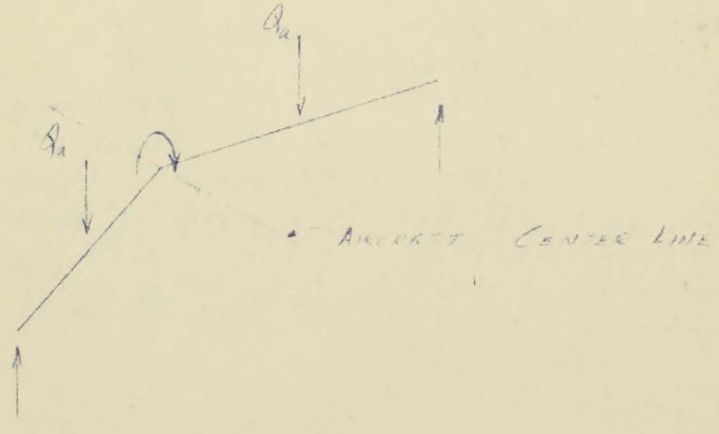
AIRCRAFT: C-105

Centre Section

DETERMINATION OF T_4 MATRIX

THIS MATRIX EXPRESSES THE STATICALLY DETERMINATE STRESSES OF THE CENTER SECTION WING IN TERMS OF THE LOADS ACTING ON THE CENTER SECTION WING.

TO UNDERSTAND THE INITIALLY ASSUMED LOAD PATH OF LOADS APPLIED BETWEEN THE FUSELAGE SIDE BEAMS, IT IS NECESSARY TO CONSIDER (A) ANY ONE SPAR AS A WHOLE FROM ONE SIDE BEAM TO THE OTHER SIDE BEAM AND (B) THE LOADS ARE APPLIED SYMMETRICALLY ABOUT THE AIRCRAFT CENTER LINE. THEN THE SPAR BENDS AS A SIMPLE BEAM WITH THE EXCEPTION THAT THE CENTER LINE RIB APPLIES A TORQUE TO THE SPAR TO MAINTAIN EQUILIBRIUM. (SEE FIGURE BELOW)

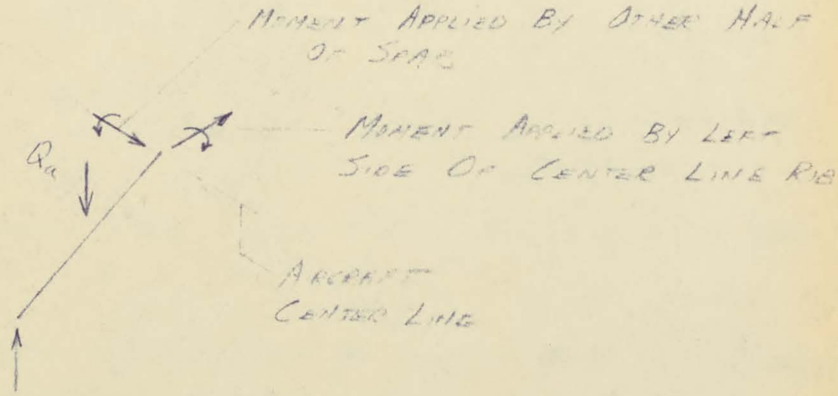


THE LOADS AND MOMENTS ACTING ON THE LEFT HALF OF THE SPAR ARE SHOWN IN FIGURE ON NEXT SHEET.

FORM 1319A

A. V. ROE CANADA LIMITED MALTON - ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT NO. 7/0510/7 SHEET NO. 5-84	
AIRCRAFT: C-105	Centre Section	PREPARED BY: R.N. SHERRY	DATE: FEB 15/55
		CHECKED BY: RO	DATE:

DETERMINATION OF T_4 MATRIX (CONT.)



THE SPAR ACTS ON THE LEFT HALF OF THE CENTER LINE RIB WITH A MOMENT CALLED SPAR KINK MOMENT (EQUAL & OPPOSITE TO THE ONE SHOWN IN THE ABOVE FIGURE) WHICH CAUSES RIB MOMENTS (SEE SHEET 5-89 & 90) AND TORSIONAL MOMENTS IN THE CELLS 136, 137 & 138 (SEE SHEET 5-91 & 92). THESE TORSIONAL MOMENTS CAUSE RIB MOMENTS IN THE FUSELAGE SIDE BEAM RIB AT STRESS POINTS i.e. 119 & 121 (SEE SHEET 5-93) AND THE FUSELAGE SIDE BEAM RIB CARRIES THE LOADS TO THE SUPPORT POINTS a - 81 & 82.

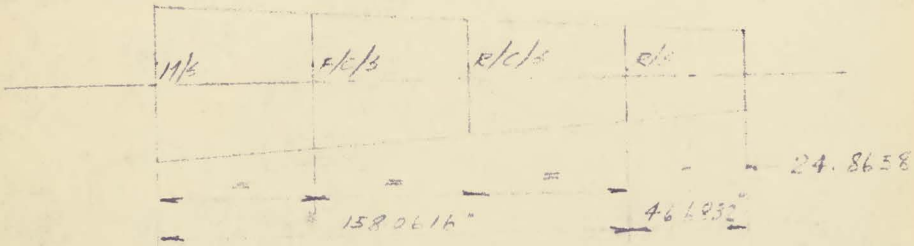
FORM 1310A

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

REPORT NO.	7/0510/7
SHEET NO.	5-85
PREPARED BY	DATE
R.N. SHERRY	FEB 14/55
CHECKED BY	DATE
D.J.	

AIRCRAFT: C-105 — Centre Section

DETERMINATION OF T_4 MATRIX (CONT)



SECTION THRU WING AT FUSELAGE SIDE RIB

$\frac{d}{9} = 17.5624$ $\frac{2d}{9} = 35.1248$ $\frac{e}{3} = 15.5611$ $\frac{2e}{3} = 31.1221$ $e = 46.6832$

i	h_i	A_{r_i}	$h_i A_{r_i}$	$\frac{1}{h_i A_{r_i}}$	$\frac{d}{9}\frac{1}{h_i A_{r_i}}$	$\frac{2d}{9}\frac{1}{h_i A_{r_i}}$	$\frac{e}{3}\frac{1}{h_i A_{r_i}}$	$\frac{2e}{3}\frac{1}{h_i A_{r_i}}$	$e\frac{1}{h_i A_{r_i}}$
119	12.7162	4.4982	57.2000	.0174825	.307035	.614069	.272047	.544092	.816139
121	9.5784	4.4992	43.1755	.0231613	.406768	.813536	.360415	.720828	1.081244
123	6.404	5.9655	25.3836	.0393955	.691880	1.383759	.613037	1.226071	1.839109

ABOVE TABLE NOT REQUIRED FOR DETERMINATION OF PRELIMINARY T_4 MATRIX, MAY BE USED TO CHECK T_4 MATRIX

A. V.

AIRCR.

WEIGH

C. G.

$\not{c}$ A/C $\mathfrak{L}$

LOAD POINT a

STRESS POINT i

$\frac{\delta x}{i}$ r f

— FUSELAGE SIDE BEAM RIB

PLAN VIEW TYPICAL SPAR

B. MOMENT AT LOAD POINT a DUE TO LOAD Q_a (APPLIED AT FUSE. SIDE BEAM RIB)

$= \frac{\delta x}{\cos \phi} Q_a$

B. STRESS $= \frac{1}{I/C} \frac{\delta x}{\cos \phi} Q$ = STRESS FACTOR × Q

7	8	9	10	11	12	13	14	15	16	17	18	19	20
62 [illegible]/2	62 SHT	64 4-67	65	66	67	68 →	STRESS FACTOR SEE SHT		LOAD A →	56	57	58	59
		34.7339 Q_{64}					.00882509		127				-.310312
			32.5591 Q_{65}				.00785121		129				
				30.6153 Q_{66}			.01053753		131				
29.3572 Q_{62}					32.3572 Q_{67}		.02318091		133				
	28.8409 Q_{63}					29.8409 Q_{68}	.07405213		135				
		27.7454 Q_{64}					.00759235		140				-265.373
			25.9162 Q_{65}				.00684741		142				
				30.3406 Q_{66}			.00742365		146				
29.3572 Q_{62}					32.7144 Q_{67}		.02104554		149				
	28.8409 Q_{63}					57.2[illegible] Q_{68}	.01623814		151				

B. STRESS AT STRESS POINT

THE [illegible] OF

PRELIMINARY TA[illegible]

A. V. ROE CANADA LIMITED MALTON - ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**		REPORT No. 7/0510/7 SHEET No. 5-90	
AIRCRAFT: C-105	Centre Section	PREPARED BY: R.N. SHEARLY	DATE: FEB 15/55
		CHECKED BY: DJ	DATE:

DETERMINATION OF T_4 MATRIX (CONT.)

DUE TO SPAR KINK MOMENT AT R/S

DUE TO Q_{62}	M/S	−5.6927	×	0	=	0
	F/C/S	"	×	.439284	=	−2.50071
	R/C/S	"	×	.774374	=	−4.40828
	R/S	"	×	.216583	=	−1.23294
DUE TO Q_{67}	M/S	−11.3854	×	0	=	0
	F/C/S	"	×	.439284	=	−5.00142
	R/C/S	"	×	.774374	=	−8.81656
	R/S	"	×	.216583	=	−2.46588

DUE TO SPAR KINK MOMENT AT FALSE SPAR

DUE TO Q_{63}	M/S	−1.5349	×	0	=	0
	F/C/S	"	×	.439284	=	−.67426
	R/C/S	"	×	.774374	=	−1.18859
	R/S	"	×	1.000000	=	−1.53490
	F/S	"	×	0	=	0
DUE TO Q_{64}	M/S	−3.0698	×	0	=	0
	F/C/S	"	×	.439284	=	−1.34851
	R/C/S	"	×	.774374	=	−2.37717
	R/S	"	×	1.000000	=	−3.06980
	F/S	"	×	0	=	0

...m Rib Moments

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

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

REPORT NO. - 7/25/7
SHEET - - 5-94
DATE - - [illegible]
PREPARED BY - [illegible]

	7	8	9	10	11	12	13	14	15	16	17	18	19
	65	66	67	68									
...348	3.194996	2.200645	1.206283	.325245			← Ref	Sht 5-93					
	$\overline{35.1249}$	$\overline{17.5624}$	—	15.5611			← From	Eq. On	Sht	5-85			
...348	$\overline{31.9298}$	$\overline{15.3619}$	1.206283	15.8863									
...821	3.247980	2.237139	1.226287	.330639			← Ref	Sht 5-93					
—	$\overline{17.5624}$	$\overline{35.1249}$	—	31.1221			← From	Eq. On	Sht	5-85			
...821	$\overline{14.3144}$	$\overline{32.8877}$	1.226287	31.4527									

REPORT 7/05/0/1 SHT 5-97

T_4 MATRIX

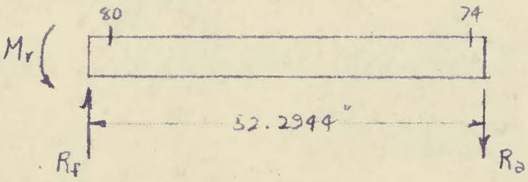
$i \downarrow$ \ $a \rightarrow$	56	57	58	59	60	61	62	63	64	65	66	67	68
119	$\overline{.614070}$	$\overline{.367035}$	.272047	.036622	$\overline{.586141}$	$\overline{.287739}$	.010544	.274890	.073233	$\overline{.558213}$	$\overline{.268563}$	.021083	.277732
121	$\overline{.406767}$	$\overline{.813535}$	.720827	.049320	$\overline{.369154}$	$\overline{.787627}$	.014200	.724656	.098639	$\overline{.331540}$	$\overline{.761721}$	.028403	.728484
123			1.839108					1.839108					1.839108
124				.036313	.027634	.019075	.010456	.002819	.072626	.055388	.038150	.020912	.005638
125				.035047	.026728	.018410	.010091	.002721	.070094	.053457	.036820	.020183	.005442
126				.033439	.025502	.017565	.009629	.002596	.066879	.051005	.035131	.019257	.005192
127			2.495266					2.495266					2.495266
128				$\overline{.310312}$					$\overline{.310312}$				
129					$\overline{.256483}$					$\overline{.256483}$			
130													
131						$\overline{.322609}$					$\overline{.322609}$		
132													
133							$\overline{.680527}$					$\overline{.680527}$	
134													
135								$\overline{2.135730}$					$\overline{2.135730}$
136				.031049	.023679	.016310	.008340	.002411	.062037	.047358	.032619	.017880	.004821
137				.030657	.023380	.016104	.008827	.002380	.061314	.046761	.032208	.017655	.004760
138				.030186	.023021	.015857	.008692	.002344	.060372	.046043	.031713	.017384	.004687
139													

Interaction Loads.

From these calculations

$$(1)\quad \begin{cases} M_s = .909548\, M_{102} - 1.76373\, M_{103} \\ M_r = .103171\, M_{102} + 1.98176\, M_{103} \end{cases}$$

Now, replace M_r by a force acting down at load point 19 and an equal force acting up at load point 43.



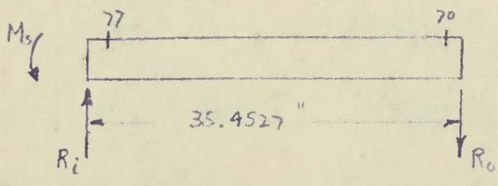
T/J rib.

$$R_f = R_a = \frac{M_r}{52.2944} = .0191225\, M_r$$

In making this substitution a stress is induced at stress point 80 of:

$$\sigma_{80} = \frac{M_r}{h_{80} A_{80}} = \frac{M_r}{6.5739 \times 8.2070} = .018534995\, M_r$$

Now, replace M_s by a force acting down at load point 19, and an equal force acting up at load point 42.



F/S.

$$R_i = R_o = \frac{M_s}{35.4527} = .028207\, M_s$$

AVRO AIRCRAFT LIMITED TECHNICAL DEPARTMENT (Aircraft)		REPORT NO. 7/0510/7 SHEET NO. 5-100	
AIRCRAFT: C105	Tia	PREPARED BY D.L. Turner	DATE 15 June '55
		CHECKED BY R. N. Healy	DATE June 15/55

Interaction Loads.

In making this substitution a stress is induced at stress point 77 of

$$\sigma_{77} = \frac{M_s}{h_{77}A_{77}} = \frac{M_s}{6.5739 \times 5.05} = .030122120\, M_s$$

Moments M_{102} & M_{103} have now been replaced by a set of forces.

$.0191225\, M_r + .028207\, M_s$ acting down at load point 19

$.0191225\, M_r$ acting up at load point 43

$.028207\, M_s$ acting up at load point 42

and stresses $\sigma_{80} = .018534995\, M_r$

$\sigma_{77} = .030122120\, M_s$

Substituting equations (1) of sht. 5-99 in these expressions gives loads:

$.001973\, M_{102} + .037896\, M_{103} + .025656\, M_{102} - .049750\, M_{103} =$

$= .027629\, M_{102} - .011854\, M_{103}$ acting down at load point 19

$.001973\, M_{102} + .037896\, M_{103}$ acting up at load point 43

$.025656\, M_{102} - .049750\, M_{103}$ acting up at load point 42.

FORM 1119A

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT C105	T_{1a}	REPORT NO. 7/0510/7	SHEET NO. 5-101
		PREPARED BY [signature]	DATE 15 June 55
		CHECKED BY [signature]	DATE

Interaction Loads

and stresses

$$\sigma_{80} = .001912274\,M_{102} + .036731912\,M_{103}$$

$$\sigma_{77} = .027397514\,M_{102} - .053127287\,M_{103}$$

Stresses in the remainder of the structure due to these substituted loads will be induced in the same manner as stresses due to Q_{13}, Q_{93} & Q_{42}.

Coefficients of the T_{1a} matrix for these loads may therefore be determined according to:

Coeff. of M_{102} = .027629 × Coeff. of Q_{13} − .001973 × Coeff. of Q_{93} − .025656 × Coeff. Q_{42}

Coeff. of M_{103} = −.011854 × Coeff. of Q_{13} − .037896 × Coeff. of Q_{93} + .049750 × Coeff. of Q_{42}

These coefficients are determined in the table on sht. {5-104, 5-105, 5-106}

Data pertinent to these calculations is given on shts 5-102 & 5-103

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO	7/0510/7
SHEET NO	5-103
PREPARED BY	R N SHERRLY
DATE	MAR. 15/55
CHECKED BY	D L [illegible]
DATE	2 May '55

AIRCRAFT: C-105

Tia

Interaction Loads

VALUE OF CONSTANTS (CONT)

1	$1/L$	$= .0116555$	
2	c/L	$= .609517$	
3	$1/\sin\beta$	$= 1.98444$	
4	$\sin\delta/\sin\beta$	$= .103171$	
5	$\cos\delta/\sin\beta$	$= 1.98176$	
6	$\sin(\beta-\delta)/\sin\beta$	$= .909548$	
7	$\cos(\beta-\delta)/\sin\beta$	$= 1.76373$	
8	c/a	$= 1.47505$	
9	b/a	$= .792112$	
10	$(\sin(\beta-\delta)/\sin\beta)\times c/a$	$= 1.341629$	6 × 8
11	$(\cos(\beta-\delta)/\sin\beta)\times c/a$	$= 2.601590$	7 × 8
12	$(\sin(\beta-\delta)/\sin\beta)\times b/a$	$= .720464$	6 × 9
13	$(\cos(\beta-\delta)/\sin\beta)\times b/a$	$= 1.39707$	7 × 9
14	$(\sin(\beta-\delta)/\sin\beta)\times \frac{b}{a}\cos\alpha$	$= .555924$	12 × cos α
15	$(\cos(\beta-\delta)/\sin\beta)\times \frac{b}{a}\cos\alpha$	$= 1.07801$	13 × cos α
16	$(\sin(\beta-\delta)/\sin\beta)\times \frac{b}{a}\sin\alpha$	$= .458273$	12 × sin α
17	$(\cos(\beta-\delta)/\sin\beta)\times \frac{b}{a}\sin\alpha$	$= .888648$	13 × sin α
18	$(\sin(\beta-\delta)/\sin\beta)\times\left(\frac{c}{a}-\frac{b}{a}\cos\alpha\right)$	$= .785705$	10 − 14
19	$(\cos(\beta-\delta)/\sin\beta)\times\left(\frac{c}{a}-\frac{b}{a}\cos\alpha\right)$	$= 1.52358$	11 − 15

FORM 2119A

A. V. R

Determination of Tie Elements for M_{102} & M_{103}

AIRCRAFT

WEIGHT

C. G. PO

i	1	2	3	4	5	6	7	8	9	10	
	Q_{19}	Q_{42}	Q_{43}	.02709(1)	$\overline{.025658}$(2)	$\overline{.01973}$(3)	Coeff. of M_{102} (4)+(5)+(6)		$\overline{.011859}$(1)	.049750(2)	$\overline{.03?}$
71		.64079			$\overline{.0164401}$		$\overline{.0164401}$			.0318793	
73		.38155			$\overline{.0097890}$		$\overline{.0097890}$			.0189821	
74	.52291	.11850		.0144475	$\overline{.0030402}$		.0114073		$\overline{.0061986}$	.0058954	
75	.45288	.29384		.0125121	$\overline{.0075388}$		.0049738		$\overline{.0053684}$	.0146185	
77	—	—	—				* .0273975				
78	.26782	.17380		.0073996	$\overline{.0044590}$		.0029406		$\overline{.0031747}$	.0086466	
80	—	—	—				* .0019123				
84	$\overline{.17539}$	$\overline{.11380}$		$\overline{.0048458}$	.0029197		$\overline{.0019261}$		.0020791	$\overline{.0056616}$	
85	$\overline{.17906}$	$\overline{.11619}$		$\overline{.0049472}$	.0029810		$\overline{.0019662}$		.0021226	$\overline{.0057804}$	
86	$\overline{.18214}$	$\overline{.11818}$		$\overline{.0050323}$	.0030320		$\overline{.0020003}$		.0021591	$\overline{.0058794}$	
87	.39770	.70656	.39770	.0109881	$\overline{.0181275}$	$\overline{.0007847}$	$\overline{.0079241}$		$\overline{.0047143}$	.0351514	$\overline{.015}$
94	$\overline{.14397}$	$\overline{.09341}$		$\overline{.0039777}$	.0023965		$\overline{.0015812}$		.0017066	$\overline{.0046471}$	
95	$\overline{.15317}$	$\overline{.09939}$		$\overline{.0042319}$	.0025499		$\overline{.0016820}$		.0018157	$\overline{.0049446}$	
96	$\overline{.16154}$	$\overline{.10483}$		$\overline{.0044632}$	.0026895		$\overline{.0017737}$		.0019149	$\overline{.0052153}$	
97	.86720	1.18657	.86720	.0239597	$\overline{.0304503}$	$\overline{.0017110}$	$\overline{.0082014}$		$\overline{.0102798}$	.0590968	$\overline{.03?}$
104	$\overline{.11817}$	$\overline{.07660}$		$\overline{.0032638}$	.0019653		$\overline{.0012975}$		.0014003	$\overline{.0038128}$	

FORM 1543

M_{102} & M_{103}

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

REPORT NO. - 7/[illegible]
SHEET - 5-104
DATE - 15 [illegible]

AIRCRAFT - C105
WEIGHT -
C. G. POSITION -
PREPARED BY - [illegible]

6	7	8	9	10	11	12	13	14	15	16	17	18	19
	Coeff. of M_{102}					Coeff of M_{103}							
$\overline{[illegible]973}$(3)	(4)+[illegible]+(6)		$\overline{.01185[illegible]}$(1)	.049750(2)	$\overline{.03758[illegible]}$(3)	(9)+(10)+(11)							
	$\overline{.0164401}$ $\overline{.0097890}$			.0318793 .0189821		.0318793 .0189821							
	.0114073		$\overline{.0061986}$	.0058954		$\overline{.0003032}$							
	.0049738		$\overline{.0053684}$	.0146185		.0092501							
	* .0273975					* $\overline{.0531273}$							
	.0029406		$\overline{.0031747}$	.0086466		.0054719							
	* .0019123					* .0367319		* ref sht. 5-101					
	$\overline{.0019261}$		.0020791	$\overline{.0056616}$		$\overline{.0035825}$							
	$\overline{.0019662}$		.0021226	$\overline{.0057804}$		$\overline{.0036578}$		Note that these elements are					
	$\overline{.0020003}$		.0021591	$\overline{.0058794}$		$\overline{.0037203}$	to be multiplied by 100 since unit						
$\overline{.0007847}$	$\overline{.0079241}$		$\overline{.0047143}$	.0351514	$\overline{.0150712}$	.0153659	moment is taken as 100 in.lb						
	$\overline{.0015812}$		.0017066	$\overline{.0046471}$		$\overline{.0029905}$							
	$\overline{.0016820}$		.0018157	$\overline{.0049946}$		$\overline{.0031283}$							
	$\overline{.0017737}$		.0019149	$\overline{.0052153}$		$\overline{.0033004}$							
$\overline{.0017110}$	$\overline{.0082014}$		$\overline{.0102798}$	.0590968	$\overline{.0328634}$	.0159036							
	$\overline{.0012975}$		.0014003	$\overline{.0038128}$		$\overline{.0024125}$							

Determination of Tie Elements for M_{102} & M_{103}

A. V. R

AIRCRAF

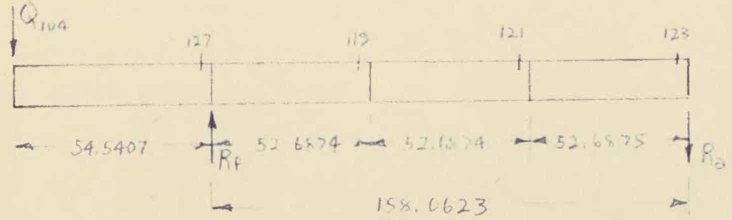
WEIGHT

C. G. PO

	1	2	3	4	5	6	7	8	9	10	
	Coefficient of Q_{101}	Q_{102}	Q_{A}	.027629[illegible]	$\overline{.025651}$(2)	$\overline{.001933}$(3)	Coeff of M_{102} (4)+(5)+(6)		$\overline{.0118\ldots4}$[illegible]	.049756[illegible]	$\overline{.03}$
105	$\overline{.13068}$	$\overline{.08480}$		$\overline{.0036106}$	.0021756		$\overline{.0014352}$		.0015421	$\overline{.0043188}$	
106	$\overline{.14308}$	$\overline{.09284}$		$\overline{.0039532}$	.0023819		$\overline{.0015713}$		.0016961	$\overline{.0046188}$	
107	1.24000	1.52905	1.24000	.0342600	$\overline{.0392293}$	$\overline{.0024465}$	$\overline{.0024158}$		$\overline{.0146990}$	.0760702	$\overline{.04}$
114	$\overline{.09580}$	$\overline{.06281}$		$\overline{.0026295}$	.0016114		$\overline{.0010631}$		.0011425	$\overline{.0031248}$	
115	$\overline{.11118}$	$\overline{.07215}$		$\overline{.0030718}$	.0018511		$\overline{.0012207}$		.0013179	$\overline{.0035895}$	
116	$\overline{.12695}$	$\overline{.08205}$		$\overline{.0034937}$	.0021051		$\overline{.0013886}$		.0014983	$\overline{.0040820}$	
117	1.46909	1.71251	1.46909	.0405825	$\overline{.0439312}$	$\overline{.0028985}$	$\overline{.0062952}$		$\overline{.0174146}$	.0851374	$\overline{.05}$
118											
119	$\overline{.10634}$	$\overline{.13982}$	$\overline{.13704}$	$\overline{.0029381}$	.0035872	.0002704	.0003195		.0012606	$\overline{.0069560}$	.00
121	$\overline{.12904}$	$\overline{.17918}$	$\overline{.18463}$	$\overline{.0035652}$	.0045970	.0003643	.0013961		.0015296	$\overline{.0089142}$	.00
124	$\overline{.13599}$	$\overline{.15852}$	$\overline{.13599}$	$\overline{.0037573}$	.0040670	.0002683	.0005780		.0016120	$\overline{.0078864}$	.00
125	$\overline{.13115}$	$\overline{.15288}$	$\overline{.13115}$	$\overline{.0036235}$	.0039223	.0002588	.0005576		.0015546	$\overline{.0076058}$	.00
126	$\overline{.12514}$	$\overline{.14587}$	$\overline{.12514}$	$\overline{.0034575}$	.0037424	.0002469	.0005318		.0014839	$\overline{.0072570}$	.00
128	1.16154	1.35400	1.16154	.0320922	$\overline{.0347382}$	$\overline{.0022917}$	$\overline{.0049327}$		$\overline{.0137689}$	.0673615	$\overline{.0}$
136	$\overline{.11625}$	$\overline{.13557}$	$\overline{.11625}$	$\overline{.0032119}$	.0034766	.0002294	.0004941		.0013780	$\overline{.0067416}$	.00

Interaction Loads (Q_{104})

Since this load is reacted at load points 81 & 82 it will cause stresses in the fuselage side rib only.



$$R_a = \frac{54.5407}{158.0623} Q_{104} = .345058\, Q_{104}$$

$$R_f = 1.345058\, Q_{104}$$

$$M_{127} = 54.5407\, Q_{104}$$

$$M_{119} = (54.5407 + 52.6874) Q_{104} - 52.6874\, R_f$$

$$= 107.2281\, Q_{104} - 52.6874 \times 1.345058\, Q_{104} = 36.3605\, Q_{104}$$

$$M_{121} = 52.6875 \times .345058\, Q_{104} = 18.1802\, Q_{104}$$

$$M_{123} = 0$$

now $\sigma = \frac{M}{hA}$

$$\sigma_{127} = \frac{54.5407}{15.0010 \times 4.4982} Q_{104} = .808280\, Q_{104}$$

$$\sigma_{119} = \frac{36.3605}{12.7162 \times 4.4982} Q_{104} = .635673\, Q_{104}$$

$$\sigma_{121} = \frac{18.1802}{9.5984 \times 4.4982} Q_{104} = .421077\, Q_{104}$$

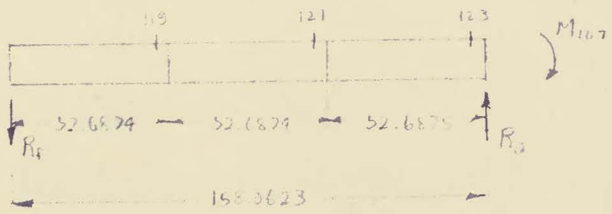
$$\sigma_{123} = 0$$

AVRO AIRCRAFT LIMITED TECHNICAL DEPARTMENT (Aircraft)		REPORT No 7/0510/7 SHEET No 5-108	
		PREPARED BY	DATE
AIRCRAFT: C105	T_{12}	D.L. Turner	14 June '55
		CHECKED BY	DATE
		[illegible]	[illegible]

Interaction Loads (Q_{105}, Q_{106})

The elements of loads Q_{105} & Q_{106} in the T_{12} matrix will be identical to those of loads Q_{56} & Q_{57} respectively.

M_{107}.
This load will cause stresses in the fuselage side rib only.



$$R_a = R_f = \frac{M_{107}}{158.0623} = .0063266\ M_{107}$$

$$M_{123} = M_{107}$$

$$M_{121} = M_{107} - 52.6875 \times .0063266\ M_{107} = .666667\ M_{107}$$

$$M_{119} = .333333\ M_{107}$$

now $\sigma = \dfrac{M}{hA}$

$$\sigma_{123} = \frac{1}{6.411 \times 3.9655}\ M_{107} = .03933557\ M_{107}$$

$$\sigma_{121} = \frac{.666667}{9.5984 \times 4.4982}\ M_{107} = .01544085\ M_{107}$$

$$\sigma_{119} = \frac{.333333}{12.7162 \times 4.4982}\ M_{107} = .00582750\ M_{107}$$

Note: The coefficients here are to be multiplied by 100 for entry in the matrix, as unit M. is taken as 100 in. lb.

FORM 1319A

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT C105	Tia	REPORT NO. 7/0510/7	
		SHEET NO. 5-109	
		PREPARED BY	DATE
		D. F. Turner	14 June 55
		CHECKED BY	DATE
		R. M. [illegible]	June 15/55

Interaction Loads. (Q_{108})

This load consists of a pair of equal loads, one acting down at the fuselage ℄ (identified on fig. 8), the other acting up at the fuselage side at load point 104.

Hence the elements of the Tia matrix for Q_{108} will consist of the sum of two sets, one set calculated for a down load at the fuselage ℄, the other calculated for an up load at the fuselage side.

Consider the fuselage side component.

The stresses due to this component will be equal in value, but opposite in sign to those due to Q_{104}.

Hence: $\sigma_{127} = -.808280 \; Q_{108}$

$\sigma_{119} = -.635673 \; Q_{108}$

$\sigma_{121} = -.421077 \; Q_{108}$

Consider the fuselage ℄ component.

This portion of Q_{108} is transferred to load point 64, giving a shear at load point 64 and a bending moment in the fuselage ℄ rib.

The moment on the rib at load point 64 will be:

$T_1 = 15.0000 \; Q_{108}$ (tension in top)

The shear is now transferred through the wing structure to reaction points 81 & 82, giving rise to stresses in the same way as does Q_{64}. The coefficients of Q_{108} due to this shear will therefore be the same as those of Q_{64}. These coefficients are tabulated on sht. 5-112.

Interaction Loads

The moment T_1 applied to the fuselage ℄ rib at the main spar (tension in top) will generate a system of stresses in the centre section wing in the same way that the moment $M_{\phi_{S1}} \sin \phi_S$ did in the T_3' calculations.
Referring to these calculations (sht 5-74 & 5-82) the following date are obtained.

Due to a moment, applied to the fuselage ℄ rib at the main spar, of:

$$M = M_{\phi_{S1}} \sin \phi_S = .565953\, M_{\phi_{S1}} \quad \text{(compr. in top. ref. sht. 5-79)}$$

the following stresses are generated in the wing structure in transferring this moment to the main reaction points (81 & 82)

i	Coeff. of $M_{\phi_{S1}}$
119	-.001048
121	-.001412
124	-.001040
125	-.001003
126	-.000957
136	-.000889
137	-.000878
138	-.000864
141	
142	-.010141
145	-.005758

ref. sht. 5-82

Note: For i = 128 & 140, stresses due to $M_{\phi_{S1}}$ in the table on sht 5-82 are due to this moment being transferred inboard along the main spar before it is resolved at the ℄. Since the moment due to Q_{108} is applied at the ℄ rib, these stresses will be zero.
For i = 141, stress due to $M_{\phi_{S1}}$ is factored down in the table of sht 5-82 according to the work of sect 7. In the case under consideration, the full value of the stress is to be used, (ref sect. 7) hence it will be recalculated.

For the case in question:

$$M = 15.0000\, Q_{108} \quad \text{(tension in top of rib)}$$

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/7	SHEET NO. 5-111	
AIRCRAFT: C105	Tia	
PREPARED BY: D. Dunne	DATE: 14 June '55	
CHECKED BY: [illegible]	DATE: June 15/55	

Interaction Loads.

Hence the above coefficients must be factored according to:

$$\text{Coeff. of } Q_{108} = -\frac{15.0000}{.565953} \times \text{Coeff. of } M_{\phi_{57}}$$

$$= -26.5039665 \times \text{Coeff. of } M_{\phi_{57}}$$

These coefficients are computed in the table below:

(1) i	(2) Coeff. of $M_{\phi_{57}}$	(3) Coeff. of Q_{108} $-26.5039665 \times (2)$
119	−.001048	.027776
121	−.001412	.037424
124	−.001040	.027564
125	−.001003	.026583
126	−.000957	.025364
136	−.000889	.023562
137	−.000878	.023270
138	−.000864	.022899
* 141		.386567
142	−.010141	.268777
145	−.005758	.152610

$$* \quad \sigma_{141} = \frac{15.0000\, Q_{108}}{h_{141}\, A_{141}} = .386567\, Q_{108} \qquad \frac{1}{h_{141} A_{141}} \text{ derives from sht. 5-82}$$

These coefficients combine with others tabulated on sht. 5-112

FORM 1104

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO 7/0510/7

SHEET NO 5-112

AIRCRAFT: C105

Tia

PREPARED BY D L Dunner DATE 14 June 55

CHECKED BY K McLeod DATE June 15/55

Interaction Loads

Summary of stresses in wing due to load Q_{108}

	Due to fus. comp	Due to fus. & comp. Due to shear at 64	Due to moment at 64	Total Coeff.
119	−.635673	.073239	.027776	−.534658
121	−.421077	.098639	.037424	−.285014
124		.072626	.027564	.100190
125		.070094	.026583	.096677
126		.066879	.025364	.092243
127	−.808280			−.808280
128		−.310312		−.310312
136		.062097	.023562	.085659
137		.061314	.023270	.084584
138		.060372	.022899	.083271
140		−.530446		−.530446
141		.283723	.386567	.670290
142		.708478	.268777	.977255
145		.402280	.152610	.554890

AVRO AIRCRAFT LIMITED TECHNICAL DEPARTMENT (Aircraft)		REPORT NO 7/0510/7	
		SHEET NO 5-113	
AIRCRAFT: C105	Tia	PREPARED BY	DATE
		[signature]	14 June 55
		CHECKED BY	DATE
		[signature]	June /55

Interaction Loads (Q_{105})

This load consists of a pair of equal loads, one acting down at the fuselage ℄ (identified on fig. 8), the other acting up at the fuselage side at load point 81.

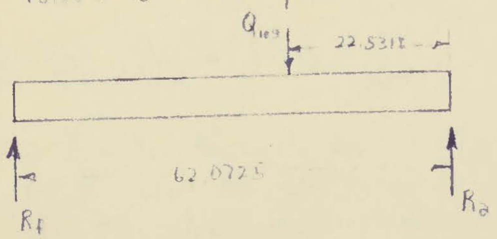
Hence the elements of the Tia matrix for Q_{109} will consist of the sum of two sets, one set calculated for a down load at the fuselage ℄, the other calculated for an up load at the fuselage side.

Consider the fuselage side component.

This component will cause no stress, as it is applied at main reaction point 81.

Consider the fuselage ℄ component.

This load is split (for the purpose of calculating wing stresses) into loads at load points 64 & 65.



$$R_f = \frac{22.5318}{62.0725} Q_{109} = .362992 \; Q_{109} \qquad R_a = .637008 \; Q_{109}$$

Stresses due to R_f will be identical to those due to Q_{64}, stresses due to R_a will be identical to those due to Q_{65}.

Hence the Tia elements for the fuselage ℄ component of Q_{109} will be:

Coeff. of Q_{109} = .362992 × Coeff of Q_{64} + .637008 × Coeff. of Q_{65}

These are calculated in the table on sht. 5-118

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C105	T_{ia}	REPORT NO 7/0510/7	
		SHEET NO 5-114	
		PREPARED BY: D L Turner	DATE: 14 June '55
		CHECKED BY: [illegible]	DATE: June 1?/55

Interaction Loads (Q_{110})

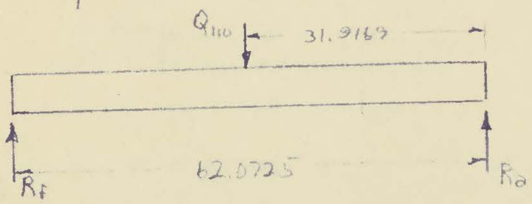
This is the same type of load as Q_{109}, and is handled in the same way.

Consider the fuselage side component.

Stresses due to this component will be equal in value but opposite in sign to those due to Q_{56}. These are tabulated in the table on sht 5-118

Consider the fuselage ℄ component

This load is split into loads at load points Q_{65} & Q_{66}.



$$R_f = \frac{31.9162}{62.0725} Q_{110} = .514187\, Q_{110} \qquad R_a = .485813\, Q_{110}$$

Stresses due to R_f will be identical to those due to Q_{65}, stresses due to R_a will be identical to those due to Q_{66}.

Hence the T_{ia} elements for the fuselage ℄ component of Q_{110} will be:

$$\text{Coeff. of } Q_{110} = .519187 \times \text{Coeff. of } Q_{65} + .485813 \times \text{Coeff. of } Q_{66}$$

These are calculated in the table on sht. 5-118 & combined with the fuselage side component values.

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C105	T_{1a}	REPORT NO 7/0510/7	SHEET NO 5-115
		PREPARED BY D L Dwyer	DATE 14 June 55
		CHECKED BY [illegible]	DATE June 15/55

<u>Interaction Loads</u> (Q_{111})

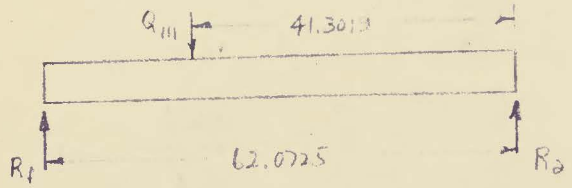
This is the same type of load as Q_{110} and is handled in the same way.

Consider the fuselage side component

Stresses due to this component will be equal in value but opposite in sign to those due to Q_{57}. These are tabulated in the table on sht. 5-118

Consider the fuselage ℄ component.

This load is split into loads at load points Q_{66} & Q_{67}



$$R_f = \frac{41.3013}{62.0725} Q_{111} = .665382\ Q_{111} \qquad R_2 = .334618\ Q_{111}$$

Stresses due to R_f will be identical to those due to Q_{66}, stresses due to R_2 will be identical to those due to Q_{67}.

Hence the T_{1a} elements for the fuselage ℄ component of Q_{111} will be:

$$\text{Coeff. of } Q_{111} = .665382 \times \text{Coeff. of } Q_{66} + .334618 \times \text{Coeff. of } Q_{67}$$

These are calculated in the table on sht. 5-118 & combined with the fuselage side component values.

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C105	Tia	REPORT NO 7/0310/7	SHEET NO 5-116
		PREPARED BY D. Turner	DATE 15 June 55
		CHECKED BY R.N. Sheary	DATE June 15/55

Interaction Loads (Q_{112})

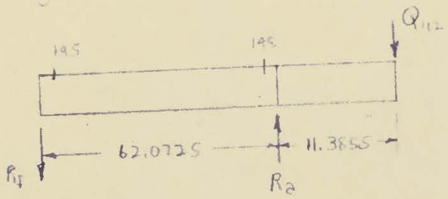
This load consists of a pair of equal loads, one acting down at the fuselage ℄ (identified on fig. 8), the other acting up at the fuselage side at load point 82.

Hence the elements of the Tia matrix for Q_{112} will consist of the sum of two sets, one set calculated for a down load at the fuselage ℄, the other calculated for an up load at the fuselage side.

The fuselage side component will cause no stress in the wing, as it is applied at main reaction point 82.

Consider the fuselage ℄ component.

This portion of the load is split (for the purpose of calculating wing stresses) into loads at load points 66 & 67.



$$R_f = \frac{11.3855}{62.0725} Q_{112} = .183423\, Q_{112} \qquad R_a = 1.183423\, Q_{112}$$

This splitting results in a stress at stress point 148, which is neglected as it is considered that load 112 will transfer from load point 112 to load points 66 & 67 via the fin rather than the rib (remaining wing stresses being unchanged in value); a down load at load point 67 of $1.183423\, Q_{112}$; and an up load of $.183423\, Q_{112}$ at load point 66.

Hence the elements of the Tia matrix for this load, in addition to σ_{148} as defined above, will be formed according to:

$$\text{Coeff of } Q_{112} = 1.183423 \times \text{Coeff. of } Q_{67} - .183423 \times \text{Coeff. of } Q_{66}$$

These computations are carried out on sht. 5-118

FORM 1319A

Q_{111} & Q_{112}

A. V

AIRC

WEI

C. G

7	8	9	10	11	12	13	14	15	16	17	18	19	20
Coeff of $Q_{[illegible]}$	Due to Fus. side comp.	.514187 x(2)	.485813 x(?)	Total Coeff. of Q_{110} (8)+(9)+(10)		Coeff of $Q_{[illegible]}$	Due to Fus. side comp.	.665382 x(7)	.334618 x(13)	Total Coeff. of Q_{111} (14)+(15)+(16)		1.183423 x(13)	-.183423 x(7)
$\overline{.26856}$	.61407	$\overline{.2[illegible]}$	$\overline{.13097}$	.19658		.02109	.30703	$\overline{.17869}$	.00706	.13590		.02496	.04[illegible]
$\overline{.26172}$	.40677	$\overline{.1[illegible]}$	$\overline{.37005}$	$\overline{.13335}$		.02590	.81353	$\overline{.50683}$	.00[illegible]	.31620		.02361	.13722
.03815		.02545	.01853	.04701		.02091		.02538	.00700	.03238		.02474	$\overline{.00700}$
.03682		.02749	.01782	.04537		.02018		.02952	.00675	.03125		.02358	$\overline{.00675}$
.03513		.02623	.01707	.04330		.01926		.02337	.00649	.02981		.02279	$\overline{.00649}$
		$\overline{.13188}$		$\overline{.13188}$									
$\overline{.32261}$			$\overline{.15673}$	$\overline{.15673}$				$\overline{.21466}$		$\overline{.21466}$			.05917
						$\overline{.68053}$			$\overline{.22772}$	$\overline{.22772}$		$\overline{.80535}$	
.03262		.02435	.01585	.04020		.01788		.02170	.00598	.02768		.02116	$\overline{.00598}$
.03221		.02499	.01565	.03[illegible]		.01765		.02193	.00571	.02754		.02089	$\overline{.00571}$
.03171		.02367	.01541	.03708		.01738		.02110	.00582	.02692		.02057	$\overline{.00582}$
$\overline{.29156}$		.23008	$\overline{.19164}$	$\overline{.11156}$		$\overline{.15982}$		$\overline{.19900}$	$\overline{.05348}$	$\overline{.24798}$		$\overline{.18213}$	.05348
		$\overline{.22831}$		$\overline{.22591}$									
$\overline{.35697}$		.15775	$\overline{.12484}$	.03291		$\overline{.39255}$		$\overline{.17028}$	.13303	$\overline{.30401}$		$\overline{.40047}$	.04713
$\overline{.57702}$			$\overline{.28032}$	$\overline{.28032}$				$\overline{.38394}$		$\overline{.38394}$			.10584
						$\overline{.17835}$			$\overline{.05968}$	$\overline{.05968}$		$\overline{.21106}$	
						$\overline{1.23568}$			$\overline{.41348}$	$\overline{.4134[illegible]}$		$\overline{1.46233}$	

AVRO AIRCRAFT LIMITED

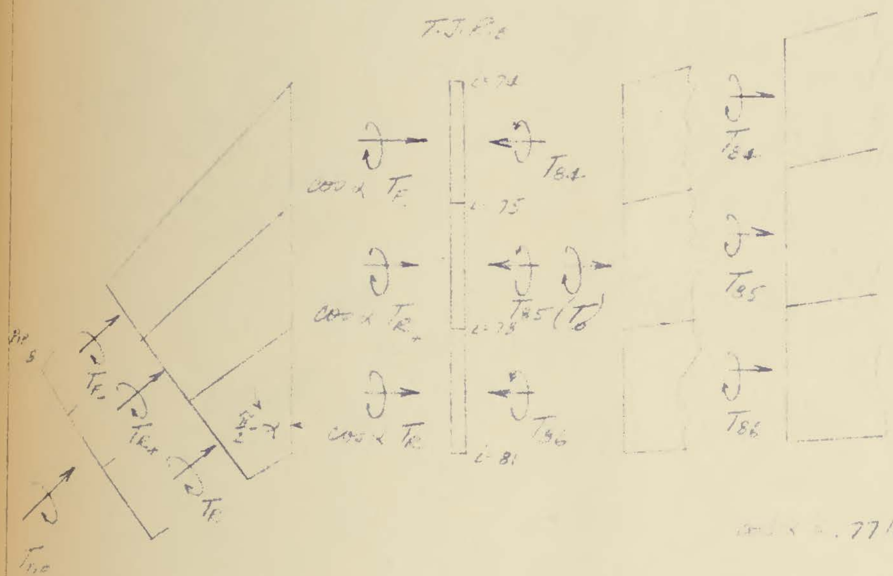
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT		REPORT NO 7/0510/7	
C-105	T_{LL}	SHEET NO 5-119	
		PREPARED BY	DATE
		P.M. SHEARLY	JULY 5/55
		CHECKED BY	DATE

DUE TO AN ERROR ON SHT 5-80 WHICH GIVES $\frac{1}{.1504'}$ FOR STRESS POINT $L = 124$ AS .00418124 INSTEAD OF .00413880 ALL ELEMENTS IN THE T_{LA} MATRIX FOR $L = 124$ WERE CALCULATED IN ERROR. (DIRECTLY FOR UNIT LOADS & INDIRECTLY FOR INTER ACTION LOADS)

THE SIMPLEST WAY TO CORRECT THIS IS TO FACTOR THESE ELEMENTS BY THE FACTOR .00413880/.00418124 = 1.00420 TO OBTAIN THE CORRECT ELEMENTS. THE ONLY PLACE THESE CORRECT ELEMENTS APPEAR IS ON THE FINAL MATRIX.

THE CHECK AT THE T.J. RIB (SHTS 7-44 g - 7-44 l) BROUGHT TO LIGHT AN OMISSION IN COMPUTING THE STATICALLY DETERMINATE STRESSES AT STRESS POINTS L-75 & 78. THE CORRECTION IS COMPUTED BELOW & APPLIED DIRECTLY TO THE T_{EC} HAUSER.



$\cos\alpha = .7763$ SHT 5-54

$T_E = .279274\, T_{TIP}$ $\quad T_{R+} = .342002\, T_{TIP}$ $\quad T_R = .378724\, T_{TIP}$ SHT 5-24

$\cos\alpha\, T_E = .215496\, T_{TIP}$ $\quad \cos\alpha\, T_{R+} = .265499\, T_{TIP}$ $\quad \cos\alpha\, T_R = .292235\, T_{TIP}$

$T_{84} = .366878\, (T)_o$ $\quad T_{85} = .345649\, (T)_o$ $\quad T_{86} = .287473\, (T)_o$ SHT 5-54

$$(T)_o = .7763\, T_{TIP} \qquad \text{SHT 5-44}$$

$T_{84} = .283110\, T_{TIP}$ $\quad T_{85} = .266713\, T_{TIP}$ $\quad T_{86} = .221807\, T_{TIP}$

THE TORQUE T_{TIP} APPLIED TO THE OUTER WING AT RIB 13 IS DISTRIBUTED INTO THE THREE REAR CELLS OF THE OUTER WING IN THE RATIO OF THE CELL AREAS AT RIB 13. THESE PORTIONS OF THE TORQUE REMAIN CONSTANT TO THE

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C105	Wing Analysis	REPORT NO 7/0510/7	SHEET NO 6-1
		PREPARED BY [signature]	DATE 17 June 55
		CHECKED BY	DATE

The calculations of this report determined the elements of the T_{ia} and K_{ip} matrices according to the geometry of figures 3-8 inclusive. Since these calculations were made, areas and panel thicknesses in some regions have again been changed (see figure 10 for revised data.) The T_{ia} & K_{ip} matrices (now termed preliminary matrices) have therefore been factored to produce the final matrices. The factors used are determined on sht. 6-2.

Changes in elements of the C_{ik} matrix due to these geometry changes have been calculated, and are tabulated, at the end of section 3 of this report.

Factors To Convert Elements Of Preliminary T_{ia} & K_{ip} Matrices Into Elements Of Final T_{ia} & K_{ip} Matrices (For Center Section)

A. V.

AIRCRA

WEIGHT

C. G. F

Stress Point i	1 Preliminary Area or Thickness	2 New Area or Thickness June 9/55	3 Factor ①/②	4	5	6	7	8 Stress Point i	9 Preliminary Area or Thickness	10 New Area or Thickness
117	5.9333	5.5293	1.073065					132	4.3200	4.4640
118	8.5560	8.5560	1					133	6.7313	6.2666
119	4.4982	4.5702	.984246					134	4.3200	0
120	8.7727	8.7727	1					136	.150	.155
121	4.4982	4.5702	.984246					137	.150	.155
122	5.9348	5.9348	1					138	.150	.155
123	3.9655	3.0000	1.321833					140	7.3318	6.1037
124	.150	.155	.967742					141	2.1600	1.0000
125	.150	.155	.967742					142	2.1600	2.2320
126	.150	.155	.967742					143	10.0801	10.5915
127	4.4932	2.2520	1.997424					145	2.1600	4.0930
128	6.8278	5.7079	1.196133					146	10.3354	11.4098
129	9.3181	9.7908	.951720					147	—	—
130	4.3200	4.4640	.967742					148	2.1600	2.7400
131	9.5541	10.5472	.905842					149	7.4231	7.3251

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No 7/0510/7	SHEET No 7-1	
AIRCRAFT: C105	Miscellaneous	PREPARED BY [signature] DATE 5 May '55
		CHECKED BY DATE

Introduction

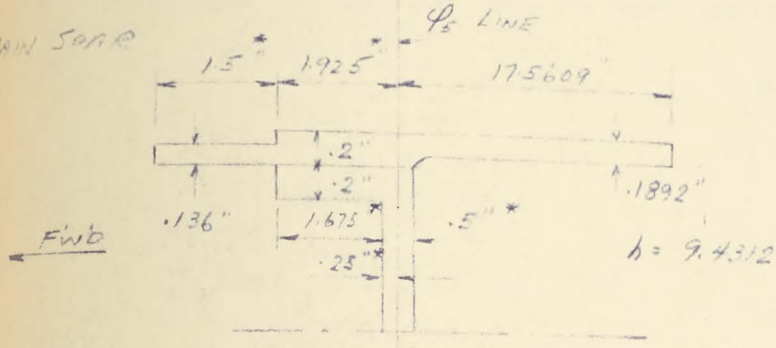
In this section of the report, individual calculations and derivations are made which are required for work in other sections of the report, but which, in themselves, do not belong in these other sections.

These consist of:

(1) Calculations concerning boundary conditions for statically determinate stresses where the spars and/or ribs change direction (e.g. wing transport joint, fuselage ℄)
(2) The development of expressions for the stress energy in a single wing tip bay. (where a statically determinate stress system is assumed).
(3) Formal checking techniques & calculations.

SPAR AREAS AT THE T.J.

MAIN SPAR



* DIMENSIONS MARKED THUS MEASURED PERP. TO SPAR.

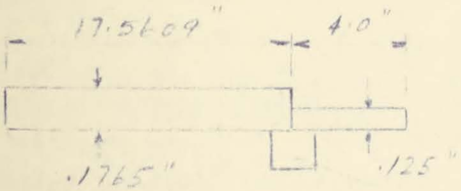
AREA FORWARD
= 1.675 × .2 + 1.925 × .2 +
+ 1.5 × .136 + 4.7156 × .25
= 2.1029

AREA AFT
= 2.7392 + 4.7156 × .25 =
REF. SHT 1-30
= 3.9181

$$w_f = 17.5609 \times \frac{2.1029}{3.9181} = 9.4258''$$

F/C/S & R/C/S - SPAR AREAS FORWARD & AFT ARE EQUAL.

REAR SPAR



.90 SQ. IN. SPAR FLANGE

AREA FWD = (17.5609 × .1765 + .45) × .98109 AREA AFT = (4 × .125 + .45) .98109

= 3.4821 = .9320

$$w_a = 17.5609 \times \frac{.9320}{3.4821} = 4.7002''$$

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

AIRCRAFT: C-105	Boundary Conditions	REPORT NO. 7/0510/7	SHEET NO. 7-11
		PREPARED BY R.N. SHEARLY	DATE JAN. 27/55
		CHECKED BY O. Gardner	DATE June '55

CALCULATION OF THE COEFFICIENTS RELATING RIB CAP MOMENT TO EXTERNAL RIB MOMENT AT T.J.

EXTERNAL RIB MOMENT M_F, M_R $\curvearrowright +$
RIB MOMENT $MR_{i'}$

$$MR_{i'} = K_F(i')\,M_F + K_R(i')\,M_R$$

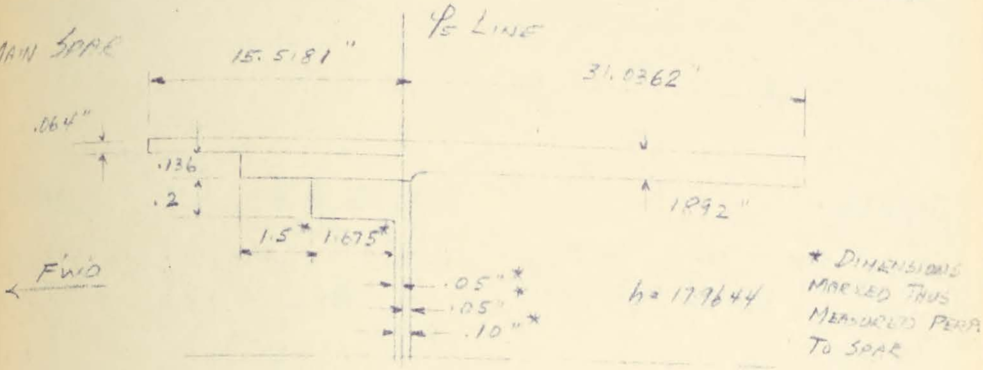
i'	1	2	3	4
$K_F(i')$	-1.000000	-.633101	-.287453	0
$K_R(i')$	0	.366899	.712547	1.000000

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

AIRCRAFT: C-105	Boundary Conditions	REPORT NO. 71/0510/1
		SHEET NO. 7-13 & 7-14
		PREPARED BY: R.N. SPEARLY — DATE: JAN 31/55
		CHECKED BY: [illegible] — DATE: June '56

SPAR AREAS AT THE CENTER LINE OF AIRCRAFT

MAIN SPAR



AREA FORWARD

$\{15.5181 \times .064\}.824438 + (1.5 + 1.725) \times .136 + 1.675 \times .2 + \frac{17.9644}{2} \times .05 = 2.0414$ □"

AREA AFT

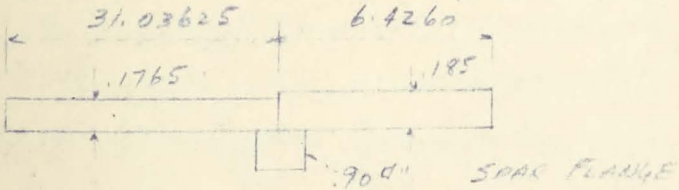
$\{31.0362 \times .1892\}.824438 + \frac{17.9644}{2} \times .05 = 5.2904$

TOTAL AREA = 7.3318

$$w_f = 31.0362 \times \frac{2.0414}{5.2904} = 11.9759"$$

FWD CENTER SPAR } SPAR AREAS FORWARD & AFT ARE
REAR CENTER SPAR } EQUAL.

REAR SPAR



AREA FWD = $(31.03625 \times .1765 + .45) \times .981019 = 5.8154$

AREA AFT = $(6.4260 \times .185 + .45) \times .981019 = 1.6077$

TOTAL AREA = 7.4231

$$w_a = 31.03625 \times \frac{1.6077}{5.8154} = 8.5802"$$

FORM 1319A

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

REPORT NO. 7/0510/7
SHEET NO. 7-17

AIRCRAFT: C-105

Boundary Conditions

PREPARED BY: R.M. SHEARLY — DATE: JAN 26/55
CHECKED BY: J. Gardner — DATE: June '5

CALCULATION OF THE COEFFICIENTS RELATING RIB CAP MOMENT TO EXTERNAL RIB MOMENT AT ℄

EXTERNAL RIB MOMENT M_F, M_R $\curvearrowright +$
RIB MOMENT $MR_{i''}$

$$MR_{i''} = K_F(i'')\, M_F + K_R(i'')\, M_R$$

i''	1	2	3	4
$K_F(i'')$	-1.000 000	-.560716	-.225626	0
$K_R(i'')$	0	.439284	.774374	1.000 000

A.V.ROE CANADA LIMITED MALTON — ONTARIO **TECHNICAL DEPARTMENT (Aircraft)**	REPORT NO. 7/0510/7 SHEET NO. 7-25
AIRCRAFT C105 | Wing Tip	PREPARED BY: D L Turner | DATE: 14 Jan. '55 CHECKED BY: | DATE:

Similarly, considering the last term gives the expression:

$$V_{M_1^2} = \frac{L}{6}\,\frac{M_1^2\,\Delta\xi}{EI_{x_1}}$$

Consider now the middle term of the integral:

$$2\int_{\xi_1}^{\xi_2} \frac{(\xi-\xi_1)(\xi_2-\xi)}{\xi^3}\,d\xi$$

$$=2\int_{\xi_1}^{\xi_2} \frac{-\xi^2+\xi(\xi_1-\xi_2)-\xi_1\xi_2}{\xi^3}\,d\xi$$

$$=2\left[-\log_e\frac{\xi_2}{\xi_1}+(\xi_1+\xi_2)\left(\frac{1}{\xi_1}-\frac{1}{\xi_2}\right)-\frac{1}{2}\xi_1\xi_2\left(\frac{1}{\xi_1^2}-\frac{1}{\xi_2^2}\right)\right]$$

$$=2\log_e\frac{\xi_1}{\xi_2}+\left(1+\frac{\xi_2}{\xi_1}-1-\frac{\xi_1}{\xi_2}\right)$$

$$=2\log_e\frac{\xi_1}{\xi_2}+\left(\frac{\xi_2}{\xi_1}-\frac{\xi_1}{\xi_2}\right)$$

$$=\left[\left(\log_e\frac{\xi_1}{\xi_2}\right)-\frac{\xi_1}{\xi_2}\right]+\left[\left(\log_e\frac{\xi_1}{\xi_2}\right)+\frac{\xi_2}{\xi_1}\right]$$

$$=\left[\left(\log_e\frac{\xi_1}{\xi_2}\right)-\frac{\xi_1}{\xi_2}\right]-\left[\left(\log_e\frac{\xi_2}{\xi_1}\right)-\frac{\xi_2}{\xi_1}\right]$$

$$=\left[\log_e\left(1+\frac{\xi_1-\xi_2}{\xi_2}\right)-\left(1+\frac{\xi_1-\xi_2}{\xi_2}\right)\right]-\left[\log_e\left(1+\frac{\xi_2-\xi_1}{\xi_1}\right)-\left(1+\frac{\xi_2-\xi_1}{\xi_1}\right)\right]$$

A.V.ROE CANADA LIMITED
MALTON – ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/7
SHEET NO. 7-27

PREPARED BY: D L Turner
DATE: 14 Jan. '55
CHECKED BY:
DATE:

AIRCRAFT C105

Wing Tip

$$= \frac{(\xi_2 - \xi_1)^3}{3\xi_1^2\xi_2^2} \; \frac{\xi_2 + \xi_1}{2} \left[3 - 2\, \frac{\xi_2^2 - \xi_1\xi_2 + \xi_1^2}{\xi_1\xi_2} \right]$$

$$= \frac{(\xi_2 - \xi_1)^3}{3\xi_1^2\xi_2^2} \; \frac{\xi_2 + \xi_1}{2} \left[3 - 2\, \frac{\xi_2^2 - 2\xi_1\xi_2 + \xi_1^2 + \xi_1\xi_2}{\xi_1\xi_2} \right]$$

$$= \frac{(\xi_2 - \xi_1)^3}{3\xi_1^2\xi_2^2} \; \frac{\xi_2 + \xi_1}{2} \left[1 - 2\, \frac{(\xi_2 - \xi_1)^2}{\xi_1\xi_2} \right]$$

hence:

$$V_{M_1M_2} = \frac{M_1M_2}{2} \; \frac{L}{EI_r \frac{t}{t_r}} \; \frac{\xi_{av}}{\Delta\xi^2} \; \frac{\Delta\xi^3}{3\xi_{gav}^4} \left(1 - 2\,\frac{\Delta\xi^2}{\xi_{gav}^2}\right)$$

where $\xi_{gav}^2 = \xi_1\xi_2$

$$V_{M_1M_2} = \frac{M_1M_2}{2} \; \frac{L}{EI_r \frac{t}{t_r}} \; \frac{\xi_{av}\,\Delta\xi}{3\xi_{gav}^4} \left(1 - 2\,\frac{\Delta\xi^2}{\xi_{gav}^2}\right)$$

$$= \frac{1}{6} \; \frac{L}{EI_r} \; \frac{t_r}{t} \; \frac{\xi_{av}\,\Delta\xi}{\xi_{gav}^4} \left(1 - 2\,\frac{\Delta\xi^2}{\xi_{gav}^2}\right)$$

Total bending energy then will be:

$$V = \frac{1}{6} \; \frac{L}{EI_r} \; \frac{t_r}{t} \left[\nu(\xi_2)\, M_2^2 + \nu(\xi_{2,1})\, M_2M_1 + \nu(\xi_1)\, M_1^2 \right]$$

where $\nu(\xi_2) = \dfrac{\Delta\xi}{\xi_2^3}$

$\nu(\xi_1) = \dfrac{\Delta\xi}{\xi_1^3}$

$\nu(\xi_{2,1}) = \dfrac{\xi_{av}\,\Delta\xi}{\xi_{gav}^4} \left(1 - 2\,\dfrac{\Delta\xi^2}{\xi_{gav}^2}\right)$

A.V.ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/7			
SHEET NO. 7-29			
AIRCRAFT C105	Wing Tip	PREPARED BY: D.L. Dume	DATE: 14 Jan. '55
		CHECKED BY:	DATE:

To put these energy expressions in terms of stresses at the designated stress points.

$$\sigma = \frac{M}{Z_r \frac{t}{t_r} \xi^2}$$ from sht. 7-21

hence: $M = \sigma Z_r \frac{t}{t_r} \xi^2$

substitute $Z_r = \frac{2 I_r}{h_r}$

then: $M = \frac{2 I_r}{h_r} \frac{t}{t_r} \xi^2 \sigma$

$$\tau_{gav} = \frac{T}{\Theta_r \frac{t}{t_r} \xi^2_{gav}}$$ from sht. 7-21

hence: $T = \tau_{gav} \Theta_r \frac{t}{t_r} \xi^2_{gav}$

substitute $\Theta_r = J_r \frac{b_r}{A_r}$

then: $T = J_r \frac{b_r}{A_r} \frac{t}{t_r} \xi^2_{gav} \tau$

A.V.ROE CANADA LIMITED
MALTON – ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

REPORT NO. 7/0510/7
SHEET NO. 7-30
PREPARED BY: D L Turner DATE: 14 Jan '55
CHECKED BY: DATE:

AIRCRAFT C105

Wing Tip

Substituting this expression for M in the bending energy equation:

$$V = \frac{1}{6}\frac{L}{EI_r}\frac{t_r}{t}\left(\frac{2I_r}{h_r}\frac{t}{t_r}\right)^2\left[\frac{\Delta\xi}{\cancel{\xi_2^3}}\xi_2^4\sigma_2^2 + \frac{\xi_{av}\Delta\xi}{\cancel{\xi_{gav}^4}}\left(1-2\frac{\Delta\xi^2}{\xi_{gav}^2}\right)\cancel{\xi_1^2\xi_2^2}\sigma_1\sigma_2 + \frac{\Delta\xi}{\cancel{\xi_1^3}}\xi_1^4\sigma_1^2\right]$$

$$= \frac{1}{6}\frac{L}{E\cancel{I_r}}\frac{\cancel{t_r}}{\cancel{t}}\frac{4I_r^2}{h_r^2}\frac{t^2}{t_r^2}\left[\Delta\xi\,\xi_2\sigma_2^2 + \xi_{av}\Delta\xi\left(1-2\frac{\Delta\xi^2}{\xi_{gav}^2}\right)\sigma_1\sigma_2 + \Delta\xi\xi_1\sigma_1^2\right]$$

$$= \frac{1}{2}\frac{4LI_r}{3Eh_r^2}\frac{t}{t_r}\Delta\xi\left[\xi_2\sigma_2^2 + \bar{\xi}_{av}\sigma_1\sigma_2 + \xi_1\sigma_1^2\right]$$

where $\bar{\xi}_{av} = \xi_{av}\left(1-2\frac{\Delta\xi^2}{\xi_{gav}^2}\right)$

Substituting the expression for T in the torsional energy equation:

$$V = \frac{1}{2}\frac{L}{GJ_r}\frac{t_r}{t}\frac{\xi_{av}\Delta\xi}{\xi_{gav}^4}\left(J_r\frac{b_r}{A_r}\frac{t}{t_r}\xi_{gav}^2 T\right)^2$$

$$= \frac{1}{2}\frac{L}{G\cancel{J_r}}\frac{\cancel{t_r}}{\cancel{t}}\frac{\xi_{av}\Delta\xi}{\cancel{\xi_{gav}^4}}\frac{J_r^2 b_r^2}{A_r^2}\frac{t^2}{t_r^2}\cancel{\xi_{gav}^4}T^2$$

$$= \frac{1}{2}\frac{J_r b_r L}{GA_r^2}\frac{t}{t_r}\xi_{av}\Delta\xi\, T^2$$

A.V. ROE CANADA LIMITED
MALTON – ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT C105	Wing Tip	REPORT NO. 7/0510/7	SHEET NO. 7-31
		PREPARED BY: D L Turner	DATE: 14 Jan. 55
		CHECKED BY:	DATE:

Summary of Energy Equations for a Single Bay

In terms of M & T

$$V=\frac{1}{2}\left\{\frac{1}{3}\,\frac{L}{EI_r}\,\frac{t_r}{t}\left[\nu(\xi_2)M_2^2+\nu(\xi_{2,1})M_1M_2+\nu(\xi_1)M_1^2\right]\right.$$
$$\left.+\frac{L}{GJ_r}\,\frac{t_r}{t}\,K(\xi)T^2\right\}$$

where $\nu(\xi_2)=\dfrac{\Delta\xi}{\xi_2^3}$ $\quad\nu(\xi_1)=\dfrac{\Delta\xi}{\xi_1^3}$ $\quad\nu(\xi_{2,1})=\dfrac{\xi_{av}\Delta\xi}{\xi_{geo}^4}\left(1-2\dfrac{\Delta\xi^2}{\xi_{geo}^2}\right)$

$$K(\xi)=\frac{\xi_{av}\,\Delta\xi}{\xi_{geo}^4}$$

In terms of stresses (σ & τ)

$$V=\frac{1}{2}\left\{\frac{4LI_r}{3Eh_r^2}\,\frac{t}{t_r}\,\Delta\xi\left[\xi_2\sigma_2^2+\bar{\xi}_{av}\sigma_1\sigma_2+\xi_1\sigma_1^2\right]+\frac{J_r b_r^2 L}{GA_r^2}\,\frac{t}{t_r}\,\xi_{av}\,\Delta\xi\,\tau^2\right\}$$

where $\bar{\xi}_{av}=\xi_{av}\left(1-2\dfrac{\Delta\xi^2}{\xi_{geo}^2}\right)$

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

REPORT No. 7/0510/7 Check
SHEET No. 7.32

AIRCRAFT		PREPARED BY	DATE
C105	Wing Analysis	[signature]	24 May '55
		CHECKED BY	DATE

Check on Tia Matrix (Wing)

The Tia matrix (wing) is checked, (except for rib stresses), according to the method outlined below. Note that the outer wing and inner wing are checked separately due to the change in coordinate systems.

First the torque, bending moment and shear, at each rib and the X (or X_0) axis, are calculated from the basic geometry, assuming unit loads applied at each load point.

Then, using the spar and panel stresses, (at the rib under consideration), as defined in the Tia matrix, and again assuming unit loads, the torque, bending moment, and shear at the same points as above are calculated.

These two sets of loads should check against each other.

Rib stresses may be checked quickly by plotting them on the grid layout of the structure and checking visually.*

Note that this check can only be applied to the wing portion of the structure.

* A FURTHER CHECK IS MADE ON THE T.J., FUSELAGE SIDE & CENTRE LINE RIBS SEE SHTS 7.44 a TO 7.44 i

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

REPORT NO.	7/0510/7 Check
SHEET NO.	7-34
AIRCRAFT: C105	Wing Analysis
PREPARED BY	[illegible signature]
DATE	23 June [illegible]
CHECKED BY	
DATE	

Check on Cix Matrix

To date no mechanical means of checking the Cix matrix has been found other than by the following not too positive checks

1. Repeat the calculations
2. Check patterns on dummy matrix
3. Plot coefficients on the structural grid and check numerical trends across the structure.

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

AIRCRAFT: C105	Wing Analysis	REPORT NO. 7/0510/7 Check SHEET NO. 7-35	
		PREPARED BY: D L Turner	DATE: 23 June '55
		CHECKED BY	DATE

Matrix Checks

On the following pages, values of shear, torque & bending moment at the appropriate sections (as suggested on shts 7-32 et seq.) are calculated from the basic geometry. These values are used for cross checking the T_{ia} checks carried out on Check Figures 1 & 2

On sheets 7-45 et seq. the redundancy check as suggested is carried out.

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

AIRCRAFT:		REPORT NO. 7/0510/7 Check	
C105	Wing Analysis Outer Wing	SHEET NO. 7-36	
		PREPARED BY	DATE
		[signature]	23 June 5?
		CHECKED BY	DATE

Check on Tia Matrix

Determination of torque, bending moment and shear at rib-X (or X_0) axis intersection from basic geometry for unit load at each load point.

Outer wing (X_0 axis) ref. fig. 5

At $X_0 = 173.3489$ (rib 18) Total shear = 12

$$\begin{aligned} B.M. &= (173.3489 - 57.1701) + 2(173.3489 - 38.6262) + 3(173.3489 - 78.3819) \\ &\quad + 3(173.3489 - 110.3819) + 3(173.3489 - 142.3819) \\ &= 952.3272 \end{aligned}$$

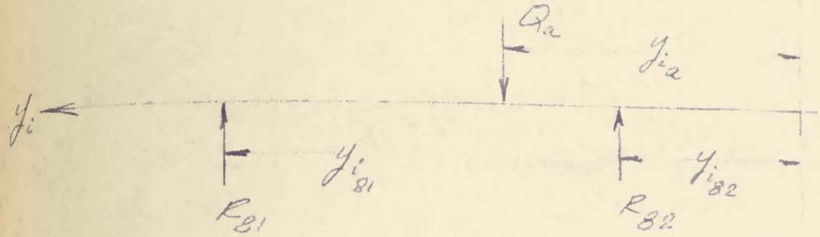
$$\begin{aligned} \text{Torque} = -\Sigma Y_0 &= -(21.0826 - 1.4110 - 16.4110 + 28.2049 - 2.8633 \\ &\quad - 17.8633 + 40.2055 - 4.0323 - 19.0323 + 52.5062 \\ &\quad - 5.2012 - 2.2012) \\ &= -56.1836 \end{aligned}$$

Section		Shear Force	M. Arm	ΔM	M	$-\Sigma Y_0 = \Delta T$	T
Rib 18	A	12			952.3272		$\overline{56}.1836$
$X_0 = 189.8159$	B	$Q_A + Q_{13} + \sum_{20}^{24} Q_a = 18$	16.4670	296.4060	1248.7332	$\overline{128}.9732$	$\overline{185}.1568$
$X_0 = 208.8917$	C	$Q_B + Q_{19} + \sum_{25}^{29} Q_a = 24$	19.0758	457.8192	1706.5524	$\overline{142}.6496$	$\overline{327}.8064$
$X_0 = 231.2323$	D	$Q_C + \sum_{30}^{34} Q_a = 29$	22.3406	647.8774	2354.4298	$\overline{181}.1238$	$\overline{508}.9302$
$X_0 = 253.5731$	E	$Q_D + \sum_{35}^{38} Q_a = 33$	22.3408	737.2464	3091.6762	$\overline{208}.9414$	717.8716
$X_0 = 275.9137$	F	$Q_E + \sum_{39}^{41} Q_a = 36$	22.3406	804.2616	3895.9378	$\overline{207}.7804$	$\overline{925}.6520$

Inner Wing (X axis) ref. fig. 10

DETERMINATION OF THE REACTIONS R_{81} & R_{82} IN TERMS OF THE APPLIED LOAD Q_a FOR $a = 1$ TO 67 EXCLUSIVE OF 53 & 63

CONSIDER THE PROJECTION OF ALL LOADS & REACTIONS IN THE PLANE OF SYMMETRY OF THE AIRCRAFT.



MOMENTS ABOUT REACTION POINT OF R_{82}

$$R_{81} = \frac{1}{y_{i_{81}} - y_{i_{82}}} \sum \left(y_{i_a} - y_{i_{82}}\right) Q_a = \sum r_{81,a} Q_a$$

WHERE $r_{81,a} = \dfrac{y_{i_a} - y_{i_{82}}}{y_{i_{81}} - y_{i_{82}}}$

SIMILARLY $R_{82} = \sum r_{82,a} Q_a$ WHERE $r_{82,a} = 1 - r_{81,a}$

THE VALUES OF $r_{81,a}$ ARE COMPUTED ON THE FOLLOWING SHEET.

FROM THE FOLLOWING SHEET IT CAN BE SEEN THAT

$$\left.\begin{aligned} \sum r_{81,a} &= 5.992751 \\ \sum r_{82,a} &= 65 - \sum r_{81,a} = 59.007249 \end{aligned}\right\} \text{FOR UNIT VALUES FOR } Q_a\text{'s}$$

158.0623

A. V. R

ENG

AIRCRAFT

WEIGHT

C. G. POS

7	8	9	10	11	12	13	14	15	16	17	18	19	20	
f_2 − 63.9171	$G_{3,2}$ = (7) / 158.0623		x	f_2	f_2 − 63.9171	$G_{3,2}$ = (12) / 158.0623		x	f_2	f_2 − 63.9171	$G_{3,2}$ = (17) / 158.0623		x	
6.7623	−.233847		31	91.5321	27.6132	.174710		46	86.0248	22.1077	.139867		61	126.
41.7624	−.264129		32	78.0988	[illegible]	.087722		47	47.1548	−16.7623	−.106049		62	69
.1625	−.228786		33	62.0670	−1.8591	−.011705		48	181.2387	117.3218	.742250		63	−
6.3527	.862652		34	42.6084	−21.3087	−.134812		49	138.2214	74.3043	.470095		64	26
[illegible]	.07416		35	128.5488	64.5897	.408634		50	95.2038	31.2867	.197939		65	11
3.3078	−.01664		36	110.3162	46.4294	.293741		51	52.1763	−11.7358	−.074216		66	13
.4655	−.085191		37	95.4764	31.5593	.199664		52	200.5774	136.6603	.864599		67	−
.7124	−.169360		38	77.7309	13.8137	.087394		53	152.9698	89.0527	.563402			
2.1031	−.271520		39	141.0751	25.1588	.538767		54	105.3622	4.4451	.262207			
.457	.162387		40	129.1608	65.2437	.412772		55	57.7546	−6.1625	−.038188			
5500	.07372		41	112.8532	48.9361	.309600		56	169.2920	105.3749	.666667			
.566	−.004154		42	169.6448	105.7277	.668899		57	116.6146	52.6975	.333333			
[illegible]	−.086318		43	147.9754	84.0583	.531805		58						
1060	−.208124		44	163.7649	99.8478	.631699		59	241.7497	177.8321	1.125071			
[illegible]	.278522		45	124.8349	60.1778	.385783		60	184.3697	120.4526	.762058			

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT		REPORT NO. 7/0510/7 Check	
		SHEET NO. 7-39	
C-105	Inner Wing & Centre Section	PREPARED BY: R.N. Shearly	DATE: June 13/55
		CHECKED BY	DATE

MOMENT, TORQUE & SHEAR FORCE ACTING ON SECTIONS OF INNER WING, FOR UNIT Q_a's (FOR $a = 1 - 67$)

SECTION G $(x = 215.5596)$

TOTAL SHEAR FORCE $= \sum_{a=1}^{14} Q_a + \sum_{a=30}^{42} Q_a = \underline{34}$ (POSITIVE DOWN)

EXCLUDING 34, 38 & 41

BENDING MOMENT $= Q_a\,(215.5596 - x_a) =$

$= 34 \times 215.5596 - 2 \times 80.2204 - 91.6120 - \cdots$

$\cdots - 197.6952 = \underline{1925.9084}$ (POSITIVE TENSION IN TOP)

TORQUE $= -\sum y_a = \underline{-1453.1028}$ (POSITIVE NOSE UP)

SECTION	x	SHEAR FORCE (POS. DOWN)	MOMENT ARM	ΔM	MOMENT (POS. TENSION IN TOP)	$\sum y = -\Delta T$	TORQUE (POSITIVE NOSE UP)
G	215.5596	34			1925.9084		−1453.1028
H	239.5602	$Q_{a} + Q_{30} + Q_{38} + Q_{41} + Q_{43} + Q_{9} + Q_{15} = 40$	23.0006	920.0240	2845.9324	−594.0460	−2047.1488
J	264.0150	$Q_{H} + Q_{16} + \sum_{40} Q_a = 45$	25.4548	1145.4660	3991.3984	−438.9942	−2486.1430
K	292.1859	$Q_J + Q_{17} + \sum_{48}^{51} Q_a = 50$	28.1709	1408.5450	5399.9434	−489.0187	−2975.1617
L	323.3626	$Q_K + Q_{18} + \sum_{52}^{55} Q_a = 55$	31.1766	1714.7130	7114.6564	−544.4186	−3519.5803
M	352.1625	$Q_L + Q_{56} + Q_{57} + R_{81} + R_{82} = -8$	28.8000	−230.4000	6884.2564	485.9429	−1296.3626
N	380.9625	$Q_M + \sum_{59}^{66} Q_a = -4$	28.8000	−115.2000	6769.0564	−622.7190	−673.6436

NOTE FOR UNIT Q_a's

$R_{81} = 5.992751$ up $R_{82} = 51.007249$

DETERMINATION OF THE REACTIONS R_{81} & R_{82} IN TERMS OF THE INTERACTION LOADS Q_a FOR $a =$ 101 TO 114 WHERE Q_a APPLIES TO LOADS, MOMENTS & TORQUES.

IN THIS WORK IT MUST BE REMEMBERED THAT THE INTERACTION LOADS ARE NOT NECESSARILY SINGLE LOADS ACTING AT A POINT BUT ARE IN FACT AS LISTED BELOW

TYPE (1) SINGLE LOAD ACTING AT A POINT [101], [104], [105], [106], [113] & [114].
TYPE (2) MOMENT ACTING AT A POINT [102] [103] & [107].
TYPE (3) GROUP OF LOADS MAKING UP ONE INTERACTION LOAD [108], [109], [110], [111] & [112].

CONSEQUENTLY THE DETERMINATION OF THE REACTIONS IN TERMS OF THE INTERACTION LOADS WILL HAVE TO BE MADE WITH CARE.

TYPE (1) SEE SHT.
MOMENTS ABOUT REACTION POINT OF R_{82}

$$R_{81} = \frac{1}{y_{i_{81}} - y_{i_{82}}} \Sigma (y_{ia} - y_{i_{82}}) Q_a = \Sigma r_{81,a} Q_a$$

WHERE $r_{81,a} = \dfrac{y_{ia} - y_{i_{82}}}{y_{i_{81}} - y_{i_{82}}}$

$R_{82} = \Sigma r_{82,a} Q_a$ WHERE $r_{82,a} = 1 - r_{81,a}$

a	① y_{ia}	② $y_{ia} - 63.9171$	③ $r_{81,a} = ② / 153.0623$
101	200.2693	136.3527	.862652
104	276.5201	212.6030	1.345058
105	169.2920	105.3749	.666667
106	116.6046	52.6875	.333333
113	137.3751	73.4580	.464741
114	75.3026	11.3855	.072032

TYPE (2)

THE COMPONENT OF THE MOMENT PERPENDICULAR TO THE LINE JOINING THE TWO REACTION POINTS WHEN DIVIDED BY THE DISTANCE BETWEEN THESE POINTS YIELDS THE VALUE OF THE REACTIONS.

a	COMPONENT	REACTION
102	.888739	.00562271
103	.458414	.00290021
107	1	.00632662

$\beta = 27° - 17.09'$

$\cos\beta = .888739$

$\sin\beta = .458414$

FOR MOMENTS OF 100 "# THE ABOVE VALUES TABULATED IN THE COMPONENT & REACTION COLUMNS SHOULD BE MULTIPLIED BY 100

TYPE (3)

THIS TYPE HAS NO AFFECT ON THE REACTIONS. (ANOTHER AND PERHAPS BETTER WAY OF CONSIDERING IN THIS TYPE WILL BE DEALT WITH ON SHT. 7-41 a)

SUMMARY

$$R_{81} = .862652\, Q_{101} + .00562271\, Q_{102} + .00290021\, Q_{103} + 1.345058\, Q_{104} + + .666667\, Q_{105} + .333333\, Q_{106} - .00632662\, Q_{107} + .464741\, Q_{113} + + .072032\, Q_{114}$$

$$R_{82} = .137348\, Q_{101} - .00562271\, Q_{102} - .00290021\, Q_{103} - .345058\, Q_{104} + + .333333\, Q_{105} + .666667\, Q_{106} + .00632662\, Q_{107} + .535259\, Q_{113} + + .927968\, Q_{114}$$

FOR UNIT VALUES OF Q_a THESE EQUAL

$R_{81} = 3.746679$ $\qquad$ $R_{82} = 2.253321$

FOR UNIT VALUES OF Q_a FOR $a = 101, 104-106$ incl & $108-114$ incl. AND VALUE OF MOMENTS Q_{102}, Q_{103} & Q_{107} EQUAL 100 "#

$R_{81} = 3.964113$ $\qquad$ $R_{82} = 2.035887$

IF THE TYPE (3) INTERACTION LOADS ARE CONSIDERED AS FOLLOWS THE FOLLOWING REACTIONS ARE OBTAINED.

THE COMPONENT OF INTERACTION LOAD $\begin{Bmatrix}108\\110\\111\end{Bmatrix}$ ON THE FUSELAGE SIDE BEAM RIB CANCELS INTERACTION LOAD $\begin{Bmatrix}104\\105\\106\end{Bmatrix}$

AND THE COMPONENT OF INTERACTION LOADS 109 & 112 ON THE FUSELAGE SIDE BEAM RIB IS IMMEDIATELY REACTED.

CONSEQUENTLY FOR UNIT LOADS & 100 "# MOMENTS

$$R_{81} = .862652\,Q_{101} + .562271\,Q_{102} + .290021\,Q_{103} - .632662\,Q_{107} +$$
$$+ 1.345058\,Q_{108} + Q_{109} + .666667\,Q_{110} + .333333\,Q_{111} +$$
$$+ .464741\,Q_{113} + .072032\,Q_{114} = 4.964113$$

$$R_{82} = .137348\,Q_{101} - .562271\,Q_{102} - .290021\,Q_{103} + .632662\,Q_{107} -$$
$$- .345058\,Q_{108} + .333333\,Q_{110} + .666667\,Q_{111} + Q_{112}$$
$$+ .535259\,Q_{113} + .927968\,Q_{114} = 3.035887$$

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT		REPORT NO 7/0510/7 Check	
C-105	Inner Wing & Centre Section	SHEET NO 7-42	
		PREPARED BY	DATE
		E.M. Sheanly	June 14/55
		CHECKED BY	DATE

SHEAR FORCE, MOMENT & TORQUE ACTING ON SECTIONS OF INNER & CENTER SECTION WING FOR UNIT INTERACTION LOADS, Q_a FOR a = 101 TO 114

SECTION G (x = 215.5596)

TOTAL SHEAR FORCE $= Q_{101} = 1$ (POSITIVE DOWN)

BENDING MOMENT $= -\sin\beta\, Q_{102} + \cos\beta\, Q_{103} =$

$= -.458414 + .888739 = .430325$ (POSITIVE TEN. IN TOP)

$= 43.0325$ " For Moment = 100"#

TORQUE $= -y_{i_{101}} Q_{101} - \cos\beta\, Q_{102} - \sin\beta\, Q_{103} =$

$= -200.2698 - .888739 - .458414 = -201.616953$ (POSITIVE NOSE UP) For unit Moments

FOR LOADS = 1# & MOMENTS = 100"# $= -334.9851$ For Moment of = 100"#

SECTION	x	SHEAR FORCE (POS. DOWN)	MOMENT ARM	ΔM	MOMENT (POS. TENSION IN TOP)	ΔT	TORQUE (POSITIVE NOSE UP)
G	215.5596	1			430.325		-334.9851
H	238.5602	$Q_G = 1$	23.0006	23.0006	66.0331		-334.9851
J	264.0150	$Q_H = 1$	25.4548	25.4548	91.4879		-334.9851
K	292.1859	$Q_J = 1$	28.1709	28.1709	119.6588		-334.9851
L	323.3626	$Q_K = 1$	31.1766	31.1766	150.8354		-334.9851
M	352.1625	$Q_L - R_{81} - R_{92} - Q_{104} - Q_{112} = -7$	23.8000	-201.6000	-50.7646	4.9641134×211.5974 + 3.0350501×63.9171 + 100	1060.9908
N	380.9625	$Q_M = -7$	23.8000	-201.6000	-252.3646		1060.9908

INCLUDING ℄ LOADS SHEAR FORCE $= Q_N + 7 = 0$

$B.M. = B.M._N = -252.3646$

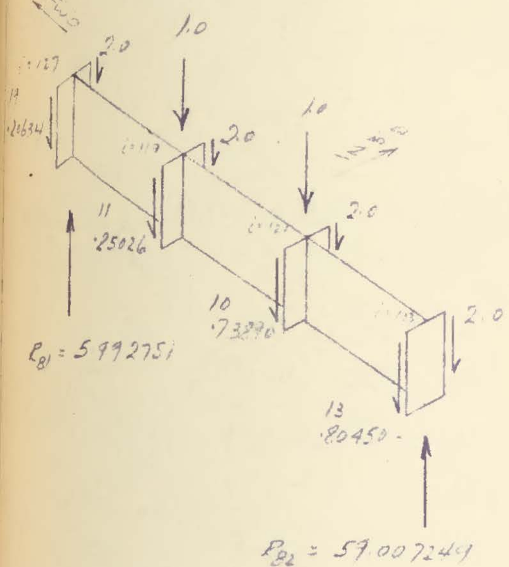
$T = T_N - 1\{276.5201 + 221.9794 + 169.2920 + 116.6046 + 137.3751 + 116.6046 + 75.3026 + 63.9171\} =$
$= 1060.9908 - 1060.9909 \approx 0$

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

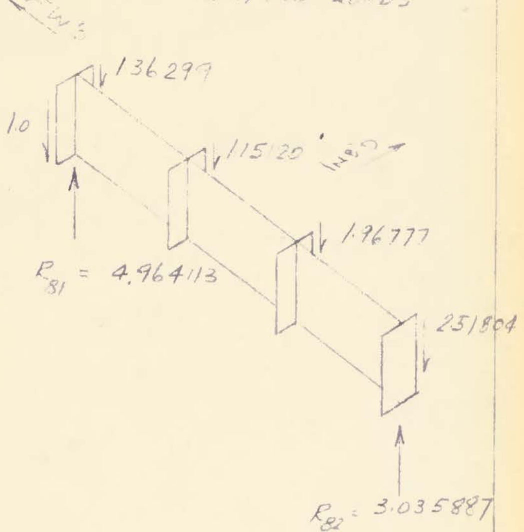
AIRCRAFT: C-105	INNER WING & CENTER SECTION WING
REPORT NO 7/0510/7 CHECK	SHEET NO 7-44a
PREPARED BY R.N. SHERRY	DATE JULY 7/55
CHECKED BY	DATE

THE FUSELAGE SIDE BEAM RIB CAN CHECKED BY CONSIDERING THE RIB TO BE ISOLATED AND APPLYING THE LOADS (SHEAR FORCES, LOADS, TORQUES) TO IT AND OBTAIN MOMENTS AT STRESS POINTS L=119 & 121 WHICH CHECK WITH THE VALUES OBTAINED BY MULTIPLYING THE APPROPRIATE STRESS BY THE APPROPRIATE RIB AREA & HEIGHT. LOADS & INTERACTION LOADS EQUAL TO 1# & INTERACTION MOMENTS EQUAL TO 100"# USED THROUGH OUT.

FOR UNIT LOADS



FOR INTERACTION LOADS

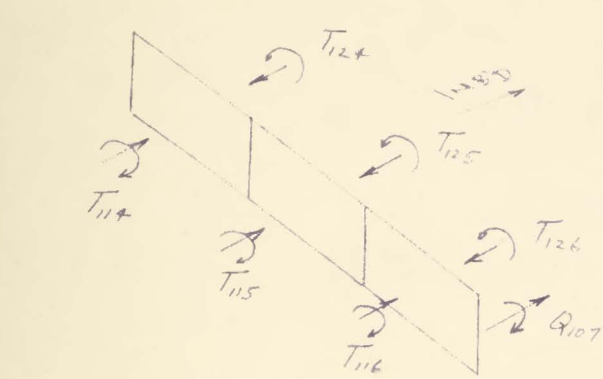


VERTICAL LOADS, SHEAR FORCES & REACTIONS ACTING ON RIB

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT		REPORT NO. 7/0510/7 CHECK
C-105	INNER WING & CENTER SECTION WING	SHEET NO. 7-44 b
		PREPARED BY: R.M. SHEMRLY — DATE: JULY 8/55
		CHECKED BY: — DATE:

FOR UNIT LOADS & INTERACTION LOADS



TORQUES

i	$\frac{1}{4t_{NEW}}$ CHECK FIG. 2 COL. 9	τ TIA MATRIX	$T = \frac{\tau}{\frac{1}{4t_{NEW}}}$ IN.LB.
114	.0050449	1.85313	367.3274
115	.0061516	2.12778	345.9230
116	.0084107	3.17777	377.8247
124	.0040634	$\bar{5}.80752$	$\overline{14}29.2268$
125	.0051195	$\bar{5}.57881$	$\overline{10}89.7177$
126	.0072546	$\bar{5}.32272$	$\overline{7}33.7303$

$Q_{107} = 100$ IN LB.

$\therefore$ MOMENT AT $(i = 123) =$ 100 $= \underline{100}$ IN.LB.

MOMENT AT $(i = 121) =$ $100 + 377.8247 + 733.7303 + (13.80450 + 2.0 + 2.51804 - 59.007249 - 3.035887)\ 52.6872 = \underline{\overline{1}091.9608}$ IN.LB.

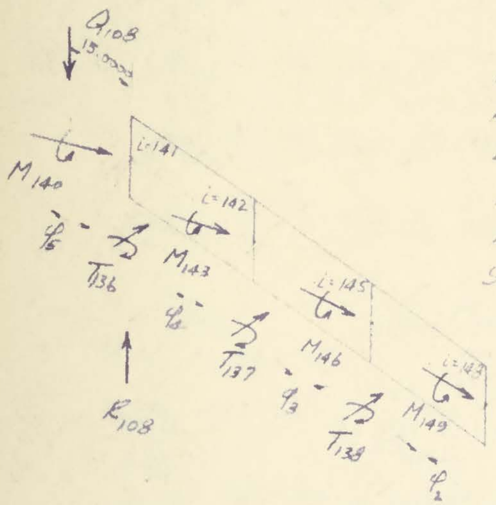
MOMENT AT $(i = 119) =$ $100 + 377.8247 + 345.9230 + 733.7303 + 1089.7177 +$
$+ (13.80450 + 2.0 + 2.51804 - 59.007249 - 3.035887)\ 2 \times 52.6872 +$
$+ (10.73870 + 1.0 + 2.0 + 1.16777)\ 52.6872 = \underline{\overline{1}132.2954}$ IN LB.

MOMENT AT $(i = 127) =$ $100 + 377.8247 + 345.9230 + 367.3274 + 733.7303 + 1089.7177 + 1429.2268 +$
$+ (-43.720596)\ 158.0623 + (15.70667) \times 2 \times 52.6872 +$
$+ (11.25226 + 1.0 + 2.0 + 1.15120)\ 52.6872 = 6910.6906 - 6910.5780 = .1126$

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C-105	CENTER SECTION WING	REPORT NO. 7/0510/7 CHECK	
		SHEET NO. 7-44 d	
		PREPARED BY	DATE
		R.N. SHEARLY	JULY 14/55
		CHECKED BY	DATE

THE AIRCRAFT CENTER LINE RIB CAN BE CHECKED BY CONSIDERING THE RIB TO BE ISOLATED AND APPLYING THE LOADS (SPAR MOMENTS, CELL TORQUES & INTERACTION LOAD Q_{108}) TO IT AND OBTAIN MOMENTS AT STRESS POINTS $i = 141, 142, 145$ & 148 WHICH CHECK WITH THE VALUES OBTAINED BY MULTIPLYING THE APPROPRIATE STRESS BY THE APPROPRIATE RIB AREA & HEIGHT. LOADS & INTERACTION LOADS EQUAL TO 1 # & INTERACTION MOMENTS EQUAL TO 100 "# USED THROUGH OUT.



R_{108} REPRESENTS THE REACTION TO INTERACTION LOAD Q_{108} WHICH IS IN FACT THE SHEAR FORCE IN THE MAIN SPAR WEB.

$$\left.\begin{aligned} \sin\varphi_5 &= .565753 \\ \sin\varphi_4 &= .463818 \\ \sin\varphi_3 &= .339219 \\ \sin\varphi_2 &= .193912 \end{aligned}\right\} \text{Ex. Fig. 10}$$

SPAR MOMENTS, CELL TORQUES & INTERACTION LOAD Q_{108} ACTING ON ℄ RIB.

EQUILIBRIUM CHECK :

$$T_{136} + T_{137} + T_{138} + M_{108} + \sin\varphi_5\, M_{140} + \sin\varphi_4\, M_{143} + \sin\varphi_3\, M_{146} +$$

$$+ \sin\varphi_2\, M_{149} = 0$$

WHERE $M_{108} = -15\, Q_{108}$

$$\text{OR } T_{136} + T_{137} + T_{138} + (M_{141})_F + (M_{141}) + (M_{142}) + (M_{145}) + (M_{148}) = 0$$

SEE NEXT SHEET.

RIB MOMENT CHECK :

$$M_{141} = -1 \times (M_{141})_F \; - .164464 \times (M_{141})$$

$$M_{142} = -.560716\{(M_{141})_F + (M_{141})\} - .060716 \times (M_{142}) + .439284\{(M_{145}) + (M_{148})\}$$

$$M_{145} = -.225626\{(M_{141})_F + (M_{141}) + (M_{142})\} + .274374 \times (M_{145}) + .774374 \times (M_{148})$$

$$M_{148} = .162736 \times (M_{148})$$

REF SHT 7-20

WHERE $(M_{141})_F = M_{108}$

$(M_{141}) = \sin \phi_5 \; M_{140}$

$(M_{142}) = \sin \phi_4 \; M_{143}$

$(M_{145}) = \sin \phi_3 \; M_{146}$

$(M_{148}) = \sin \phi_2 \; M_{149}$

$t_{old}/t_{new} = .150/.155 = .967742$

	1	2	3	4	5	6	7	8	9
i	h	A_s or A_r	$h A_s$ or $h A_r$	$\frac{1}{4' t_{old}}$	$\frac{1}{4' t_{new}}$	σ	τ	M	T
SH.	FIG. 10	FIG 10	① × ②	SHT 5-80	④ × .967742	T_{14} MATRIX	T_{18} MATRIX	③ × ⑥	⑦/⑤
136				.00357506	.00345974		$\bar{4}.94464$		$\overline{1429.194}$
137				.00462753	.00447826		$\bar{4}.88264$		$\overline{1090.298}$
138				.00676713	.00654884		$\bar{4}.80561$		$\overline{733.811}$
140	17.9644	6.1089	109.743			30.15631		3309.444	
143	14.4880	10.5915	153.450			10.05883		1543.527	
146	10.2672	11.4098	117.147			11.15461		1306.729	
149	6.4011	7.3251	46.889			25.89862		1214.360	
141	17.9644	1.0000	17.9644			$\overline{16.31313}$		$\overline{293.056}$	
142	14.4880	2.2320	32.3372			$\overline{24.34389}$		$\overline{787.213}$	
145	10.2672	4.0930	42.0236			$\overline{6.58715}$		$\overline{276.816}$	
148	6.4011	2.7400	17.5390			2.18515		38.325	

$(M_{141})_F = -15$

$(M_{141}) = .565953 \times 3309.444 = 1872.990$

$(M_{142}) = .463818 \times 1543.527 = 715.916$

$(M_{145}) = .339219 \times 1306.729 = 443.267$

$(M_{148}) = .193912 \times 1214.360 = 235.479$

EQUILIBRIUM CHECK:

$-1429.194 - 1090.298 - 733.811 - 15.000 + 1872.990 + 715.916 +$

$+ 443.267 + 235.479 =$

$= -3268.303 + 3267.652 = -.651$ % ERROR ↓ .0199

RIB MOMENT CHECK:

$M_{141} = -1 \times -15 - .164464 \times 1872.990 = -293.039$ .0055

$M_{142} = -.560716 \times 1857.990 - .060716 \times 715.916 + .439284 \times 678.746 =$

$= -787.110$ .0131

$M_{145} = -.225626 \times 2573.906 + .274374 \times 443.267 + .774374 \times 235.479 =$

$= -276.770$ .0166

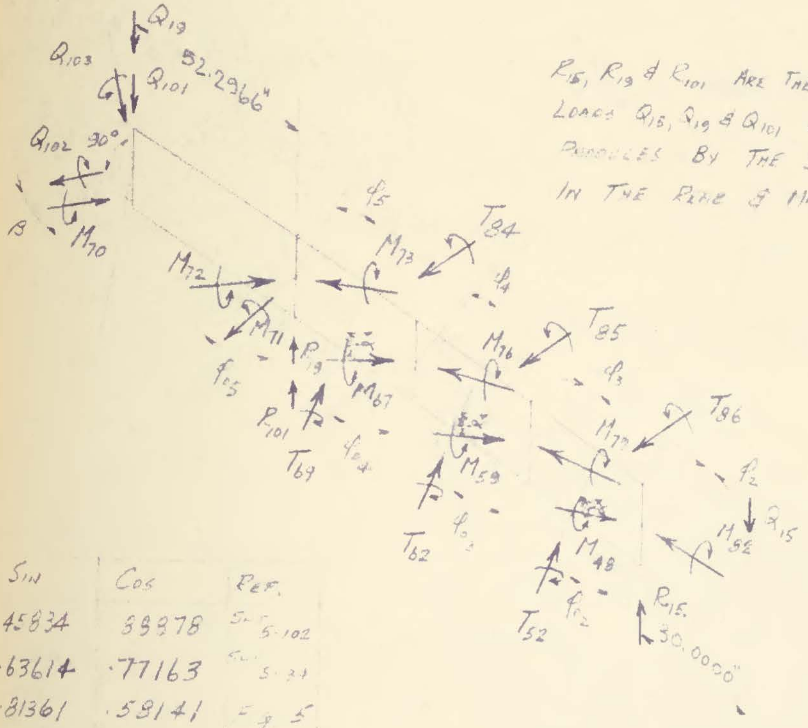
$M_{148} = .162736 \times 235.479 = 38.321$ .0164

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT		REPORT NO. 7/0510/7 CHECK
C-105	OUTER WING / INNER WING	SHEET NO. 7-44 9
		PREPARED BY: R.N. SHEPPLY — DATE: JULY 15 /55
		CHECKED BY: — DATE:

THE TRANSPORT JOINT RIB CAN BE CHECKED BY CONSIDERING THE RIB ISOLATED AND APPLYING THE LOADS (SPAR MOMENTS, CELL TORQUES, LOADS Q_{15} & Q_{19}, INTERACTION LOAD Q_{101} & INTERACTION MOMENTS Q_{102} & Q_{103}.) TO IT AND OBTAIN MOMENTS AT STRESS POINTS $i = 74, 75, 78$ & 81 WHICH MUST CHECK WITH THE VALUE OBTAINED BY MULTIPLYING THE APPROPRIATE STRESS BY THE APPROPRIATE RIB AREA & HEIGHT. LOADS & INTERACTION LOADS EQUAL TO 1# & INTERACTION MOMENTS EQUAL TO 100"# USED THROUGH OUT.

R_{15}, R_{19} & R_{101} ARE THE REACTIONS TO LOADS Q_{15}, Q_{19} & Q_{101} WHICH ARE PRODUCED BY THE SHEAR FORCE IN THE REAR & MAIN SPAR WEBS



ANGLE	SIN	COS	REF.
β	.45834	.88878	SH. 5.102
α	.63614	.77163	SH. 5.34
φ_{05}	.81361	.58141	FIG. 5
φ_{04}	.76503	.64399	"
φ_{03}	.69867	.715446	"
φ_{02}	.60752	.79430	"
φ_5	.565955	—	FIG. 10
φ_4	.463818	—	"
φ_3	.339219	—	"
φ_2	.193912	—	"

SPAR MOMENTS, RIB MOMENT, CELL TORQUES, LOADS, INTERACTION LOAD & INTERACTION MOMENTS ACTING ON T.J. RIB.

AVRO AIRCRAFT LIMITED

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT		REPORT NO. 7/0510/7 CHECK
C-105	OUTER WING, INNER WING	SHEET NO. 7-41 h
		PREPARED BY: R.N. SHEYRLY — DATE: JULY 15/55
		CHECKED BY: — DATE:

EQUILIBRIUM CHECK :

$$-52.2944(Q_{19}+Q_{101}) - \cos\beta\, Q_{102} - \sin\beta\, Q_{103} + M_{74}' + \sin\phi_{05}\, M_{72} -$$

$$- \cos\alpha\, M_{71} + \sin\phi_{04}\, M_{67} + \sin\phi_{03}\, M_{59} + \sin\phi_{02}\, M_{48} + \cos\alpha(T_{67}+T_{62}+T_{52}) +$$

$$+ 30.0000\, Q_{15} - \sin\phi_5\, M_{73} - \sin\phi_4\, M_{76} - \sin\phi_3\, M_{79} - \sin\phi_2\, M_{82} -$$

$$-(T_{84} + T_{85} + T_{86}) = 0$$

WHERE $M_{74}' = \frac{52.2944}{35.4527} M_{70} = 1.47505\, M_{70}$

OR

$$(M_{74})_{FA} + (M_{74})_A + (M_{75}) + (M_{78}) + (M_{81}) + \cos\alpha(T_{67}+T_{62}+T_{52}) +$$

$$+ (M_{81})_R - (T_{84} + T_{85} + T_{86}) = 0$$

WHERE $(M_{74})_{FA} = -52.2944(Q_{19}+Q_{101}) - \cos\beta\, Q_{102} - \sin\beta\, Q_{103} + M_{74}'$

$(M_{74})_A = \sin\phi_{05}\, M_{72} - \cos\alpha\, (M_{71})_A - \sin\phi_5\, M_{73}$

$(M_{75}) = \sin\phi_{04}\, M_{67} - \sin\phi_4\, M_{76}$

$(M_{78}) = \sin\phi_{03}\, M_{59} - \sin\phi_3\, M_{79}$

$(M_{81}) = \sin\phi_{02}\, M_{48} - \sin\phi_2\, M_{82}$

$(M_{81})_R = 30.0000\, Q_{15}$

THE SUBSCRIPT A IN $(M_{74})_{FA}$, $(M_{74})_A$ & $(M_{71})_A$ ABOVE REFERS TO ONE METHOD (METHOD A) OF HANDLING THE INTERACTION MOMENTS Q_{102} & Q_{103}. IN THIS METHOD THE MOMENTS ARE APPLIED AT LOAD POINT $x = 19$. CONSEQUENTLY THEIR EFFECT ON M_{71} MUST BE SUBTRACTED BEFORE IT IS INCLUDED IN THE EQUILIBRIUM CHECK.

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: C-105	OUTER WING / INNER WING	REPORT NO. 7/0510/1 CHECK; SHEET NO. 7-44 i	
		PREPARED BY: E.N. SHEARLY	DATE: JULY 15/55
		CHECKED BY	DATE

RIB MOMENT CHECK:

$$M_{74} = -1 \times (M_{74})_{F_B} - .349376 \times (M_{74})_B$$

$$M_{75} = \Delta T_F - .633101 \left\{ (M_{74})_{F_B} + (M_{74})_B \right\} - .133101 \times (M_{75}) + .366899 \left\{ (M_{78}) + (M_{81}) + (M_{81})_E \right\}$$

$$M_{78} = \Delta T_R - .287453 \left\{ (M_{74})_{F_B} + (M_{74})_B + (M_{75}) \right\} + .212547 \times \{M_{78}\} + .712547 \left\{ (M_{81}) + (M_{81})_E \right\}$$

$$M_{81} = .211137 \times (M_{81}) + 1 \times (M_{81})_E$$

THE SUBSCRIPT B IN $(M_{74})_{F_B}$, $(M_{74})_B$ & $(M_{71})_B$ REFERS TO ANOTHER METHOD (METHOD B) OF HANDLING THE INTERACTION MOMENTS Q_{102} & Q_{103}. IN THIS METHOD THE MOMENTS ARE <u>NOT</u> APPLIED AS MOMENTS AT LOAD POINT $a=19$ BUT ARE APPLIED AS THE EQUIVALENT LOADS AT $a=19$, 42 & 43. (REF. SHT 5-100) IN THIS METHOD M_{71} MUST INCLUDE THE EFFECT OF Q_{102} & Q_{103}.

THEN $$(M_{74})_{F_B} = -52.2944 \left(Q_{19} + Q_{101} + .027629\, Q_{102} - .011854\, Q_{103} \right) + M_{74}'$$

$$(M_{74})_B = \sin\phi_{05}\, M_{72} - \cos\alpha\, (M_{71})_B - \sin\phi_5\, M_{73}$$

$$(M_{75}) = \sin\phi_{04}\, M_{67} - \sin\phi_4\, M_{76}$$

$$(M_{78}) = \sin\phi_{03}\, M_{54} - \sin\phi_3\, M_{79}$$

$$(M_{81}) = \sin\phi_{02}\, M_{43} - \sin\phi_2\, M_{82}$$

$$(M_{81})_E = 30.0000\, Q_{15}$$

$$\Delta T_F = (T_{84})_c - \cos\alpha\, T_{67}$$

$$\Delta T_R = \cos\alpha\, T_{62} - (T_{86})_c$$

THE SUBSCRIPT C IN $(T_{84})_c$ & $(T_{86})_c$ ABOVE REFERS TO THAT PORTION OF THE TORQUE IN CELLS 84 & 86 RESP. THAT ARE

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT		REPORT NO. 7/05[illegible]/7 CHECK
C-105	OUTER WING, INNER WING	SHEET NO. 7-44 j
		PREPARED BY: R.H. SHEARER — DATE: JULY 5/55
		CHECKED BY: — DATE:

To Balance The Component Perpendicular To The T.J. Rib Of The Torques In Cells 52, 62 & 69.

i	1 h	2 A_s or A_r	3 hA_s or hA_r	4 $\frac{1}{4t}$	5 σ	6 τ	7 M	8 T
	Sht 5-59	Sht 5-57	①×② or 7/05[illegible]/2 4.41	Sht 5.56 5.55 5.57	To Main	To Main	③·⑤	⑥/④
48			23.3640		13.55[illegible]		316.789	
59			47.5466		13.37814		[illegible]	
67			50.5710		14.58445		[illegible]	
72			56.8700		14.38160		[illegible]	
73	7.4312	6.020	56.7452		18.43874		[illegible]	
74	7.3858	5.7035	53.5319		9.60486		514.166	
77	8.3414	5.8480	48.7805		9.13327		445.764	
80	6.4011	4.4141	28.2551		8.32310		235.170	
70			25.1[illegible]		18.55725		467.3838	
71 A			43.925		12.70054		556.601	
71 B			43.825		14.24446		624.263	
52				.0143602		1.02[illegible]		71.8070
62				.0122110		.45[illegible]		37.5719
69				.0118080		.36[illegible]		30.[illegible]
84				.0091406[illegible]		3.35365		367.4402
85				.00990667		3.42732		345.9603
86				.01211667		3.49[illegible]		287.7[illegible]
74	9.4312	10.6038	100.0066		$\bar{3}$.51504		$\overline{3}$57.5277	
75	9.3958	7.7988	75.1041		1.17153		8[illegible]	
78	8.3414	6.7288	56.1276		1.62538		91.2287	
81	6.4011	6.2638	40.[illegible]52		1.52150		61.0045	

$* \ 14.24446 + 1.64401 - 3.18793 = 2.70054$

$M_{74} = 14[illegible] + 467.3838 = 68[illegible].4142$

Total Torque In Cells 52, 62 & 69 = $\overline{T_{52}}$ + [illegible] — [illegible] Sheets Vol. I

= 107.85770 + 30 = 137.85770

Comp. Perp. To T.J. Rib, T_o = cos [illegible] 137.85770 = .7763 × 137.85770 = 107.0184

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT		REPORT No. 7/[illegible] CHECK
C-105	OUTER WING INNER WING	SHEET No. 7-44 1

PREPARED BY	DATE
R.H. SHEARER	July 1952
CHECKED BY	DATE

Re Moment Check (Cont.)

$M_{15} = 15.9168 - .633101 \{ 502.3310 - 408.9411 \} - .133101 \times 329.1162 + $
$+ .366899 \{ 293.8646 + 146.8534 + 30.0000 \} = 85.8315$

$M_{18} = 24.2342 - .287453 \{ 502.3310 - 408.9411 + 329.1162 \} + $
$+ .212547 \times 293.8646 + .712547 \{ 146.8534 + 30.0000 \} = 11.2312$

$M_{2} = .21137 \times 146.8534 + 1 \times 30.0000 = 61.0062$

A. V. RO
TEC
AIRCRAFT
WEIGHT
C. G. POSIT

10	11
413.72	
267.57	
51.52	
42.20	
691.45	
411.51	
267.59	
19.75	
41.73	
—	
69.11	
411.80	
267.59	
—	
92.75	

AVRO AIRCRAFT LIMITED TECHNICAL DEPARTMENT (Aircraft)		REPORT No. 7/0510/7 CHECK SHEET No. 7-45	
AIRCRAFT C105	WING ANALYSIS	PREPARED BY V F GARDENER	DATE MAY 26 '56
		CHECKED BY	DATE

CHECK ON SELF CONTAINED REDUNDANCIES

LOADS OF +1.0 WERE APPLIED AT LOAD POINTS 13, 24, 23 & 21, AND A LOAD OF 2.9333 AT LOAD POINT 20.
THIS LOADING SYSTEM HAS A RESULTANT OF 6.9333 ACTING AT LOAD POINT 22, SO THAT A LOAD OF −6.9333 WAS PLACED AT POINT 22 TO BRING THE SYSTEM INTO EQUILIBRIUM.

THE STRESSES ARISING FROM THE APPLICATION OF THESE LOADS WERE THEN CALCULATED BY MEANS OF THE T.L MATRIX & THE RESULTS USED TO CALCULATE THE MF SYSTEM NECESSARY TO PRODUCE ZERO STRESS EVERYWHERE EXCEPT IN RIB 18.
THE ACTUAL COMPUTATION IS SELF CHECKING IN THAT THERE ARE MORE EQUATIONS OBTAINABLE FROM THE K.p MATRIX THAN UNKNOWNS.

THE VALUES OF THE REDUNDANCIES ARE SHOWN ON p 7-46

Errors

In the following pages an effort is made to determine the number of decimal places lost in the calculation of the elements of the T_{in} matrix through the use of the method of calculation outlined at the beginning of section 5 of this report.

It should be borne in mind that this work results in an estimate only. The results should give a fairly conservative estimate of the "probable" error, but will show a loss of decimal places that is less than the maximum error, due to averaging done in the estimate.

A table similar to that shown on sht. 5-5 appears on sht. , showing the results of this estimate. This table is based on the assumption (not strictly correct for all cases) that the possible error in the basic calculations is 5 in the fifth decimal place.

Error will be a function, primarily, of the size of the matrix, the number of non zero elements in the matrix, and the relative size of the elements of the matrix. This means that it is virtually impossible to determine a general solution for the "maximum" or "probable" error in any given matrix operation. The estimates calculated here apply only to the particular matrices considered, and are based on an averaging process, which means in effect that all non zero elements in each row (or column) are assumed equal, and that the row (or column) sums of the matrix are all equal to the average of the row (or column) sums for that matrix.

Values of the row & column sums used in the calculations are obtainable from the results given by the computing department. (see report 7/0510/7 Vol II — "Computation of Preliminary Matrices")

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

REPORT NO. 7/0510/7
SHEET NO. 7-43

AIRCRAFT		PREPARED BY	DATE
C105	Wing Analysis	[illegible]	4 July 55
		CHECKED BY	DATE

<u>Errors</u>

Consider the matrix operation (multiplication)

$$T_{ia}\, G_{ab} = S_{ib}$$

which may be written

$$[T_{ia} + \Delta T_{ia}][G_{ab} + \Delta G_{ab}] = S_{ib} + \Delta S_{ib}$$

where ΔT_{ia}, ΔG_{ab} & ΔS_{ib} are the error matrices associated with T_{ia}, G_{ab} & S_{ib} respectively.

From this

$$T_{ia} G_{ab} + T_{ia}\, \Delta G_{ab} + \Delta T_{ia}\, G_{ab} + \Delta T_{ia}\, \Delta G_{ab} = S_{ib} + \Delta S_{ib}$$

neglecting the term $\Delta T_{ia}\, \Delta G_{ab}$ as small

$$\Delta_{ib} = T_{ia}\, \Delta G_{ab} + \Delta T_{ia}\, G_{ab}$$

If it be assumed that the error in each element of a given matrix is the same:

$$T_{ia}\, \Delta G_{ab} = |n_i|\, \Delta_G$$

$$\Delta T_{ia}\, G_{ab} = |m_b|\, \Delta_T$$

where $|n_i|$ is the sum of the absolute values of the non-zero elements in the i^{th} row of the T_{ia} matrix, and $|m_b|$ is the sum of the absolute values of the non zero elements in the b^{th} column of the G_{ab} matrix.

Errors

$T_3' \times G_3$

$\Delta T_3' = 5 \times 10^{-7}$ $\qquad \Delta G_3 = 5 \times 10^{-3}$

$|n_i|$ for $T_3' = 0.017$ $\qquad |m_b|$ for $G_3 = 434.0$

$$|n_i|\Delta_G + |m_b|\Delta_T = 0.017 \times 5 \times 10^{-3} + 434 \times 5 \times 10^{-7}$$
$$= 3.02 \times 10^{-4}$$

say 3 in the 4^{th} decimal

$H_1 \times G_1$

$\Delta H_1 = 5 \times 10^{-5}$ $\qquad \Delta G_1 = 5 \times 10^{-5}$

$|n_i|$ for $H_1 = 50.0$ $\qquad |m_b|$ for $G_1 = 99.0$

$$|n_i|\Delta_G + |m_b|\Delta_H = 50 \times 5 \times 10^{-5} + 99 \times 5 \times 10^{-5}$$
$$= 7.45 \times 10^{-3}$$

$T_2' \times H_1 \times G_1$

$\Delta T_2' = 5 \times 10^{-6}$ $\qquad \Delta(H_1 \times G_1) = 7 \times 10^{-3}$ (from above)

$|n_i|$ for $T_2' = 0.42$ $\qquad |m_b|$ for $(H_1 \times G_1) = 219.0$

$$|n_i|\Delta_{H_1 \cdot G_1} + |m_b|\Delta_{T_2'} = .42 \times 7 \times 10^{-3} + 219 \times 5 \times 10^{-6}$$
$$= 4.05 \times 10^{-3}$$

say 4 in the 3^{rd} decimal

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

REPORT NO. 7/0510/7
SHEET NO. 7-53

AIRCRAFT:		PREPARED BY	DATE
C105	Wing Analysis	D L Turner	4 July '55
		CHECKED BY	DATE

Errors

$$H_2 \times G_2$$

$\Delta H_2 = 5 \times 10^{-3}$ $\Delta G_2 = 5 \times 10^{-4}$

$|n_l|$ for $H_2 = 120.0$ $|m_b|$ for $G_2 = 71.0$

$$|n_l| \Delta_{G_2} + |m_b| \Delta H_2 = 120 \times 5 \times 10^{-4} + 71 \times 5 \times 10^{-3}$$
$$= 4 \times 10^{-1}$$

$$T_3' \times H_2 \times G_2$$

$\Delta T_3' = 5 \times 10^{-7}$ $\Delta(H_2 \times G_2) = 4 \times 10^{-1}$ (from above)

$|n_l|$ for $T_3' = 0.0035$ $|m_b|$ for $H_2 \times G_2 = 164.0$

$$|n_l| \Delta_{H_2 \times G_2} + |m_b| \Delta_{T_3'} = .0035 \times 4 \times 10^{-1} + 164 \times 5 \times 10^{-7}$$
$$= 1.48 \times 10^{-3}$$

say 2 in the third decimal

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

REPORT NO. 7/0510/7
SHEET NO. 7-59

AIRCRAFT:		PREPARED BY	DATE
C105	Wing Analysis	D L Tu...	4 July 55
		CHECKED BY	DATE

Errors.

$$H_2 \times (H \times G_1)$$

$\Delta H_2 = 5 \times 10^{-3}$ $\quad\quad \Delta(H_1 \times G) = 7 \times 10^{-3}$ (sht.

$|n_1|$ for $H_2 = 53.0$ $\quad\quad |m_b|$ for $H_1 \times G_1 = 219.0$

$|n_1| \Delta_{H_1 \times G_1} + |m_b| \Delta H_2 = 53 \times 7 \times 10^{-3} + 219 \times 5 \times 10^{-3}$
$= 1.47$

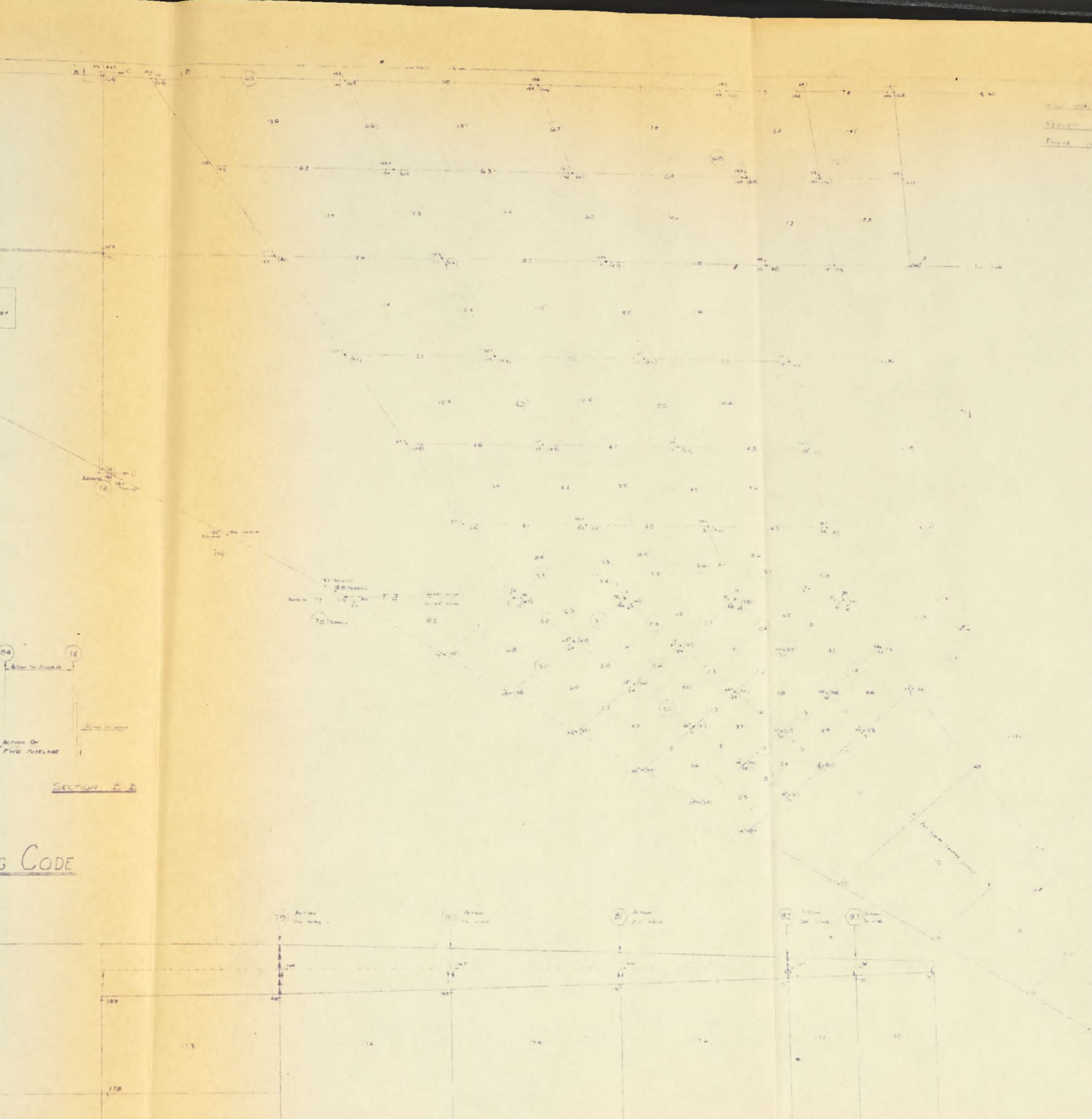
$$T_3' \times (H_2 \times H_1 \times G_1)$$

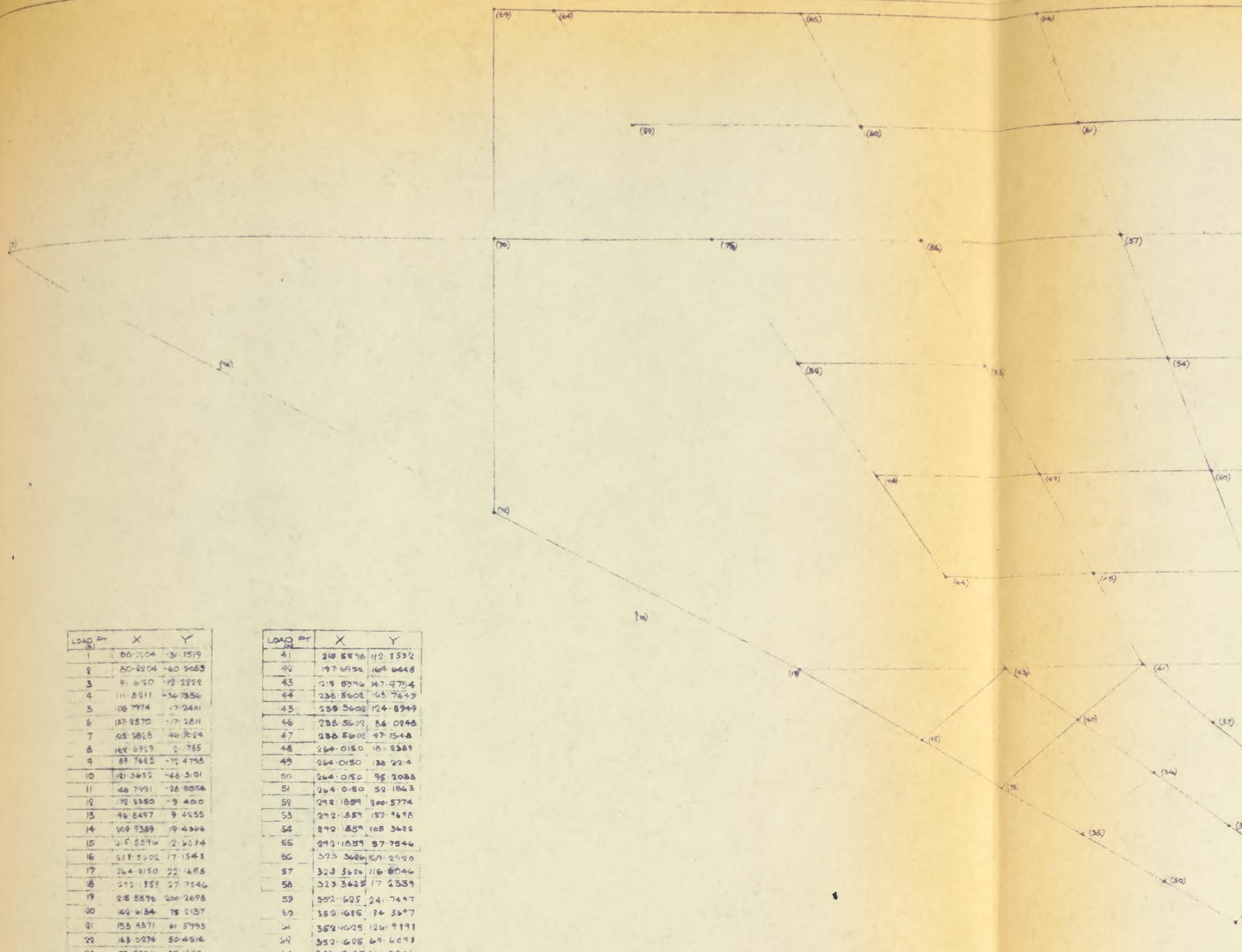
$\Delta T_3' = 5 \times 10^{-7}$ $\quad\quad \Delta(H_2 \times H_1 \times G_1) = 1.5$ (see above)

$|n_1|$ for $T_3' = 0.0034$ $\quad\quad |m_b|$ for $H_2 \times H_1 \times G_1 = 228.0$

$|n_1| \Delta_{H_2 H_1 G_1} + |m_b| \Delta_{T_3'} = .0034 \times 1.5 + 228 \times 5 \times 10^{-7}$
$= 5.32 \times 10^{-3}$

say 5 in the 3rd decimal


ACTION ON FWD FUSELAGE
SECTION B-B
G CODE
SECTION A-A



LOAD PT (R)	X	Y
1	80·2204	-31·7519
2	80·2204	-60·9053
3	91·6120	-12·2222
4	111·8211	-36·7356
5	108·7974	17·2401
6	137·2570	-17·2811
7	125·9825	46·7084
8	168·6929	2·1785
9	89·7625	-72·4798
10	181·3632	-48·3101
11	146·7991	-28·8856
12	172·2350	-9·4010
13	96·8497	9·4255
14	209·9389	19·4366
15	215·5596	12·6074
16	228·5602	17·1548
17	264·0150	22·1685
18	292·1859	27·7546
19	215·5596	200·2698
20	149·6134	75·2137
21	153·4371	61·5993
22	163·5274	50·4516
23	173·9952	37·1477
24	187·3076	21·0000
25	151·4569	90·3747
26	163·7464	75·4671
27	173·8101	63·2605
28	185·5199	48·6728
29	200·3968	31·0[illegible]
30	161·7015	107·9379
31	175·2265	91·5321
32	186·3011	78·0985
33	199·5176	62·0670
34	215·5596	42·6034
35	173·6993	128·5068
36	188·2708	110·3465
37	200·9298	95·4764
38	215·5596	77·7838
39	185·6172	149·0755
40	202·1152	129·5[illegible]

LOAD PT (R)	X	Y
41	218·5596	112·1532
42	197·6932	169·6448
43	215·5596	147·9754
44	238·5602	165·7649
45	238·5602	124·8949
46	238·5602	86·0848
47	238·5602	47·1548
48	264·0150	181·2389
49	264·0150	138·2214
50	264·0150	95·2088
51	264·0150	52·1863
52	292·1859	200·5774
53	292·1859	152·9698
54	292·1859	105·3628
55	292·1859	57·7546
56	323·3626	169·2920
57	323·3626	116·8046
58	323·3625	17·2339
59	352·1625	241·7497
60	352·1625	34·3697
61	352·1625	126·9193
62	352·1625	69·6093
64	380·9625	261·5201
65	380·9625	199·4476
66	380·9625	37·3751
67	380·9625	75·3026
68	380·9625	86·3038
69	380·9625	276·5201
70	323·3625	276·5201
71	323·3625	409·2691
72	254·1909	276·5201
73	229·4937	241·1425
74	289·1367	342·8946
75	323·3626	221·9394
76	323·3626	63·9171

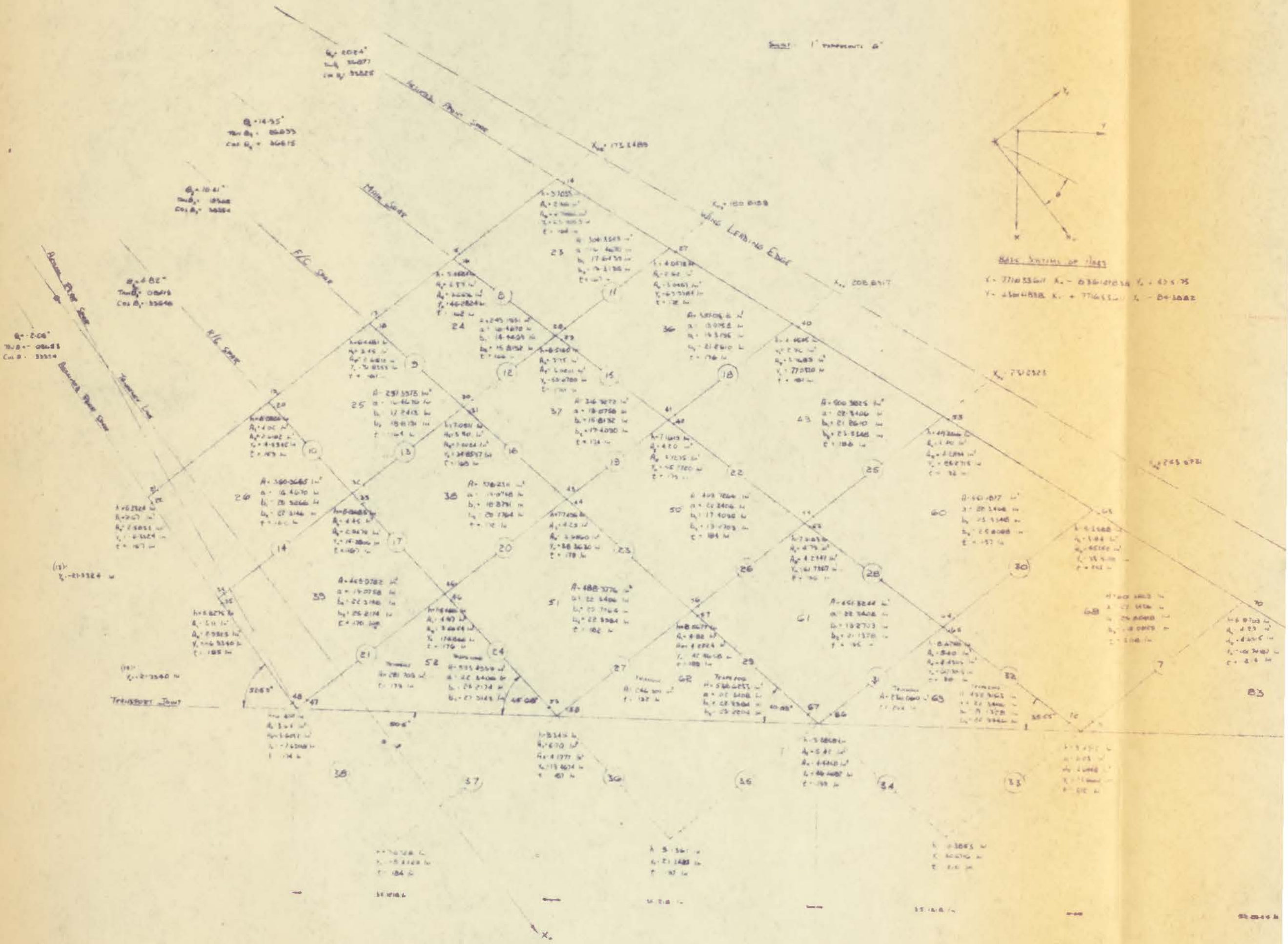
WING COORDINATE SYSTEM

SCALE 1/10

THESE LINES INT
ORIGIN OF THE COORD
ORIGIN IS LOCATE
AND 380·9625 IN

OUTER WING ASSUMED STRUCTURE

SECTION PROPERTIES



Report No. 7/0510/1 Sheet 8-7

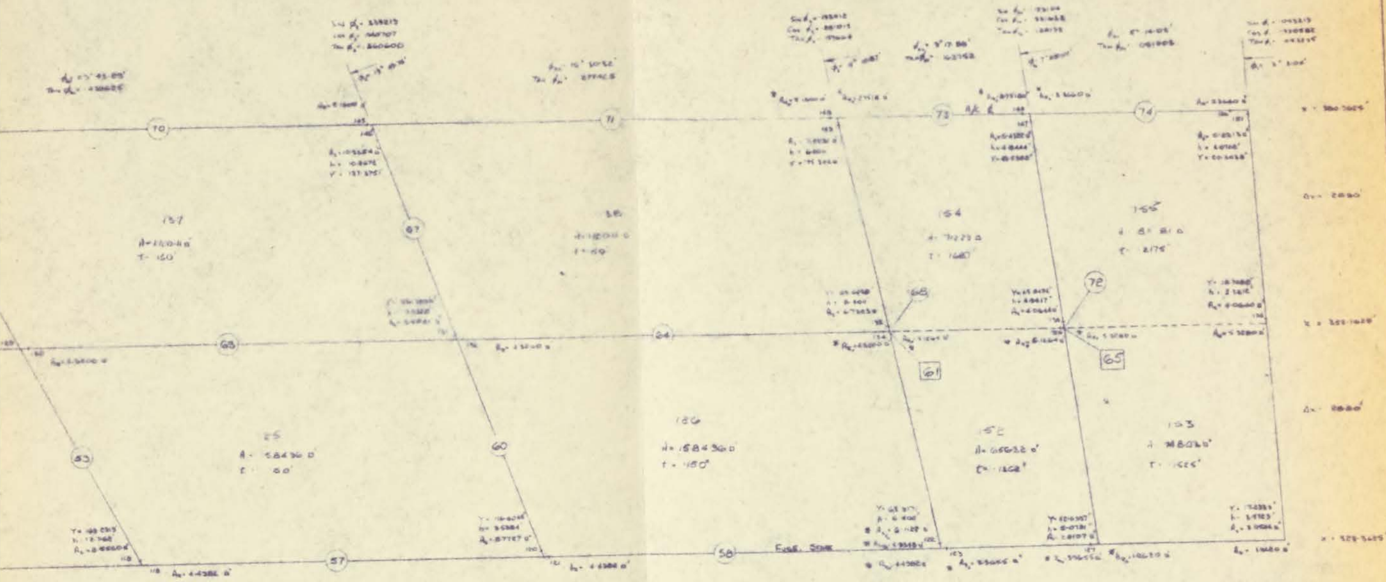
Figure 7

VF Gardiner
Jan 18 65

SECTION PROPERTIES

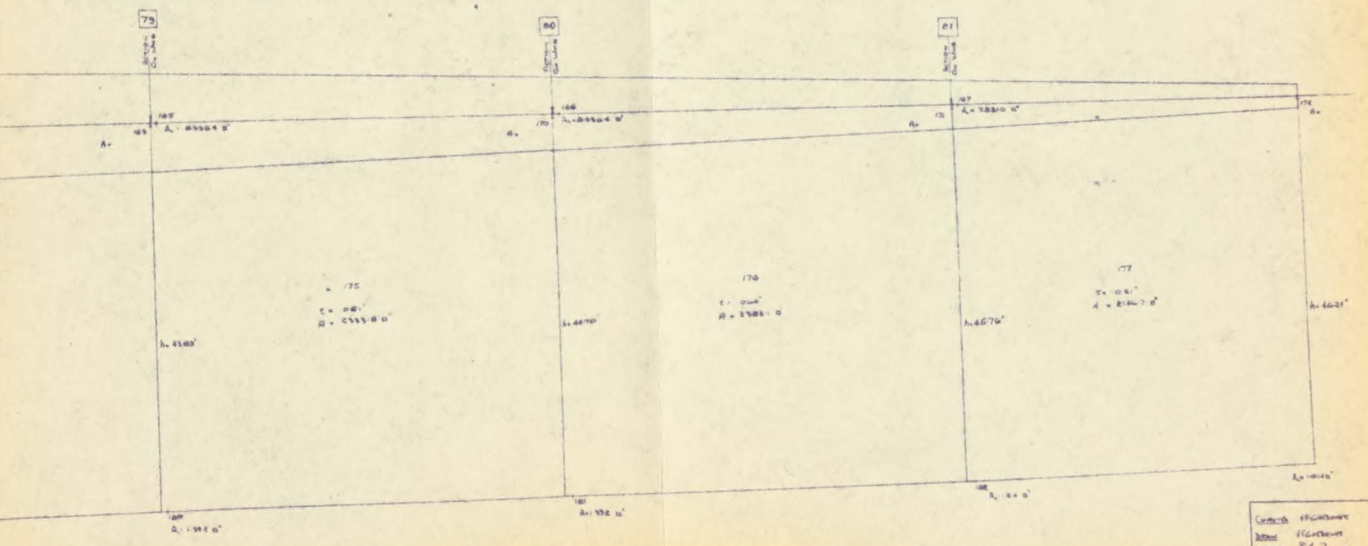
IDEALIZED STRUCTURE - CENTRE SECTION

PRELIMINARY AREAS & THICKNESSES

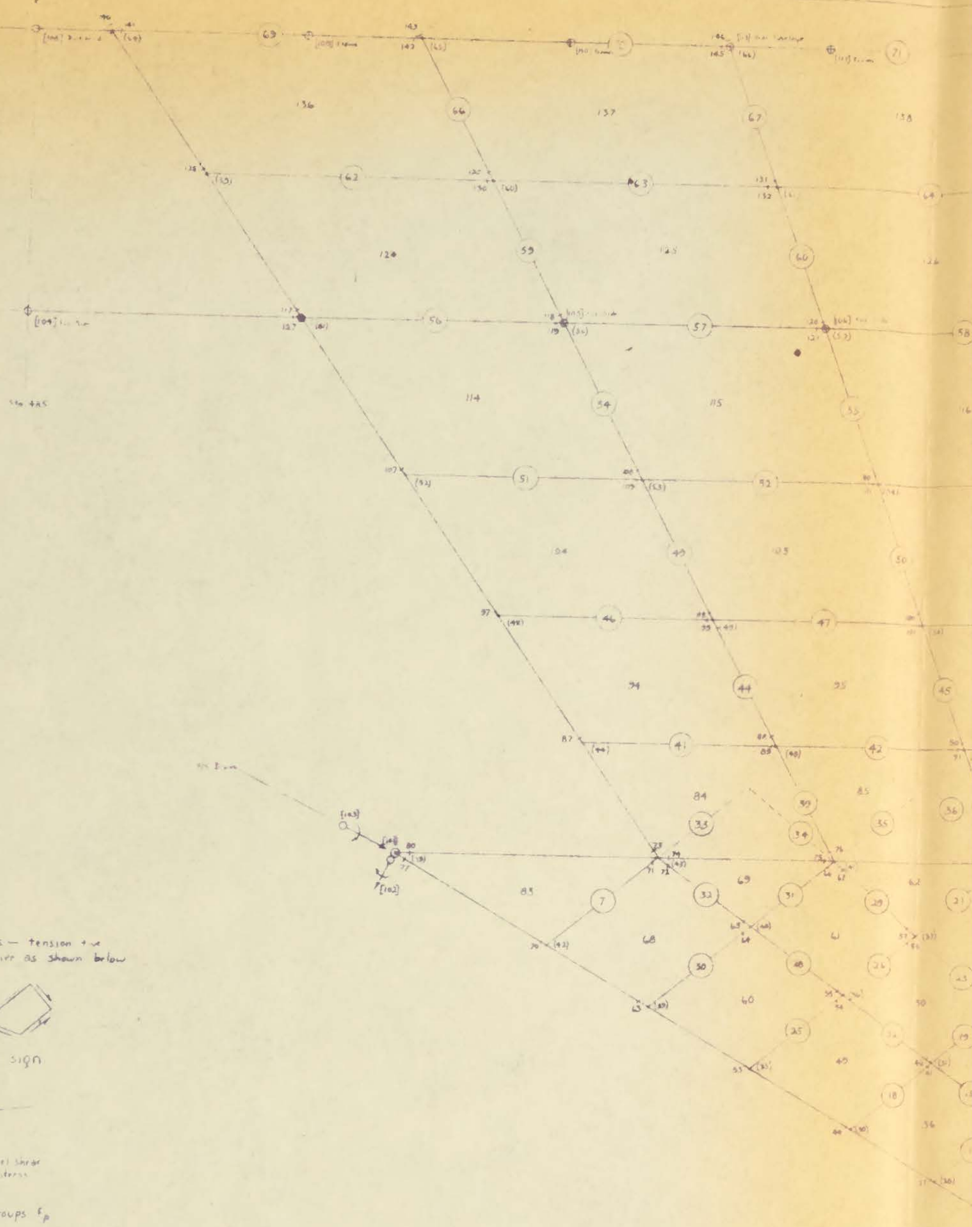


* Min Areas were used at these points for stress calculation

IDEALIZED STRUCTURE - FUSELAGE SIDE BEAM



Fwd

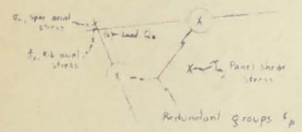


<u>Sign Conventions</u>

1. Down loads positive
2. On upper surface
   Direct stress in rib & spar caps — tension +ve
   Shear applied to panels positive as shown below

On bottom surface — opposite sign

3. Coding



Exceptions to this coding occur at the transport joint, the wing tip & in the case of some interaction loads. Interpretation is self evident

<u>Notes</u>

i,k	Stress indices	6 – 149 except 135, 139 & 144
p,q	Redundant group indices	(7) – (71) except 61, 66 & 68
a,b	Load indices	(1) – (67) except 58 & 63 Also (81) & (82) at main reactions & [101] – [114] interaction loads

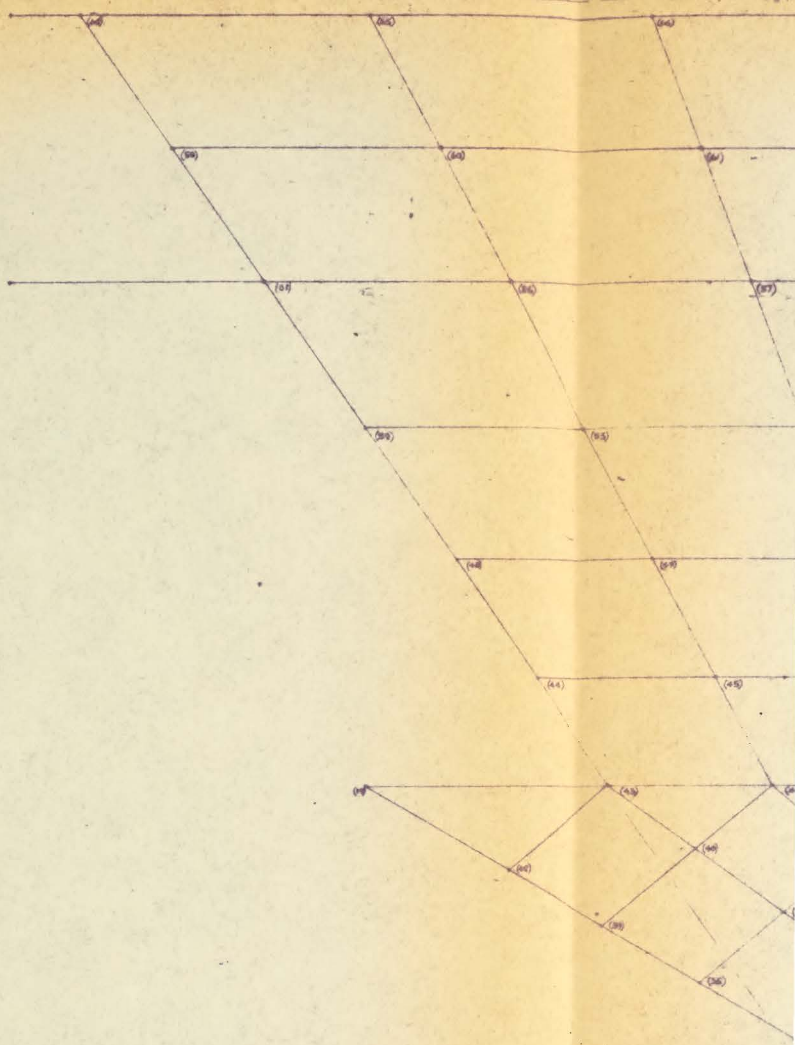
these latter include externally applied loads, designated by •, numbered (a)
interaction loads, designated by ○, numbered [a]

Main reactions are designated by ●

Outer wing and wing tip co-ordinate axes X_o, Y_o, θ

Inner wing & centre section co-ordinate axes X_i, Y_i, ϕ

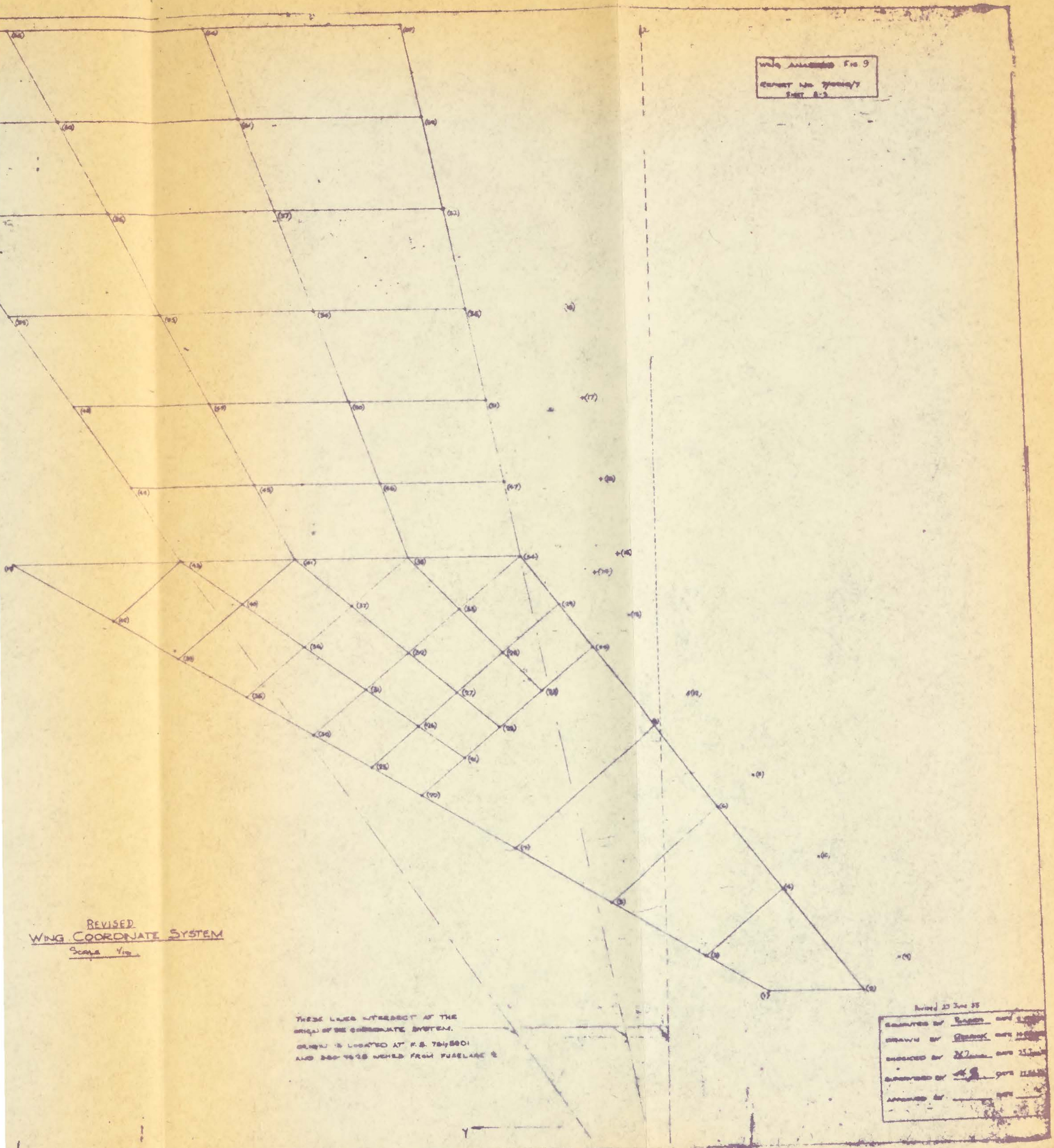
Structure & loading symmetric about ℄ aircraft



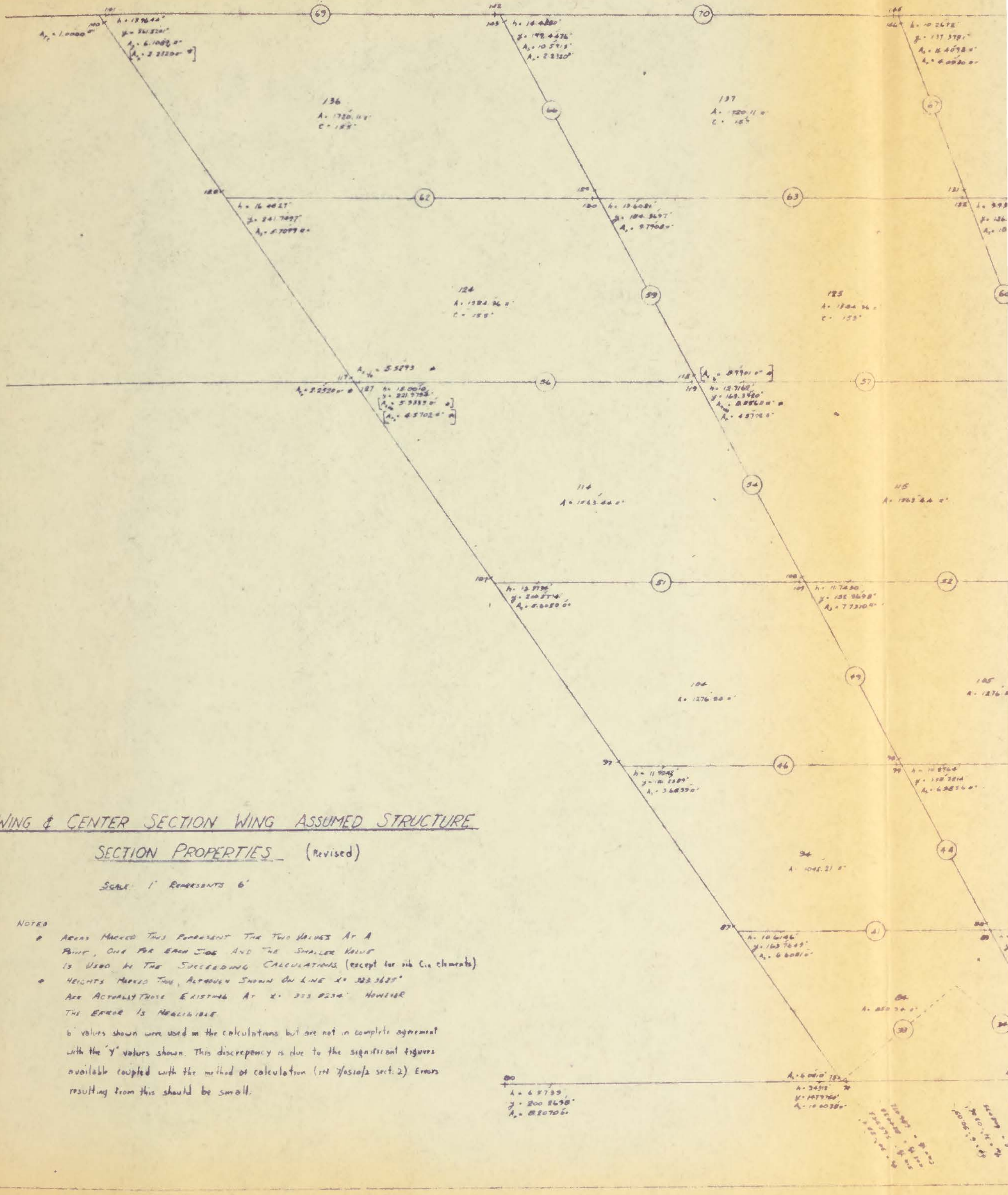
LOAD PT (N)	X	Y
1	80·2704	-31·7519
2	80·2204	-60·9058
3	91·6120	-12·2222
4	111·8211	-36·7356
5	108·7974	17·2401
6	137·2570	-17·2811
7	125·9828	46·7024
8	162·6989	-2·1735
9	89·7683	-72·4798
10	121·3692	-48·3101
11	146·7991	-28·8556
12	172·2350	-9·4010
13	196·8497	9·4255
14	209·9389	19·4366
15	215·5596	12·6074
16	238·5602	17·1541
17	264·0150	22·1683
18	292·1859	27·7546
19	215·5596	200·2698
20	142·6134	78·2137
21	153·8371	61·5998
22	163·0274	50·4516
23	175·9952	37·1477
24	187·3076	21·0000
25	151·4569	90·3747
26	163·7468	78·4671
27	173·8101	63·2605
28	185·8199	48·6928
29	200·3768	31·0111
30	161·7018	107·9379
31	175·2268	91·5321
32	186·3011	78·0988
33	195·5176	69·0670
34	215·5596	42·6084
35	173·6993	128·5068
36	188·6708	110·3468
37	200·9288	95·4764
38	215·5896	77·7808
39	185·6972	149·0789
40	202·1162	129·1608

LOAD PT (N)	X	Y
41	215·5896	112·8532
42	197·6952	169·6448
43	215·5596	147·9754
44	238·5602	163·7649
45	238·5602	124·8949
46	238·5602	86·0248
47	238·5602	47·1548
48	264·0150	181·2389
49	264·0150	138·2214
50	264·0150	95·2038
51	264·0150	52·1863
52	292·1859	200·5776
53	292·1859	152·9698
54	292·1859	105·3622
55	292·1859	57·7546
56	323·3626	169·2920
57	323·3626	116·6046
59	352·1625	241·7497
60	352·1625	184·3697
61	352·1625	126·9898
62	352·1625	69·6098
64	380·9625	261·5201
65	380·9625	199·4476
66	380·9625	137·3751
67	380·9625	75·3026
81	323.3626	221.9794
82	323.3626	63.9171

REVISED
WING COORDINATE SYSTEM
Scale 1/10



REVISED
WING COORDINATE SYSTEM
Scale 1/10



INNER WING & CENTER SECTION WING ASSUMED STRUCTURE

SECTION PROPERTIES (Revised)

SCALE 1" REPRESENTS 6'

NOTES

* AREAS MARKED THUS REPRESENT THE TWO VALUES AT A POINT, ONE FOR EACH SIDE AND THE SMALLER VALUE IS USED IN THE SUCCEEDING CALCULATIONS (except for rib C.a elements)
* HEIGHTS MARKED THUS, ALTHOUGH SHOWN ON LINE X = 323.3635" ARE ACTUALLY THOSE EXISTING AT X = 323.8234" HOWEVER THE ERROR IS NEGLIGIBLE

b' values shown were used in the calculations but are not in complete agreement with the "Y" values shown. This discrepency is due to the significant figures available coupled with the method of calculation (ref 7/0510/2 sect. 2) Errors resulting from this should be small.

Ref 7/0510/7
C_{10} Zone 3
71 - 149
71 - 149

Inner Wing

C_{10} Elastic Influence Matrix (Symmetric) Zone 3

Circled Row & Column Numbers Are For I.B.M. Use Only.

485 Non Zero Elements

Coefficients in the body of the table for use with Inch Kip Units (Multiply by 10^{-3} for use with Inch Lb. Units)

Columns: 73 74 75 76 77 78 79 80 81 82 83 84 85 86 87 88 89 90 91 92 93 94 95 96 97 98 99 100 101 102 103 104 105 106 107 108 109 110 111 112 113 114

Rows: 73 – 149

OUTER WING.

REPORT No. 7/0610/7
SHEET 8-16

Computed [illegible]
Tabulated [illegible]
Checked [illegible]
Supervised [illegible] July 1st 55
Approved

$K_{..}$ STRESS TO REDUNDANT MOMENT [ZONE 1]

CIRCLED ROW & COLUMN NUMBERS ARE FOR I.B.M. USE ONLY.

104 NON ZERO ELEMENTS

OUTER WING

T.. STATICALLY DETERMINATE STRESS -- LOAD MATRIX (ZONE 1)

Wing Analysis
Report No. [illegible]
Sheet [illegible]

CIRCLED ROW & COLUMN NUMBERS FOR I.B.M. USE ONLY.

Computed [illegible]
Tabulated V. F. Gardener
Checked [illegible]
Supervisor [illegible]
Approved

327 NON ZERO COEFFICIENTS.

(IBM)	Row	1	2	3	4	5	6	7	8	9	10	11	12	13	14
1	6	.17831	1.04858							4.07903					
2	7	1.04734	3.46256							3.46256					
3	8	.36424	.48747	.82872	.61626					1.27542	1.62240				
4	9	2.06906	2.78197	1.24064	1.24064					2.78157	1.24064				
5	10	.25117	.21655	.36860	.37225	.54461	.69722			.56484	.72220	.64717			
6	11	1.17798	2.16454	1.55504	1.55504	.66752	.66752			2.16454	1.55504	.66752			
7	12	.13822	.11608	.19880	.20041	.23422	.26769	.38984	.33496	.30622	.38881	.45800	.52538		
8	13	1.40477	1.72176	1.21368	1.21368	.80472	.80472	.39536	.39376	1.72176	1.21368	.80472	.39578		
9	14	.73168	.87188	.61444	.61444	.40740	.40740	.20036	.20038	.87160	.61444	.40760	.20036		
10	15	.16724	.09372	.22750	.19017	.32281	.24781	.41652	.34545	.29176	.38825	.46687	.54551		
11	16	1.28253	1.48708	1.04825	1.04825	.69503	.69503	.34188	.34187	1.48708	1.04825	.69503	.34198		
12	17	.11207	.06890	.16369	.12743	.21638	.17946	.27011	.25149	.19651	.26015	.31218	.36421		
13	18	1.45821	1.66777	1.17562	1.17567	.77948	.77948	.38335	.38336	1.66777	1.17562	.77948	.38655		
14	19	.06120	.03450	.08591	.08859	.11617	.09801	.15243	.12642	.10677	.14207	.17049	.18890		
15	20	1.45835	1.66791	1.17572	1.17572	.77955	.77955	.38338	.38338	1.66791	1.17672	.77955	.38350		
16	21													1.08806	
17	22	1.22782	1.42380	1.00365	1.00365	.66546	.66546	.32727	.32727	1.42380	1.00365	.66546	.32727		
18	23														
19	24	.10864	.06088	.14865	.12363	.20975	.17596	.27056	.22440	.18967	.25219	.30262	.35305		
20	25	.10997	.06182	.16077	.12904	.21232	.17605	.27387	.22714	.19184	.25527	.30632	.35737		
21	26	.11135	.06239	.15286	.12661	.21499	.17830	.27731	.23000	.19428	.25848	.31017	.36186	.34820	
22	27	.70667	.80546	.59366	.59366	.42316	.42316	.25271	.25271	.80546	.59366	.42318	.25271		
23	28														
24	29	1.20320	1.57140	1.01079	1.01079	.72053	.72053	.45026	.45026	1.57140	1.01079	.72053	.45026		
25	30														
26	31	1.60334	1.57385	1.14632	1.14632	.81643	.81643	.48753	.48755	1.55303	1.14558	.81643	.48753		
27	32														
28	33	1.45701	1.54872	1.14001	1.14001	.81264	.81264	.48527	.48527	1.54872	1.14001	.81264	.48527		
29	34													.47180	
30	35	1.10128	1.25574	.92517	.92517	.65949	.65949	.39382	.39382	1.25574	.92517	.65949	.39402	.50917	
31	36														
32	37	.08612	.04826	.11807	.09792	.16627	.13790	.21447	.17788	.15024	.19991	.25985	.27988		
33	38	.08714	.04885	.11948	.09909	.16825	.13954	.21703	.18000	.15203	.20229	.24274	.28320		
34	39	.08821	.04945	.12093	.10030	.17031	.14125	.21968	.18220	.15388	.20476	.24571	.28666	.27585	.27585
35	40	.65049	.36022	.55972	.55972	.42248	.42248	.28524	.28524	.13022	.55972	.42248	.28524		
36	41														
37	42	1.11796	1.25461	.56166	.56166	.72587	.72587	.49008	.49008	1.25461	.56166	.72587	.49008		
38	43														
39	44	1.26868	1.42365	1.09124	1.09124	.82368	.82368	.55611	.55611	1.42365	1.09124	.82368	.55611		
40	45														
41	46	1.26401	1.41940	1.08798	1.08798	.82122	.82122	.55445	.55445	1.41940	1.08798	.82122	.55445		
42	47														
43	48	.57715	1.09658	.84054	.84054	.63445	.63445	.42856	.42835	1.09658	.84054	.63445	.42835	1.62227	.81700
44	49														
45	50	.06494	.03751	.09176	.07612	.12925	.10719	.16672	.13827	.11678	.15539	.18647	.21754		
46	51	.06770	.03794	.09282	.07698	.13071	.10841	.16861	.13984	.11811	.15716	.18859	.22001		
47	52	.06889	.03860	.09445	.07834	.13301	.11032	.17157	.14230	.12018	.15992	.19190	.22388	.21540	.21540
48	53	.58161	.65021	.51600	.51600	.40798	.40798	.29995	.29995	.65021	.51600	.40798	.29996		
49	54														
50	55	1.01715	1.12552	.89380	.89380	.70620	.70620	.51921	.51921	1.12552	.89380	.70620	.51921		
51	56														
52	57	1.16074	1.28440	1.01929	1.01929	.80590	.80590	.59250	.59250	1.28440	1.01929	.80590	.59250		
53	58														
54	59	1.14469	1.26664	1.00519	1.00519	.79475	.79475	.58431	.58431	1.26664	1.00519	.79425	.58431		
55	60														
56	61	.05210	.02919	.07163	.05924	.10058	.08342	.12974	.10761	.09086	.12093	.14512	.16930		
57	62	.05293	.02966	.07257	.06019	.10220	.08476	.13183	.10933	.09234	.12287	.14745	.17202		
58	63	.53971	.58679	.47786	.47786	.35057	.35057	.30329	.30329	.58679	.47786	.35057	.30329		
59	64														
60	65	.92528	1.01700	.82850	.82890	.67750	.67790	.52609	.52609	1.01700	.82890	.67750	.52609		
61	66														
62	67	1.06525	1.16363	.94841	.94841	.77518	.77518	.60194	.60194	1.16363	.94841	.77518	.60194		
63	68														
64	69	.04177	.02341	.05727	.04750	.08065	.06683	.10404	.08628	.07288	.09697	.11636	.13578		
65	70	.49119	.53874	.44348	.44348	.37164	.37164	.29980	.29980	.53274	.44348	.37164	.29980		
66	71	.25757	.27682	.23585	.23585	.20318	.20318	.17047	.17047	.27651	.23589	.20318	.17047		
67	72	.85285	.88480	.76584	.76486	.64514	.64514	.52042	.52042	.92480	.76986	.64514	.52042		
Sum		[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
See Spar Sum															
See Zone 2															

WING ANALYSIS
REPORT No. [illegible]
SHEET 8-8

ROW & COLUMN NUMBERS FOR I.B.M. USE ONLY.

COMPUTED [illegible] Feb 55
TABULATED V F GARDENER June 55
CHECKED [illegible] July 55
SUPERVISED [illegible] July 7 55
APPROVED

31 32 33 34 35 36 37 38 39 40 41 42 43 44 45 46 47 48 49 50 51 52 53 54 55 56 57 58 59 60 61 62 63 64 65 66 67 68 69 70 71 72 73 74 75 76 77

51 52 53 54 55 56 57 58 59 60 61 62 63 64 65 66 67 68 69 70 71 72 73 74 75 76 77 78 79 80 81 82 83 84 85 86 87 101 102 103 104 105 106 107 108 109 110 111 112

.55065
.27154
.16807
.15342
.83503
.44653
.95078
.50753
.81521
.56836
.07116
.60660
.64075
.64401 .3.18795

.83508 .47184 — 1.54086 1.30683 .44653 — 2.01653 1.06660 — .64075

.64401 3.18795

INNER & CENTRE SECTION WING

CIRCLED ROW & COLUMN NUMBERS FOR I.B.M. USE ONLY.

REPORT No
SHEET
WING ANALYSIS

T.. STATICALLY DETERMINATE STRESS – LOAD MATRIX [ZONE 2]

2070 NON ZERO ELEMENTS

Circled Row & Column Numbers For I.B.M. Use Only.

Report No. 7/0510/7
Sheet B-19
Wing Analysis

Computer

Tabulated V.K. Anderson

Checked

Supervised

Approved

Matrix [Zone 2]

2070 Non Zero Elements

(41) 41	(42) 42	(43) 43	(44) 44	(45) 45	(46) 46	(47) 47	(48) 48	(49) 49	(50) 50	(51) 51	(52) 52	(53) 53	(54) 54	(55) 55	(56) 56	(57) 57	58	(58) 59	(59) 60	(60) 61	(61) 62	63	(62) 64	(63) 65	(64) 66	(65) 67	(66) 101	(67) 102	(68) 103	(69) 104	(70) 105	(71) 106	(72) 107	(73) 108	(74) 109	(75) 110	(76) 111	(77) 112	(78) 113	(79) 114	B1	BC			Total Check Sum

CIRCLED ROW & COLUMN NUMBERS FOR I.B.M. USE ONLY.

REPORT No. 7/0710/7
SHEET 8-19
WING ANALYSIS.

CALCULATED [illegible]
TABULATED V.R.[illegible]
CHECKED [illegible] July 55
[illegible]
REFERENCE

2070 NON ZERO ELEMENTS

47	48	49	50	51	52	53	54	55	56	57	58	59	60	61	62	63	64	65	66	67	66 / 101	67 / 102	68 / 103	69 / 104	70 / 105	71 / 106	72 / 107	73 / 108	74 / 109	75 / 110	76 / 111	77 / 112	78 / 113	79 / 114	81	82		TOTAL CHECK SUM