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DESIGN REQUIREMENTS FOR AEROPLANES *for the* ROYAL AIR FORCE

This handbook is issued for the information and guidance of all concerned.

By Command of the Air Council

C. Ll. Bullock

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The amendments promulgated in the undermentioned amendment lists have been made in this publication.

Amendment List.		*Amendments made by*	*Date.*
Number.	*Date.*		
1	9–35	Incorporated in reprint.	—
2		J.D.B.	1-10-38
3	10-38	J.D.B.	29-8-39

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CHAPTER III.—STRENGTH REQUIREMENTS BASED ON OTHER THAN FLIGHT CONDITIONS—*contd.*

CHAPTER IV.—NON-FACTOR REQUIREMENTS

CHAPTER IV—NON-FACTOR REQUIREMENTS—*contd.* *Para. No.*

INTRODUCTION—SCOPE OF HANDBOOK AND ARRANGEMENT

1. Scope of the handbook

The requirements given in this handbook apply to all aeroplanes built to Air Ministry contract. The handbook is intended to amplify the requirements given in aeroplane specifications. In the event of any conflict between the requirements of an aeroplane specification and those of this handbook, the requirements given in the aeroplane specification take precedence.

2. Arrangement

(i) All requirements affecting the airframe are collected into three chapters.—

Chapter II, Strength requirements based on flight conditions,

Chapter III, Strength requirements based on other than flight conditions,

Chapter IV, Non-factor requirements.

Chapters V to VIII inclusive of the handbook consist of methods of calculation and other details needed for the precise interpretation of the requirements of Chapters II, III and IV.

(ii) The non-factor requirements of Chapter IV comprise such items as flutter prevention, welding regulations, etc. They are characterized by the fact that they do not admit of specification in terms of a factor.

(iii) Chapter IX consists of airscrew requirements.

3.

Reference to any particular portion of the handbook should be made by quoting the chapter (Section for Chapters V, VIII and IX), paragraph and line number.

4. Indications of amendments

At the top of each page affected by amendment action will be found the number of the Amendment List concerned. A black line printed on the left-hand side of the text or illustrations indicates that the matter against the line has been amended or added by the Amendment List quoted. This indication on any particular page will appear only against the matter on that page affected by that Amendment List.

CHAPTER I.—GENERAL REQUIREMENTS GOVERNING THE ESTIMATION OF STRUCTURAL STRENGTH

1. Methods of strength estimation : calculation and test

(i) Any technically sound method of estimating the strength of airframes under the specified externally applied loads is acceptable. Credit may be taken for all redundancies provided sufficient information is available as to the effect of such redundancies. Compliance with strength requirements will usually be based upon calculated strength rather than upon the strength as determined by mechanical tests on a complete component. Allowable stresses to be used in such calculations are given in Chapter VIII.

(ii) When the type of construction is not amenable to strength calculation or when there is reason to doubt the accuracy of such calculations as can be made, the strength will be determined by *ad hoc* mechanical tests. Prior official concurrence should be obtained for such tests and they should be carried out under approved conditions.

2. Proof factor and ultimate factor

All calculations and mechanical tests are to be made in the light of the following requirements.

(i) Any standard structure or component shall not collapse before withstanding on strength test the external loads corresponding to the *specified ultimate factor*.

(ii) Any standard structure or component shall be capable on strength test of carrying for a period of one minute 75 per cent. of the loads corresponding to the specified ultimate factor, during and after which it shall still be in an airworthy condition. This 75 per cent. of the specified ultimate factor will be referred to as the *specified proof factor*.

The factors given in this handbook and in aeroplane specifications are specified ultimate factors unless otherwise stated. Compliance with the proof factor requirement should be checked both when approval is based entirely upon calculations and when recourse is had to mechanical tests.

3. Definition of standard and typical structures and components

(i) A *standard component* is the weakest component that could be made complying with the drawings and material specifications, all limits and tolerances being taken in the most adverse direction. Standard compression members, in addition to satisfying these conditions, are to be regarded as having the maximum allowable eccentricity.

(ii) A *typical component* is a component constructed in accordance with usual workshop procedure.

(iii) *Standard* and *typical structures* are structures built throughout of *standard* or *typical components* respectively.

(iv) *Typical*, not *standard*, structures and components will usually be available for mechanical tests, and hence the test results will have to be corrected down to standard structure conditions. Such correction will usually only be possible when the item tested is of simple design and fails in a manner to which the specified material properties are directly applicable. In other cases it will be necessary to obtain on test factors 20 per cent. greater than those specified. In the case of a test on a complete unit a convenient procedure, when practicable, is to patch up in an approved manner such members as fail prematurely in order to continue the test. Corrections to standard component conditions need then be applied only to the members which fail before the full 20 per cent. extra load has been applied.

(v) When correction down to standard component conditions is possible and reasonably easy to apply it is not permissible to waive such correction in favour of compliance with the 20 per cent. expedient. Doubtful cases should be referred to the Airworthiness Department, Royal Aircraft Establishment, South Farnborough.

4. Critical loading cases for particular components

(i) The majority of the strength requirements given in the succeeding chapters are stated without specific reference to the particular components for which they may be expected to give critical loads. Unless otherwise stated the loads corresponding to the various conditions specified should be traced through the structure sufficiently far to ensure that the aeroplane has at least the specified factors throughout the whole structure, but this does not imply that the whole structure need be stressed for every specified condition. Many of the stressing cases overlap and when it can be shown that any particular case will not give critical loads it will be unnecessary to consider that case further.

(ii) Tables are given at the beginning of Chapters II and III summarizing the requirements specified in each of these chapters and indicating the particular components for which each requirement may be expected to give design loads. It will, however, always be necessary unless otherwise stated to check that the aeroplane *as a whole* complies with all the specified requirements.

(iii) Most of the requirements of Chapter IV apply to the whole aerostructure, so that the table at the beginning of Chapter IV does not indicate the components of the aerostructure relevant to each requirement.

(iv) Table I which follows is in effect a re-arrangement of the Chapter II and III Tables, together with a few items from the Chapter IV Table, the various requirements being grouped to show which will normally need to be considered for each of the main components of the aerostructure. This list is not to be taken as over-riding the proviso that the aeroplane as a whole must comply with all the specified requirements unless otherwise stated.

TABLE I

Component.	*Relevant requirements.*	*For particulars see* Chap.	Para.
Main planes	Normal flight, C.P. forward	II	2
	Strength requirements for wings with tip slots (super-stall)	II	13
	Normal flight, C.P. back	II	3
	Terminal velocity dive	II	4
	Fast glide (seldom critical for main planes)	II	5 and 6
	Aileron wing loads	II	8
	Up and down gusts	II	9
	Inverted flight, high negative incidence	II	10
	Strength of aeroplanes under automatic control	II	21
	Catapulting	III	10
	Catapulting	V	Sect. IV.
	Landing	III	4–7
	Engine mounting (when engines are in the wings)	II	18
	Wires cut	II	23
	Static thrust and torque	III	17
	Aileron mass-balance	IV	2
	Wings folded	III	16
	Jacking loads	III	15
	Relative strength of lift and anti-lift wires	II	25
	Duplicate wires	II	24
	Wing tip float (side loads)	III	6 and 7
	Rib removed	III	22
	Rib tests	III	24
	Aerodynamic loading on long struts	II	26
	Trailing edge flaps	II	12

TABLE I—*continued*

Component.	*Relevant requirements.*	*For particulars see* Chap.	Para.
Centre section	As for main plaines. Also—		
	One wheel landing	III	4
	Side load (at axle)	III	4
	Somersault landing	III	13
	Slinging	III	12
	Salvage	III	14
Front fuselage	Normal flight, C.P. forward	II	2
	Engine mounting (if engine is in front fuselage)	II	18
	Static thrust and torque	III	17
	Side load	II	19
	Jacking loads	III	15
	Safety belt and harness loads	III	20
Rear fuselage	Terminal velocity dive	II	4
	Over-riding minimum tail load	II	7
	Fast glide	II	5 and 6
	Normal flight, C.P. forward	II	2
	Normal flight, C.P. back	II	3
	Fin and rudder loads	II	16
	Fin and rudder loads	V	Sect. II.
	Over-riding torsional loading from tail plane	II	17
	Landing	III	4–7
	Catapulting	III	10
	Catapulting	V	Sect. IV.
	Arrested landing	III	11
	Wires cut	II	23
	Jacking loads	III	15
	Fixing of ballast weights and other large masses	III	19
	Safety belt and harness loads	III	20
Engine mounting	Six times gravity loads	II	18
	Turning	II	18
	Static thrust and torque	III	17
	Side loads	II	19
	Landing—as for undercarriage	III	4–7
	Fitting of ring cowlings	III	27
Ailerons and their attachments	Mass-balance	IV	2
	20° aileron angle in horizontal flight	II	11
	Tail-to-wind loads	III	18
	Aileron instability	IV	3
	Torsional stiffness of ailerons	IV	43
	Mass-balance arms	II	28
Undercarriage	Landing cases	III	4–7
	Wheels (including tail wheels)	IV	15
	Wheel brakes	IV	14
	Brake operating gear	II	20
	Arrested landing	III	11

TABLE I—*continued*

Component.	*Relevant requirements.*	*For particulars see* Chap.	Para.
Undercarriage—*continued*	Wings folded	III	16
	Safe limit of deterioration of shock absorber legs	IV	33
	Static thrust and torque	III	17
	Retractable undercarriage	IV	41
	Undercarriage springing characteristics	IV	42
Hull. (Boat seaplanes)	Landing tail up	III	6
	Two wave landing	III	6
	Pressure over planing bottom	III	6
	Static thrust and torque	III	17
	Also relevant cases specified for the front and rear fuselage	(*see* above)	
Floats. (Seaplanes)	Landing tail up	III	7
	Two wave landing	III	7
	Pressure over planing bottom	III	7
	Static thrust and torque	III	17
Tail plane and elevator	Terminal velocity dive	II	4
	Over-riding minimum tail load	II	7
	Fast glide	II	5 and 6
	Normal flight, C.P. forward	II	2
	Normal flight, C.P. back	II	3
	Over-riding torsional loading	II	17
	Tail-to-wind (elevator)	III	18
	Wires cut	II	23
	Tail adjusting gear to be irreversible	II	20
	Rib tests	III	24
	Rib removed	III	22
	Divided elevators	IV	12
	Relative strength of lift and anti-lift wires	II	25
	Duplicate wires	II	24
	Aerodynamic loading on long struts	II	26
	Tail plane flutter	IV	4
	Torsional stiffness of elevators	IV	44
Fin and rudder	Rudder mass-balance	IV	4
	Side load	II	16
	Side load	V	Sect. II.
	Tail-to-wind (rudder)	III	18
	Wires cut	II	23
	Rudder power	IV	13
	Clearance between fin and rudder	IV	12
Control circuits	Pilot's effort loads	II	20
	Tail-to-wind	III	18
	Automatic control mechanism	II	22
	Duplication	IV	6
	Elastic stretch	IV	3
	Locking of controls	IV	10
	Cables and chains	IV	7 and 8
	Bearings	IV	9
	Trimming tab control circuits	IV	4

TABLE I—*continued*

Component.	*Relevant requirements.*	*For requirements see* *Chap.*	*Para.*
Seats, bomb racks, etc. ..	General cases	III	21
	Fixing of ballast weights and other large masses	III	19
	Catapulting	III	10
	Catapulting	V	Sect. IV.
Windscreens	General cases	II	27
Ancillary structure	.. ..	III	23
	Beaching chassis and tail trolleys of boat seaplanes	III	26

5. Definitions (*see* also British Standard Glossary of Aeronautical Terms, 1933.)

Specified ultimate factor (*see* Chapter I, para. 2).—In general the loads corresponding to the specified ultimate factor are intended to be twice the greatest loads which are expected during manœuvres appropriate to the type. The specified ultimate factor thus usually includes a factor of safety of 2.

Specified proof factor (*see* Chapter I, para. 2.)

Reserve factor.—The ratio of the load which a component or structure is capable of carrying to the load corresponding to the specified ultimate factor. A component just complying with requirements, therefore, will have a reserve factor of 1·0.

*Factor of safety.**—The factor by which the greatest expected loads are to be multiplied to give the loads corresponding to the specified ultimate factor.

Realized factor.—The reserve factor multiplied by the specified ultimate factor.

Note on loads aising from accelerated motion.—In calculating the greatest loads which are expected during manœuvres appropriate to the type (*see* definition of "specified ultimate factor" above) it is usual to consider the aeroplane as being in accelerated motion on a horizontal portion of its flight path. Thus for an aeroplane of weight W the external loads corresponding to an ultimate factor of N are, when N exceeds 2, to be interpreted as being *twice* the loads necessary to give balance at the specified attitude for a vertical force of $\frac{N}{2}W$ downwards through the centre of gravity of the aeroplane. These loads will correspond to a vertical acceleration of approximately $\left(\frac{N}{2} - 1\right)g$, though accelerometers are commonly calibrated so that in these circumstances an accelerometer reading of approximately $\frac{N}{2}g$ would be obtained. This has given rise to the common but erroneous practice of regarding the ultimate factor N as corresponding to a factor of safety of 2 on the the loads due to an acceleration of $\frac{N}{2}g$.

Chord line of an aerofoil.—The chord line is the straight line through the centres of curvature of the leading and trailing edges of an aerofoil section.

Maximum speed is the maximum indicated air speed attainable in level flight at any altitude (or at one specified altitude) and at any engine r.p.m. up to and including the maximum emergency (i.e. "all-out level") r.p.m. within the limits of permissible level flight boost. In calculating this speed an arbitrary airscrew efficiency of 85 per cent. is to be assumed for all types of variable pitch and fixed pitch airscrews.

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* It should be noticed that the Factor of Safety as thus defined is different from that generally used in other branches of engineering (*e.g. Theory of Structures*, p. 28, by A. Morley).

Normal top speed is 87 per cent. of the maximum indicated air speed as defined above.

Determination of stalling speed

(i) The stalling speed should be determined from full scale or model test results, available data being employed in the order of preference given below.

(ii) Full scale.—For the purpose of strength requirements stalling speed as deduced from full scale tests is defined as the lowest speed at which the aeroplane can be held in a straight glide, engine off, for a period of one minute. The speed corresponding to ground level conditions is to be taken.

(*a*) Official full scale tests, if available, on the actual aeroplane.

(*b*) Official full scale tests on an aeroplane aerodynamically similar.

(iii) Model.—Stalling speed as deduced from model tests is defined as the air speed at which, under ground level conditions, the total lift on the aeroplane is equal to the weight, the attitude of the aeroplane being such that the lift coefficient is a maximum. No allowance for slipstream should be made.

(*a*) Approved model tests on a complete model of the actual aeroplane.

(*b*) Approved model tests on wings of the same arrangement.

(*c*) Approved model tests on a monoplane wing.

Where model tests at different values of Vl are available, tests at the highest Vl should be taken. Corrections, based on approved tests, for aspect ratio, gap, stagger, body lift, scale effect, etc., may be applied, but no increased velocity in the slipstream is to be assumed.

Note.—When flaps or slots are fitted to give an artificially low stalling speed the normal value (corresponding to the flaps and slots in the neutral position) should be taken for stressing purposes.

(iv) Requirements specified in terms of stalling speed are to be complied with at ground level air density.

CHAPTER II.—STRENGTH REQUIREMENTS BASED ON FLIGHT CONDITIONS

1. Tabular summary

The requirements dealt with in this Chapter are summarized in Table I.

TABLE I

SUMMARY OF STRENGTH REQUIREMENTS BASED ON FLIGHT CONDITIONS

Note.—The contents of Chapters II and III have been completely re-arranged by A.L. No. 3.

Loading cases	Factor required unless otherwise specified	Components for which the loading case will usually give design loads (*Subject to Chapter I, para. 4*)	For description of case, see Chapter II
Normal flight, centre of pressure forward	As laid down in specification.	Main planes, fuselage and tail unit	Para. 2
Normal flight, centre of pressure back	As laid down in specification.	Main planes, fuselage and tail unit	Para. 3
Terminal velocity dive	2·2 for experimental aeroplanes. 2·0 for other aeroplanes.	Main planes, fuselage and tail unit	Para. 4
Fast glide—Terminal velocity class aeroplanes	$2\cdot2$ or $1\cdot5\left(\frac{\text{T.V.}}{1\cdot5\ V_{max.}}\right)^2$ (whichever is the greater) for experimental aeroplanes. $2\cdot0$ or $1\cdot5\left(\frac{\text{T.V.}}{1\cdot5 V_{max.}}\right)^2$ (whichever is the greater) for other aeroplanes.	Fuselage, tail unit. Occasionally main planes.	Para. 5
Fast glide—Non-terminal velocity class aeroplanes	2·2 for experimental aeroplanes. 2·0 for other aeroplanes.	Fuselage, tail unit. Occasionally main planes.	Para. 6
Over-riding minimum tail load	The factors specified for T.V. or fast glide cases whichever is appropriate.	Fuselage, tail unit	Para. 7
Aileron wing loadings	2·0	Main planes	Para. 8
Up and down gusts	1·5	Main planes	Para. 9
Inverted flight, high negative incidence	0·5 C.P.F. factor	Main planes	Para. 10
Ailerons.—			
(i) Single seater fighters	1·5	Ailerons and their attachments	Para. 11
(ii) All other aeroplanes	2·5	Ailerons and their attachments	
Use of trailing edge flaps.—			
(i) In a steady glide	2·0	Trailing edge flaps, main planes	Para. 12
(ii) Under an up-gust	2·0		
Strength requirements for wings with tip slots.—			
C.P. forward and up-gust	As for these cases in normal flight.	Main planes	Para. 13
Super-stall	0·5 C.P. Forward factor.		

Lateral strength of slat mechanism . .	2·0	Slat mechanism	Para. 14
Slot locking devices . .	—	Locking device	Para. 15
Fin and Rudder—side load (applicable to aeroplanes both with and without automatic control).—			
(i) For aeroplanes with thrust line (or lines) in plane of symmetry	2·0 at 1·4 times stalling speed.	Tail unit, fuselage	Para. 16 (i)
(ii) For aeroplanes with thrust lines outboard of plane of symmetry	1·5 at normal top speed.	Tail unit, fuselage	Para. 16 (ii)
(iii) Over-riding requirement for aeroplanes with thrust lines outboard of plane of symmetry	2·0 at 1·7 times stalling speed.	Tail unit, fuselage	Para. 16 (iii)
Over-riding torsional loading from tail plane.—			
(i) For aeroplanes with thrust line (or lines) in plane of symmetry	2·0	Tail unit, fuselage	Para. 17
(ii) For aeroplanes with thrust lines outboard of plane of symmetry	2·0	Tail unit, fuselage	Para. 17
Engine mounting.—			
(i) Turning in flight with engine on (6* × gravity forces + airscrew thrust and torque + gyroscopic couple)	1·0	Engine mounting	Para. 18
(ii) Normal flight and landing with engine off (6* × gravity forces)	1·0	Engine mounting	Para. 18
* *Or specified C.P. forward factor whichever is the greater*			
Side load case for engine mounting, front fuselage, seats, bomb racks, etc. (unit gravity loads acting alone and sideways)	1·0	Engine mounting, front fuselage, subsidiary structure	Para. 19
Control circuits (including tail plane adjusting gear and brake operating gears)	1·33	Control circuits	Para. 20
Strength of aeroplanes under automatic control . .	1·33 at normal top speed when incidence suddenly changed to (i) stalling incidence, (ii) first pronounced change in slope of C_L curve. 1·5 in down-gust case.	Main planes	Para. 21
Automatic control mechanism	1·33	Control circuits and mounting of automatic control apparatus	Para. 22
Wires cut.—			
(i) Main planes	Half the factor specified for the undamaged aeroplane	Main planes	Para. 23
(ii) Tail unit		Tail plane and fin	
(iii) Fuselage bulkhead bracing		Fuselage	
Duplicate wires—Requirements for duplicate wires of the same strength replacing a single wire	Each duplicate wire to be capable of taking two-thirds of the total load	Applies throughout the aero-structure	Para. 24
Relative strength of lift and anti-lift wires . .	Vertical component of strength of anti-lift to be at least half that of lift	Main planes and tail plane	Para. 25

TABLE 1—*continued*

Loading cases	Factor required unless otherwise specified	Components for which the loading case will usually give design loads (*subject to Chapter I, para. 4*)	For description of case, see Chapter II
Aerodynamic loading on long struts	As for remainder of structure	Main planes, tail unit	Para. 26
Windscreens	2·0	—	Para 27
Stiffness and strength of mass balance weight arms and attachments.—			
(i) Under normal acceleration of Ng; where N = C.P.F. factor	1·0	Mass-balance attachment	Para. 28
(ii) Under lateral acceleration of 5g	1·0	Mass-balance attachment	
(iii) Under angular acceleration of 500 radians/sec.^{2}	1·0	Mass-balance attachment	

2. Normal flight, centre of pressure forward

(i) This case is intended to represent an abrupt increase in the angle of incidence sufficient to stall the aeroplane when flying at high speed. The intention is to provide a factor of safety of 2 on the loads corresponding to the greatest normal acceleration that is likely to be experienced during flight manœuvres appropriate to the type. The centre of pressure position to be assumed in this condition of flight is tabulated for several aerofoils in Table II. For other aerofoils the most forward position of the centre of pressure within the range $\pm 5°$ of the stalling angle is to be taken, obtained from the best available data. When this condition of flight gives critical loads for the rear spar, both the most forward and the most rearward centre of pressure positions in the above range are to be considered. Both " engine on " and " engine off " conditions are to be considered. In the " engine on " case it is essential to balance out for conditions corresponding to the load $\frac{NW}{2}$ and then double the loads so found to give the specified ultimate C.P. forward loads. These loads are not those which would be obtained by balancing out for conditions corresponding to the load NW.

(ii) *Notation, etc.*—

W = maximum weight of aeroplane.

V = appropriate forward velocity of aeroplane, assumed to be horizontal unless otherwise stated.

V_S = stalling speed.

ρ = air density in slugs per cubic foot. At ground level $\rho = 0{\cdot}002378$.

S = wing area. The suffices U and L refer to the upper and lower planes respectively.

L = wing lift.

D = wing drag.

D_B = body drag, *i.e.* total drag of aeroplane less wing drag.

D_A = airscrew drag in " engine off " condition.

F = forward inertia force acting at C.G. of aeroplane. This force will be distributed throughout the structure in proportion to the masses of the component parts. It represents the additional airscrew thrust that would be required to propel the aeroplane at a speed approximately $V_S\sqrt{\frac{N}{2}}$ when the lift coefficient is such that at speed V_S the lift would balance the weight. The force is, of course, distributed among the various component masses of the aeroplane in an entirely different manner from the distribution of the airscrew thrust.

P = tail load. When considering the equilibrium of the aeroplane as a whole P should be assumed to act at one-third of the tail plane (including elevator) chord aft of the tail plane leading edge.

η = airscrew efficiency.

H = horse power of engine(s).

T = airscrew thrust $= \dfrac{550\eta H}{V \text{ (f.p.s.)}}$

α = angle of incidence for flight case considered.

ϕ = angle between flight path or resultant wind direction and line of airscrew thrust.

N = specified ultimate factor.

C_L = appropriate lift coefficient of a wing or combination of wings.

C_B = body drag coefficient, such that $D_B = \frac{1}{2} \times 0 \cdot 002378\ C_B S V^2$.

m = ratio of lift coefficient on upper main plane to lift coefficient on lower main plane. (*See* Chapter V, Section I, para. 4.)

q = ratio of drag coefficient on upper main plane to drag coefficient on lower main plane. In the absence of more accurate information this may be taken as unity on a biplane without decalage.

TABLE II.—C.P. POSITIONS OF COMMON AEROFOILS

(*See* Chapter V, Section I, para. 6)

N.B.—The C.P.B. formulae in this table are only applicable when they give a C.P. position further aft than the C.P.F. position + 0·1c and refer to small positive incidences only

Aerofoil	Monoplane		Biplane		Full scale C_L max. for biplane	Reference
	C.P.F.	C.P.B.	C.P.F.	C.P.B.		
R.A.F. 14	0·290	0·230 + 0·080/C_L	0·260	0·200 + 0·080/C_L	1·18	R. & M. 323, R. & M. 195 for contour.
R.A.F. 15	0·280	0·250 + 0·032/C_L	0·250	0·220 + 0·032/C_L	1·06	R. & Ms. 859, 872, 888, 1320, 857, 774, 816.
R.A.F. 25	0·280	0·230 + 0·037/C_L	0·250	0·200 + 0·037/C_L	0·92	R. & M. 915.
R.A.F. 26	0·305	0·250 + 0·058/C_L	0·275	0·220 + 0·058/C_L	0·94	R. & M. 943.
R.A.F. 27	0·235	0·335	0·205	0·305	1·04	R. & M. 1027.
R.A.F. 28	0·290	0·240 + 0·041/C_L	0·260	0·210 + 0·041/C_L	1·30	R. & Ms. 1027, 1706.
R.A.F. 30	0·240	0·350	0·210	0·320	1·20	R. & Ms. 928, 1052.
R.A.F. 31	0·285	0·230 + 0·059/C_L	0·255	0·200 + 0·059/C_L	1·26	R. & Ms. 928, 990, 1320.
R.A.F. 32	0·330	0·250 + 0·135/C_L	0·300	0·220 + 0·135/C_L	1·32	R. & Ms. 928, 1006.
R.A.F. 33	0·275	0·250 + 0·014/C_L	0·245	0·220 + 0·014/C_L	1·30	R. & M. 928.
R.A.F. 34	0·250	0·250 + 0·004/C_L	0·220	0·220 + 0·004/C_L	1·20	R. & Ms. 1071, 1146, 1635, 1706.
R.A.F. 36	0·340	0·250 + 0·061/C_L	0·310	0·220 + 0·061/C_L	—	R. & M. 1147.
R.A.F. 38	0·275	0·250 + 0·036/C_L	0·245	0·220 + 0·036/C_L	1·32	R. & Ms. 1543, 1706.
R.A.F. 48	0·275	0·230 + 0·044/C_L	0·245	0·200 + 0·044/C_L	—	R. & M. 1543, 1706.
A.D. 1	0·290	0·235 + 0·062/C_L	0·260	0·205 + 0·062/C_L	—	R. & Ms. 943, 1357.
No. 64	0·280	0·250 + 0·048/C_L	0·250	0·220 + 0·048/C_L	1·08	R. & Ms. 152, 772.
Gottingen 386	0·320	0·230 + 0·098/C_L	0·270	0·200 + 0·098/C_L	—	
Gottingen 387	0·320	0·240 + 0·103/C_L	0·290	0·210 + 0·103/C_L	—	R. & M. 1706
Gottingen 388	0·330	0·250 + 0·103/C_L	0·300	0·220 + 0·103/C_L	—	*Ergebnisse der Aerodynamischen Versuchsanstalt zu Gottingen.*
Gottingen 426	0·335	0·250 + 0·095/C_L	0·305	0·220 + 0·095/C_L	—	R. & M. 1243
Gottingen 429	0·245	0·250 + 0·001/C_L	0·215	0·220 + 0·001/C_L	—	
Gottingen 436	0·305	0·250 + 0·070/C_L	0·275	0·220 + 0·070/C_L	—	
Gottingen 449	0·335	0·235 + 0·112/C_L	0·305	0·205 + 0·112/C_L	—	
M.2	0·230	0·330	0·200	0·300	—	R. & M. 1070, N.A.C.A. Tech. Note No. 221.
M.6	0·235	0·335	0·205	0·305	—	N.A.C.A. Report No. 260.
M.12	0·250	0·250 + 0·022/C_L	0·220	0·220 + 0·022/C_L	—	N.A.C.A. Tech. Note No. 243.
Clark Y	0·280	0·240 + 0·082/C_L	0·250	0·210 + 0·082/C_L	—	N.A.C.A. Tech. Note No. 219.
Clark YH	0·260	0·230 + 0·033/C_L	0·230	0·200 + 0·033/C_L	—	N.A.C.A. Tech. Note. No. 240, R.&M.1706.
Bristol 1a	0·265	0·250 + 0·024/C_L	0·235	0·220 + 0·024/C_L	—	Report No. B.A. 643.
B.I.R. 33a	0·315	0·240 + 0·100/C_L	0·285	0·210 + 0·100/C_L	—	R. & M. 248.
Sloane	0·295	0·240 + 0·057/C_L	0·265	0·210 + 0·057/C_L	—	R. & M. 943.
Eiffel 339	0·230	0·330	0·200	0·300	—	*Bulletin Technique No. 12.*

Note.—The data given in this table are only intended to be an indication of the C.P. positions of monoplanes and biplanes. The C.P. positions of the more modern aerofoils should be obtained from the latest available information.

(iii) When the aeroplane is experiencing a resultant force $\frac{NW}{2}$ the forces acting on the aeroplane are as shown in fig. 1. Of these forces all except $\frac{NW}{2}$ are initially unknown, *i.e.* the nine quantities F, T, L_U, L_L, D_U, D_L, D_B, P and V. Resolving vertically and horizontally and taking moments about the C.G. give the three equations.—

$$L_U + L_L + \mathrm{P} + T \sin \phi - \frac{NW}{2} = 0 \quad \ldots \quad (1)$$

$$D_U + D_L + D_B - T \cos \phi - F = 0 \quad \ldots \quad (2)$$

$$L_U a_U + D_U b_U + T\, e - L_L a_L - D_L b_L - D_B d - P\, l = 0 \quad \ldots \quad (3)$$

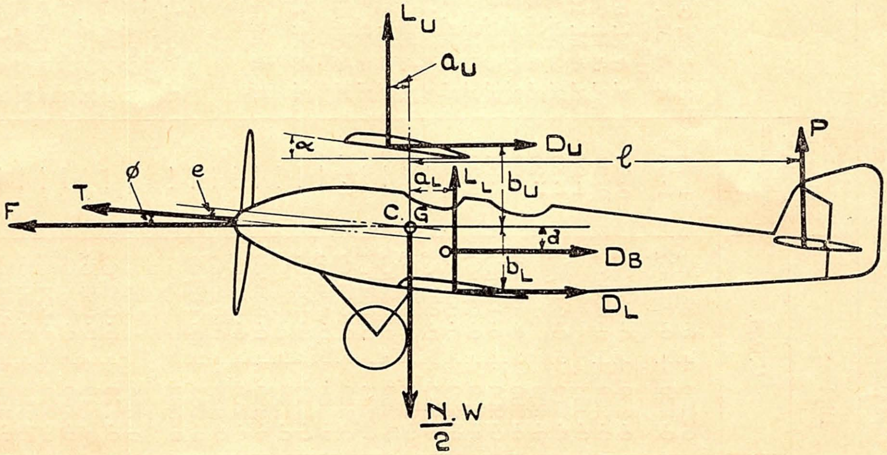

FIG. 1.—CHAP. II. C.P. Forward (engine on).

The remaining six equations needed are

$$\frac{L_U}{L_L} = m\,\frac{S_U}{S_L} \quad \ldots \quad (4)$$

$$\frac{D_U}{D_L} = q\,\frac{S_U}{S_L} \quad \ldots \quad (5)$$

$\dfrac{L_U + L_L}{D_U + D_L}$ = lift/drag ratio at the stall for the aerofoil and wing arrangement concerned, obtained from the best available data (6)

$$L_U + L_L = \tfrac{1}{2} \times 0{\cdot}002378\, C_{L_{max.}}\, SV^2 \quad \ldots \quad (7)$$

$$T = \frac{550\,\eta\, H}{V} \quad \ldots \quad (8)$$

$$D_B = \tfrac{1}{2} \times 0{\cdot}002378\, C_B S V^2 \quad \ldots \quad (9)$$

On eliminating P between equations (1) and (3) and making use of equations (4) to (9) a cubic equation for V is obtained. The value required will be approximately equal to $V_S\sqrt{\frac{N}{2}}$, V_S being the stalling speed. An acceptable approximation, which evades the solution of this cubic, is to assume V equal to $V_S\sqrt{\frac{N}{2}}$ when estimating D_U, D_L, D_B and T. The forces L $(= L_U + L_L)$, P and F, thereby found from equations (1) to (3), will usually be sufficiently accurate. The brake horse power H is to be taken as that appropriate to the r.p.m. of the engine(s) when the aeroplane is in normal horizontal flight at ground level at speed V with the engine(s) correspondingly throttled. The forces calculated as described above multiplied by two are the specified external ultimate loads on the structure. It is important to note that where account is taken of end loads, as for instance in wing spars, the full ultimate loads are to be taken in making the strength calculations. In the " engine off " condition the airscrew thrust T in the above equations should be replaced by the airscrew drag D_A acting through the centre of the airscrew boss and in the direction of the resultant wind. For purposes of calculation, the specified ultimate loads in the " engine off " case may be obtained by balancing out at a speed $Vs\sqrt{N}$, instead of by doubling the loads obtained in balancing out at a speed $Vs\sqrt{\frac{N}{2}}$; the result will be the same.

3. Normal flight, centre of pressure back

(i) This condition is intended to represent a normal acceleration developed at a smaller lift coefficient than that considered in the C.P. forward case. The lift coefficient to be taken is that corresponding to steady horizontal flight at normal top speed. The " engine off " case only need be considered.

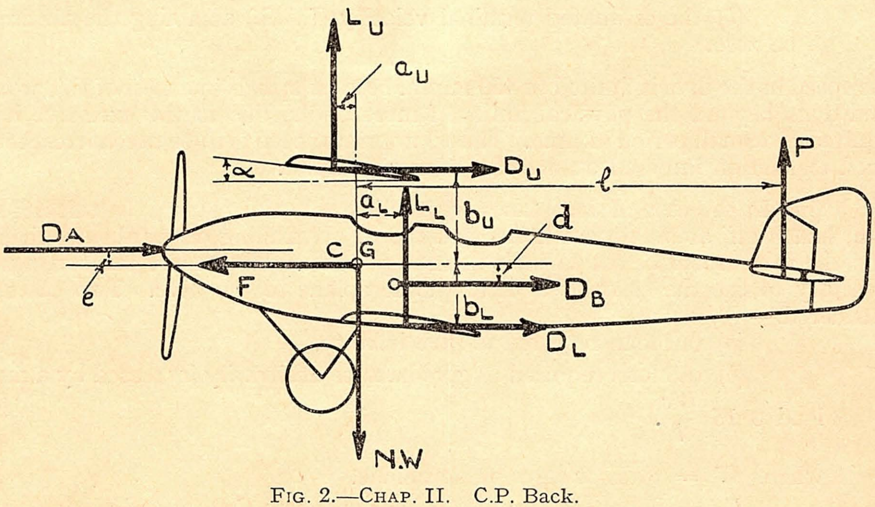

FIG. 2.—CHAP. II. C.P. Back.

(ii) With the proviso given below, the centre of pressure position to be assumed in this condition of flight is tabulated for several aerofoils in Table II, in terms of the lift coefficient C_L corresponding to flight at normal top speed. For other aerofoils the centre of pressure position should be obtained from the best available data.

(iii) The centre of pressure position taken in this C.P. back case is to be at least $0 \cdot 1$ c further aft than that specified for the C.P. forward case.

(iv) The factored forces on the aeroplane in the C.P. back condition of flight are shown in fig. 2.

(v) The calculation of the factored forces follows closely the procedure described in the C.P. forward case " engine off." As in that case balancing out may be done for the loads corresponding to the speed $V\sqrt{N}$, V being the normal top speed, and such balancing out will give the specified ultimate loads directly. Balancing out at speed $V\sqrt{\frac{N}{2}}$ and doubling the loads so obtained would give the same result. Thus the speed needed in estimating the body drag corresponding to the ultimate factor N will be approximately equal to $V\sqrt{N}$. It is immaterial that $V\sqrt{N}$ will usually be above the terminal velocity. The airscrew drag will be N times the drag at normal top speed with engine off. The C_L max. in equation (7) of para. 2 should be replaced by the C_L appropriate to steady horizontal flight at normal top speed and the 0·002378 replaced by ρ for the rated altitude.

* **4. Terminal velocity dive** (applicable to terminal velocity class aeroplanes only)

(i) All experimental aeroplanes of this class are to have a factor of 2·2 and other aeroplanes of this class a factor of 2·0 (unless otherwise specified) in a dive (engine off) at terminal velocity (*see* para. (ii)). The attitude giving maximum tail load, which may not be exactly the terminal velocity attitude is to be taken. The forces acting on the aeroplane in this condition of flight are to be based on the best available data.

(ii) The aeroplane is assumed to be in a steady or accelerated dive at whichever is the lower of the following speeds.—

(*a*) 450 m.p.h. I.A.S.,

(*b*) the estimated terminal velocity (I.A.S.) assuming the airscrew drag to be zero.

The speed in the dive is limited to 450 m.p.h. because speeds in excess of this would impose conditions beyond the physical limits of most pilots, due to the excessive height drop required in a small period of time. The appropriate speed is to be taken irrespective of any speed restriction imposed on account of engine r.p.m.

(iii) In the case of aeroplanes with adjustable tail planes, the tail is to be set to trim, hands off, at maximum horizontal speed. If trailing edge tabs are fitted instead, then the tabs are to be set in their neutral position. With these tail settings it will be necessary to use the elevator to hold the aeroplane in the dive. Two cases are to be considered.—

(*a*) tail load required to give balance,

(*b*) tail load required to give balance arbitrarily increased by a manœuvring load $0 \cdot 15\ \frac{Wc}{l}$.

where W = total weight of aeroplane.

l = distance from C.G. of aeroplane to C.P. of tail plane and elevator ; (*note*.—C.P. of tail plane and elevator is taken as 0·25 of total chord measured from the leading edge).

c = mean chord of wings.

The full factor is to be realized both with and without the addition of the manœuvring load. This additional manœuvring load is intended to represent the impulsive load due to use of elevators and will give rise to angular and linear accelerations with corresponding inertia forces.

(iv) The centre of pressure of the tail plane and elevator is implicit in the tail setting and the elevator angle. A method of determining the centre of pressure of the tail plane and elevators is given in Chapter V, Section VII. The chordwise distribution of the load is to be adjusted to give the centre of pressure position so obtained.

* Previously A.D.Ms. 341, 346 and 306.

5. Fast glide—Terminal velocity class aeroplanes

(i) This case is only applicable to terminal velocity class aeroplanes with adjustable tail planes and is mainly intended as a further strength criterion for the tailplane and elevator.

(ii) The following factors are to be taken.—

(*a*) experimental aeroplanes: $2 \cdot 2$ or $1 \cdot 5 \left(\frac{T.V.}{1 \cdot 5\, V_{max}}\right)^2$ whichever is the greater (unless otherwise specified).

(*b*) all other aeroplanes: $2 \cdot 0$ or $1 \cdot 5 \left(\frac{T.V.}{1 \cdot 5\, V_{max}}\right)^2$ whichever is the greater (unless otherwise specified).

(iii) The lowest of the following speeds is to be used.—

(*a*) 1·5 times maximum level indicated speed.

(*b*) the estimated terminal velocity (I.A.S.) assuming the airscrew drag to be zero.

(*c*) 450 m.p.h. I.A.S.

The appropriate speed is to be taken irrespective of any speed restriction imposed on account of engine r.p.m.

(iv) The attitude is to be that appropriate to a terminal velocity dive (engine off) i.e. the aeroplane will be accelerating unless the speed is terminal velocity.

(v) The tail plane is to be set in the most adverse position. Although this tail plane setting does not alter the total tail load the distribution is altered due to the larger elevator angle required to hold the aeroplane in the dive. Two cases are to be considered.—

(*a*) tail load required to give balance,

(*b*) tail load required to give balance arbitrarily increased by a manœuvring load $0 \cdot 15 \frac{Wc}{l}$, where W, c and l have the same significance as previously.

The full specified factor is to be realized both with and without the addition of the manœuvring load.

(vi) As in the terminal velocity dive case the centre of pressure of the tail plane and elevator is implicit in the tail setting and the elevator angle. A method of determining the centre of pressure of the tail plane and elevators is given in Chapter V, Section VII.

* **6. Fast glide—Non-terminal velocity class aeroplanes**

(i) Experimental aeroplanes in this class are to have a factor of 2·2 and other aeroplanes of this class a factor of 2·0 (unless otherwise specified) in a fast glide at the speed defined in sub-para. (ii) below.

(ii) The speed of 1·3 times maximum indicated air speed in level flight at 5,000 ft. (unless otherwise specified) is to be taken for the following types of aeroplanes.—

Multi-engined medium bomber.
,, ,, heavy bomber.
,, ,, bomber transport.
,, ,, general reconnaissance and torpedo bomber.
,, ,, seaplane.
All flying boats.

* Previously A.D.Ms. 306 and 351.

The speed of $1 \cdot 5$ times maximum indicated airspeed in level flight at 5,000 ft. (unless otherwise specified) is to be taken for all other aeroplanes. These speeds are to be taken irrespective of any speed restriction imposed on account of engine r.p.m., subject to a maximum of 450 m.p.h. I.A.S.

(iii) The attitude is to be that corresponding to a steady glide at the appropriate speed with engine off.

(iv) If the tail plane is adjustable it should be set at the most adverse position anywhere within the whole range of adjustment.* If the tail plane is not adjustable but trailing edge tabs are fitted instead, the tabs are to be set in their neutral position. Two cases are to be considered.—

(*a*) tail load required to give balance,

(*b*) tail load required to give balance arbitrarily increased by a manœuvring load $0 \cdot 15 \frac{Wc}{l}$ where W, c and l have the same significance as previously.

The full specified factor is to be realized both with and without the addition of the manœuvring load.

(v) As in the preceding case the centre of pressure of the tail plane and elevator is implicit in the tail setting and the elevator angle. A method of determining this centre of pressure is given in Chapter V, Section VII.

† 7. Over-riding minimum tail loads

(i) The maximum tail load obtained from the fore-and-aft balance calculations in the terminal velocity dive condition, in the fast glide case, or other appropriate case, is, generally, the load that will be used for strength calculations. In certain types of design, however, this load may be so small that it is no longer satisfactory as a criterion for strength. To guard against this possibility, the structure is to have the appropriate factors as specified for the T.V. or fast glide case under the following over-riding minimum up and down tail loads. These cases are applicable to the tail plane, elevator, tail plane attachments and the fuselage. Overall balance of forces may be achieved by assuming the wings held rigidly.

$$\textit{Up load } (a)\ P = \frac{\frac{1}{2} \times 0 \cdot 020 \times 0 \cdot 002378\, cSV^2}{l} + 0 \cdot 15 \frac{Wc}{l}$$

$$\textit{Down load } (a)\ P = \frac{\frac{1}{2} \times 0 \cdot 030 \times 0 \cdot 002378\, cSV^2}{l}$$

$$(b)\ P = \frac{\frac{1}{2} \times 0 \cdot 030 \times 0 \cdot 002378\, cSV^2}{l} + 0 \cdot 15 \frac{Wc}{l}$$

* *Note.*—If it is difficult to make structural provision for the loads due to extreme tail settings, the designer may initially make provision only for the range of tail settings which, according to calculations, correspond to a practical load at the pilot's hand. The designer must then put a temporary notice in the cockpit warning the pilot that at speeds above a stated limit the tail setting must not be outside a specified range. If during subsequent contractor's flight trials it is found that the large load at the pilot's hand precludes tail settings outside the specified range then the temporary cockpit notice may be removed. If contractor's flight trials show the reverse, then *either* the cockpit notice must be made permanent, *or* the structure must be strengthened. A ruling as to which course is to be adopted should be obtained from the Airworthiness Department.

† Previously A. D.Ms. 341 and 351.

where W = total weight of aeroplane.

S = area of main planes.

c = mean chord of wings.

l = distance from C.G. of aeroplane to C.P. of tail plane and elevator.

Note.—C.P. of tail plane and elevator is taken as 0·25 of total chord measured from the leading edge.

V = appropriate diving speed for either T.V. or non-T.V. class aeroplanes (*see* paras. 4 (ii), 5 (iii) and 6 (ii)).

(ii) The terminal velocity attitude is to be assumed for both classes of aeroplane. The centre of pressure position to be taken for down loads is that already obtained for fore-and-aft balance calculations for the T.V. or fast glide case, whichever is appropriate. For up loads two centre of pressure positions are to be taken : (*a*) on the L.E. and (*b*) at 0·5 c′ where c' is the total tail plane plus elevator chord.

* 8. Aileron wing loads

(i) *All aeroplanes.*—The wings and centre section must have a factor of at least 2·0 under the aerodynamic loads due to 20° aileron angle at 80 per cent. of maximum level indicated speed. These loads are to be taken to be the sum of the following.—

(*a*) Load due to steady level flight at 80 per cent. of maximum level indicated speed with the ailerons neutral.

(*b*) Pitching moment, at the same speed, due to an additional moment coefficient (C_m) of 0·14 applied to the section of the wing in front of the ailerons only. This moment is to be applied in opposite senses to the port and starboard wings.

(ii) *Terminal velocity class aeroplanes.*—The wings and centre section of all T.V. class aeroplanes are to have, in addition to case (i) above, a factor of at least 2·0 under the aerodynamic loads due to 3° aileron angle at the appropriate speed for this class (*see* para. 4 (ii)). These loads are to be taken to be the sum of the following.—

(*a*) Load due to steady or accelerated dive at the appropriate speed with the ailerons neutral.

(*b*) Pitching moment, at the same speed, due to an additional moment coefficient (C_m) of 0·03 applied to the section of the wing in front of the ailerons only. This moment is to be applied in opposite senses to the port and starboard wings.

(iii) *Non-terminal velocity class aeroplanes.*—The wings and centre section of all non-T.V. class aeroplanes are to have, in addition to case (i) above, a factor of at least 2·0 under the aerodynamic loads due to 6° aileron angle at the appropriate speed for this class (*see* para. 6 (ii)). If linear interpolation between 20° at 80 per cent. of maximum level indicated speed and 3° at 450 m.p.h. I.A.S. gives a smaller angle than 6°, then this smaller angle may be taken instead of 6°. The above loads are to be taken to be the sum of the following.—

(*a*) Loads due to a steady or accelerated dive at the appropriate speed with the ailerons neutral.

(*b*) Pitching moment, at the same speed, due to an additional moment coefficient (C_m) of 0·05 applied to the section of the wing in front of the ailerons only. This moment is to be applied in opposite senses to the port and starboard wings.

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* Previously A.D.M. 355.

In cases in which the above requirements appear unduly severe, due to structural limitation of aileron angle to less than 20°, or due to other structural cause, reference should be made to the Airworthiness Department for a ruling as to the loading to be assumed.

* 9. Up and down gusts

(i) The wings and their attachments to the body are to have a factor of 1·5 (unless otherwise specified) under the loads produced by up and down gusts normal to the flight path of magnitude up to and including 25 f.p.s. indicated air speed, encountered when the aeroplane is flying (zero thrust may be assumed) under the following flight conditions.—

(*a*) For terminal velocity class aeroplanes, in an accelerated dive in the terminal velocity attitude (*see* para. 4 (ii)) at 1·5 times maximum level indicated air speed.

(*b*) For non-terminal velocity class aeroplanes in a steady dive at the appropriate speed as defined in para. 6 (ii).

The speeds specified above, 1·5 V_{max} (or 1·3 V_{max}) may be greater than the terminal velocity or the limiting diving speed specified for stressing purposes. The gust requirements must nevertheless be complied with at 1·5 V_{max} (or 1·3 V_{max}) although in such a case this speed could not be attained. This artificial stressing case is necessary as the down gust requirement is intended to cover the previous low incidence inverted flight requirement which has been cancelled, but it only does so provided that a speed of 1·5 V_{max} (or 1·3 V_{max}) is taken.

(ii) The above requirements are to be fulfilled with the aeroplane fully loaded. They will not necessarily be fulfilled when the aeroplane is flying light. The detail assumptions to be made in estimating the effects of such gusts are given below.

(iii) For the purpose of routine strength calculations it will be assumed that the effect of a gust U is to change the wing incidence by an amount $\tan^{-1}U/V_i$, V_i being the appropriate indicated air speed. The instantaneous conditions to be represented in routine strength calculations are therefore.—

$$\text{Resultant wind speed} = \sqrt{V^2_i + U^2}$$
$$\text{Incidence} = \alpha_o \pm \tan^{-1} U/V_i$$

where α_o is the incidence appropriate to condition (*a*) or (*b*) of sub-para. (i) above. The centre of pressure corresponding to this incidence should be obtained from the best available data.

The lift component of the air load on the main planes on entering the gust is

$$\frac{1}{2} \times 0{\cdot}002378 \frac{dC_L}{d\alpha} \left(\alpha_o \pm \tan^{-1} \frac{25}{V_i}\right) SV_i{}^2$$

(iv) There is some evidence to show that the stalling angle of an aerofoil is increased when the incidence change takes place very rapidly. An allowance for this effect is to be made by extending the $C_L - \alpha$ curve corresponding to the aeroplane considered to a maximum lift coefficient of 2·0. If, then, the incidence change $\tan^{-1} U/V_i$ produces conditions beyond the stall, some smaller gust speed should be taken corresponding to the incidence change needed just to stall the wings.

(v) The loads to be assumed in the up gust case are shown in fig. 3. Note that the inertia and gravity forces are resolved along and normal to the line of the resultant wind. Since both resultant wind speed and incidence are specified, the forces L_U, L_L, D_U, D_L and

* Previously A.D.M. 306.

D_B are directly calculable. The airscrew drag parallel to flight path is also known. The three remaining forces F, nW and P can be obtained from the balance equations :—

$$L_U + L_L + P - nW = 0 \quad \ldots \quad (1)$$

nW being the resultant inertia and gravity force necessary for balance in the given conditions.

$$D_U + D_L + D_B + D_A - F = 0 \quad \ldots \quad (2)$$

$$L_U\, a_U + D_U\, b_U - D_A e - L_L\, a_L - D_L\, b_L - D_B d - Pl = 0 \quad \ldots \quad (3)$$

the notation used being as for the C.P. forward case.

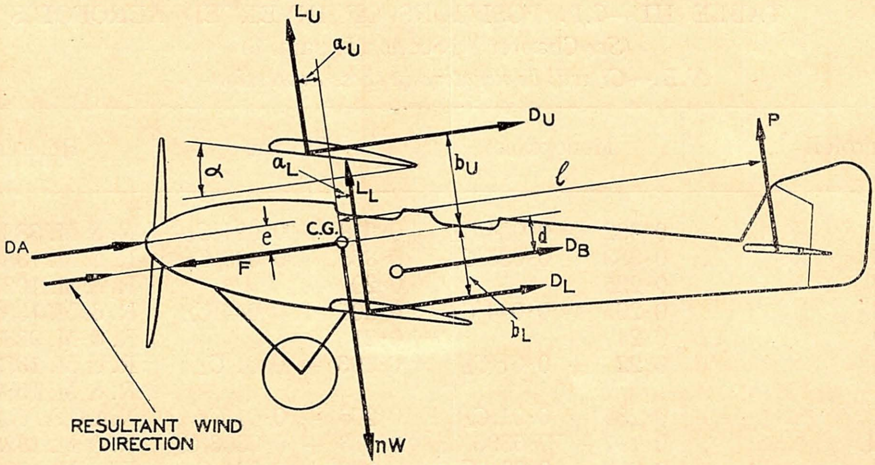

FIG. 3.—CHAP. II. Up gust case.

(vi) The loading conditions to be assumed in the down gust case are shown in fig. 4. Again all three of the forces, i.e. F', $n'W$ and P' are directly calculable, and these three forces can be obtained from the three balance equations.—

$$L'_U + L'_L - P' - n'W = 0 \quad \ldots \quad (4)$$

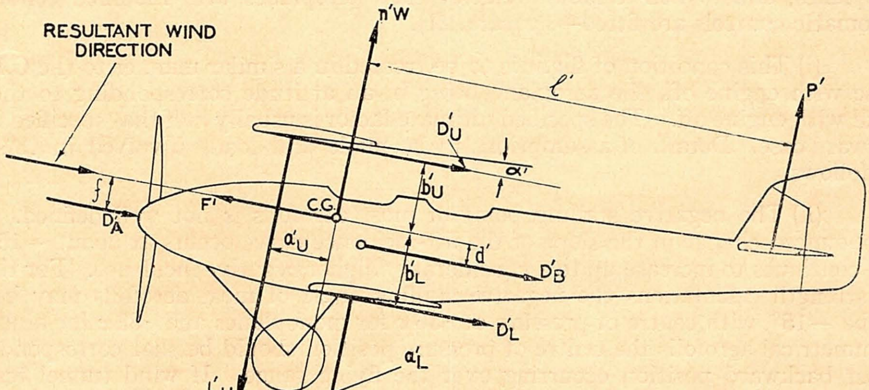

FIG. 4.—CHAP. II. Down gust case.

the resultant inertia and gravity force necessary for balance in the given conditions.

$$D'_U + D'_L + D'_B + D'_A - F' = 0 \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (5)$$

$$L'_U\, a'_U - L'_L\, a'_L + D'_L\, b'_L + D'_B\, d' + D'_A f + P'\, l' - D'_U\, b'_U = 3 \quad .. \quad (6)$$

(vii) The centre of pressure for the down gust condition is given for several aerofoils in Table III. The C.P. formulae given in this table refer to low negative incidences in the region of no lift and are, therefore, different from those for the C.P. back positions given in Table II. For the aerofoils, if no wind tunnel tests at negative incidences are available, the $C_M - C_L$ relationship can be extended backwards from positive angles.

TABLE III.—C.P. POSITIONS OF INVERTED AEROFOILS
(*See* Chapter V, Section I, para. 6)
N.B.—C_L will be negative in these expressions

Aerofoil	Monoplane	Biplane	Reference
R.A.F. 14	$0{\cdot}432 + 0{\cdot}084/C_L$	$0{\cdot}378 + 0{\cdot}084/C_L$	R. & M. 388.
R.A.F. 15	$0{\cdot}383 + 0{\cdot}035/C_L$	$0{\cdot}335 + 0{\cdot}035/C_L$	R. & M. 1383.
R.A.F. 27	$0{\cdot}225$	$0{\cdot}20$	R. & M. 1027.
R.A.F. 28	$0{\cdot}235 + 0{\cdot}044/C_L$	$0{\cdot}205 + 0{\cdot}044/C_L$	R. & M. 1383.
R.A.F. 30	$0{\cdot}24$	$0{\cdot}21$	R. & M. 928.
R.A.F. 34	$0{\cdot}22 + 0{\cdot}01/C_L$	$0{\cdot}193 + 0{\cdot}01/C_L$	R. & M. 1383. R. & M. 1087.
M.6	$0{\cdot}234 + 0{\cdot}01/C_L$	$0{\cdot}205 + 0{\cdot}01/C_L$	N.A.C.A. Report 336.
Clark Y.H.	$0{\cdot}197 + 0{\cdot}0306/C_L$	$0{\cdot}173 + 0{\cdot}0306/C_L$	R. & M. 1383.
Bristol 1A	$0{\cdot}258 + 0{\cdot}024/C_L$	$0{\cdot}226 + 0{\cdot}024/C_L$	R. & M. 1383.

(viii) The tail load is not intended to be a design criterion for any part of the structure in these up and down gust cases.

10. Inverted flight, high negative incidence (Normally this case applies to all terminal velocity class aeroplanes, and to non-terminal velocity class aeroplanes with modified conditions, e.g. when automatic controls are fitted—*see* para. 21).

(i) This condition of flight is to be treated in a similar manner to the C.P. forward case with engine off, the aeroplane being in an attitude corresponding to the inverted stall with engine off. The specified ultimate factor is usually half that specified in the C.P. forward case. Details of assumptions to be made and loads involved in this case are as follows.

(ii) The negative stalling point of most aerofoils is not well defined. The first pronounced change in the slope of the lift-incidence curve occurs at about $-15°$, but the lift continues to increase up to a considerably higher negative incidence. For the purpose of strength calculations the negative stalling angle of most aerofoils may be assumed to be $-15°$, with centre of pressure at $\cdot 36\,c$ for monoplanes and $\cdot 33\,c$ for biplanes. For symmetrical aerofoils the centre of pressure position should be that corresponding to the most backward position occurring over the flying range. If wind tunnel tests on the particular aerofoil concerned show these conditions to be inapplicable, the matter should be referred to the Airworthiness Department. Unless more accurate data are available the ratio L/D can be taken as 3.

(iii) The forces acting on the aeroplane in this condition of flight are shown in fig. 5. The calculation of these forces follow closely the procedure described in the C.P. forward case. The speed V needed in estimating the body and airscrew drag will be approximately equal to $Vs\sqrt{N}$, Vs being the stalling speed of the aeroplane when inverted, and N the specified ultimate factor in this high incidence inverted flight case.

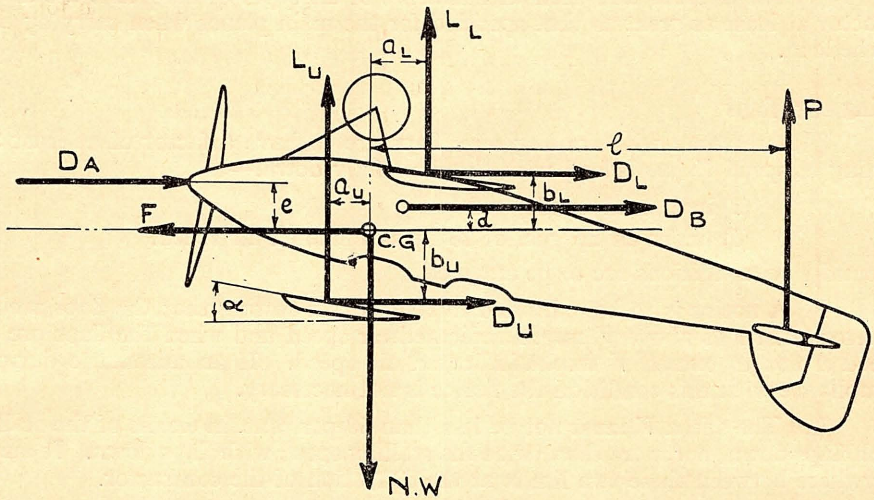

FIG. 5.—CHAP. II. Inverted flight—high negative incidence.

11. Ailerons

(i) The ailerons themselves and their attachments* are to comply with whichever of the following two requirements is quoted in the aeroplane specification.—

(*a*) A factor of 1·5 under a total aileron load given by

$$L = k_\xi \, \varrho \, S_\xi \, V^2_{max}$$

V_{max} being the *maximum* speed of steady horizontal flight and ϱ the air density at the appropriate altitude (*see* Chapter I).

(*b*) A factor of 2·5 under a total aileron load given by

$$L = k_\xi \, \varrho \, S_\xi \, V^2$$

V being 80 per cent. of the normal top speed.

In general requirement (*a*) applies to fighter aeroplanes and requirement (*b*) to all other types.

(ii) The above expressions are intended to represent the loads produced when the ailerons are suddenly moved when the aeroplane is in high speed flight.

(iii) In each case S_ξ is the total aileron area and k_ξ the normal force coefficient. The value of k_ξ and the C.P. position is to be taken corresponding to 20° aileron angle even though the maximum possible movement of the ailerons is less than 20°, as it is found in practice that the pilot uses full aileron angle at higher speeds when the maximum available movement is reduced, e.g. by low gearing. The value of k_ξ and C.P. position should be taken from the best available data or in the absence of more precise information k_ξ may be assumed to be 0·4 and the C.P. distant $0{\cdot}4c'$ aft of the aileron leading edge

* Not including the control circuit, requirements for which are given in para. 20. The aileron actuating lever is to comply with whichever is the more severe of the requirements of paras. 11 and 20.

(c' = aileron chord). It is to be assumed that the pressure distribution along the span of the aileron is proportional to the aileron chord, with no falling off in the intensity of loading due to wing tip effect.

(iv) Individual consideration will be given to any special case to which the terms of the requirement do not appear strictly applicable.

(v) The ailerons and their attachments of all types of aeroplane are also to have a factor at least as great as that specified for the main planes when carrying their share of the load.

12. Trailing edge flaps

(i) Aeroplanes with trailing edge flaps are to have a factor of at least 2·0 when gliding at speed V (*see* below) with the flaps fully down.—

(*a*) in a steady glide, and

(*b*) under an up gust of 25 f.p.s. indicated air speed.

Ground level conditions are to be assumed.

(ii) A notice is to be exhibited in the cockpit stating that the flaps are not to be lowered at speeds above V m.p.h. indicated air speed and when the flaps are down the speed is not to exceed V m.p.h. indicated air speed. If an automatic device ensures compliance with this condition the notice is not necessary.

(iii) The speed V must not be less than 50 per cent. in excess of the stalling speed with flaps down, nor more than twice the stalling speed with flaps down. It may be fixed anywhere between these two limits at the discretion of the contractor.

(iv) If it is desired to fit flaps for use at speeds greater than twice the stalling speed with flaps down different stressing requirements will apply. No standard requirements have yet been laid down, and each case will be dealt with individually.

13. Strength requirements for wings with tip slots

(i) The following strength requirements are applicable to wings with automatic tip slots, with or without interceptors, designed in accordance with the confidential memorandum "The Design of Handley Page Slots and Slot and Interceptor Control". It is not proposed at this stage to standardize strength requirements for any other types of slot, which for the present will be dealt with individually.

(ii) The figures quoted are applicable to monoplanes and biplanes of which about 40 per cent. of the top wing span is slotted, the bottom wing unslotted, and the slat chord between 12 per cent. and 15 per cent. of the main chord.

(iii) Aeroplanes with slotted wings are to comply with all the strength requirements specified for unslotted aeroplanes. Loading cases corresponding to an incidence at which the slot will be shut will be unaffected except in so far as the load on the slat gives rise to concentrated loads on the front spar. The slot will be open in two of the standard loading conditions: the up-gust and the C.P. forward case. Detailed assumptions to be made in these two cases are given in (iv) below. An additional requirement, the super-stall case, is to be complied with on slotted wings. Details of this case are given in (v) below. A criterion for the lateral strength of the slat mechanism is given in para. 14 and for the strength of locking gear (if any) in para. 15.

(iv) *C.P. forward and up-gust.*—The load distribution along the wing span, the centre of pressure position along the chord and the load distribution between upper and lower planes are to be assumed unaffected by the slot. The resultant loading on the slat at any section is to be assumed to be such that its component normal to the wing chord is one-eighth of the total normal wing loading at that section. The direction of the resultant loading on the slat is to be assumed to be at right angles to the slat chord. Hence the magnitude of the resultant load on the slat can be calculated, it being such as to give the specified component normal to the wing chord. This resultant load acts at ·45 of the slat chord from the slat leading edge. The slat chord line is a line passing through the centre of curvature of the leading edge of the aerofoil formed by the envelope enclosing the wing and slat, with slot closed, and the slat trailing edge.

(v) *Super-stall.*—Aeroplanes are to have at least half the specified ultimate C.P. forward factor under the following forces, with the proviso that the factor is to be not less than 3.—

(*a*) A normal air force on the main planes equal to the unit (i.e. corresponding to a factor of 1·0) lift load on the main planes in the C.P. forward case.

(*b*) A tangential air force zero over the unslotted portion of upper and lower planes and a quarter of the normal air force over slotted portion, acting forwards.

(*c*) Zero airscrew drag.

(*d*) A tail load to give moment balance (the moment due to body drag may be neglected).

(*e*) Inertia and gravity loads equal and opposite to resultant air loads.

(vi) *Load distribution along wing span*

(*a*) Monoplane and upper plane of biplane (rectangular plan form) : distribution to be derived from the C.P. forward lift distribution curve by increasing the ordinates over the slotted portion by 50 per cent.

(*b*) Lower plane (rectangular plan form) : as for C.P. forward case.

(*c*) Tapered wings : loading proportional to chord over the unslotted portion, and proportional to 1·5 times the chord over the slotted portion.

(vii) *Centre of pressure along chord.*—The centre of pressure along the slotted portion is to be assumed the same as in the C.P. forward case. The centre of pressure along the unslotted portion is to be $0 \cdot 1\ c$ further aft than this.

(viii) *Distribution of load between upper and lower planes.*—The following plane loading ratio is to be assumed.—

$$\frac{\text{Mean normal loading on top plane}}{\text{Mean normal loading on bottom plane}} = 0 \cdot 8.$$

(ix) *Distribution of load between slat and main wing.*—The slat is to carry one-third of the normal wing load at each section. The total resultant slat load normal to the slat chord is to be determined as in the C.P. forward case. C.P. of slat load as in C.P. forward case also.

14. Lateral strength of slat mechanism

The slat and its attachments are to have a factor of at least 2 under the centrifugal force due to an angular velocity Ω about a vertical axis through the C.G. wing span horizontal, where—

$$\Omega = \frac{150}{\text{span (ft.)}} \sqrt{\frac{\text{Wing loading (lb./sq. ft.)}}{10}} \text{ radians per second.}$$

The slat is to be assumed open. This will give a specified ultimate side load on the slat of approximately.—

$$L \text{ (lb.)} = \frac{140 \times \text{slat weight (lb.)} \times \text{Wing loading (lb./sq. ft.)} \times d \text{ (ft.)}}{\text{Wing span (ft.)}^2}$$

where d = distance (ft.) between vertical axes through C.G. of aeroplane and C.G. of slat.

15. Slat locking devices

If a catch is provided to hold the slat shut, the strength of the catch is to be less than that of the slat and attachments so that the catch would break before any part of the structure was damaged. This condition is to be fulfilled under a load tending to open the slat, uniformly distributed along the slat span, acting at any angle within 10° fore and aft of the normal to slat chord and at 0·45 of the slat chord from the slat leading edge. The catch is to be arranged so that its breakage would not cause the slat to jam.

16. Fin and rudder loads* (Applicable to aeroplanes both with and without automatic control—*see* also Chapter V, Section II).

(i) *Aeroplanes with thrust line (or lines) in plane of symmetry.*—The structure is to have a factor of at least 2 under a load P given by the expression.—

$$P = \tfrac{1}{2} \times 0 \cdot 002378\, C\, S'' \,(1 \cdot 4\, V_S)^2.$$

where $C = 1$ for ratios $\dfrac{\text{rudder area}}{\text{fin plus rudder area}}$ up to 0·7.

For values of this ratio between 0·7 and 1, C varies linearly from 1 to 1·4.

S'' = total area of fin and rudder (sq. ft.).

V_S = stalling speed (f.p.s.).

(ii) *Aeroplanes with thrust lines outboard of plane of symmetry.*—The structure is to have a factor of at least 1·5 under the load due to instantaneous application of full rudder angle at normal top speed.

In calculating this load the slipstream is to be taken into account, together with the maximum side force coefficient obtainable on the fin and rudder deduced from the best available data; this may be less than the stalling side force coefficient of the rudder owing to limitation of rudder movement. In determining this coefficient the aeroplane is to be taken in the attitude of normal top speed flight at zero angle of yaw. In the absence of wind tunnel tests on the particular fin and rudder concerned, the results recorded in R. & M. Nos. 1321, 1329 and 1330 will give a good indication of the approximate value of the side force coefficient.

(iii) *Over-riding requirement for aeroplanes with thrust lines outboard of plane of symmetry.*—As an over-riding requirement, the structure is to have a factor of at least 2 under a load given by the expression.—

$$P = \tfrac{1}{2} \times 0 \cdot 002378\, C\, S'' \,(1 \cdot 7\, V_S)^2$$

where C, S'' and V_S are as defined in (i) above. This over-riding requirement may be waived in exceptional cases if it can be shown that at normal top speed the most unfavourable combination of rudder angle, angle of yaw and slipstream distribution, taken in conjunction with the maximum probable full scale value of side force coefficient, gives a lower load than P. In such cases this lower load may replace the value of P given above. No limitation or rudder angle to that corresponding to a given hinge moment is permissible.

† (iv) To prevent failure of the attachments of the fin to the fuselage due to vibration, these attachments are to have a factor of at least twice that required in the remainder of the structure (i.e. tail plane and fuselage adjacent to the attachments). When there is difficulty in determining what comprises the fin attachments the deciding consideration is liability of the relevant portions of the structure to stress concentration under fin vibration. The fin must also be as rigid as possible. To this end the thickness-chord

* A convenient method of calculating the stresses in a framework fuselage due to fin and rudder torsion (or tail skid torsion—*see* Chapter III, para. 9 (i) (*a*)) is given in R. and M. 1586.

† Previously A.D.M. 289.

ratio of the combination of fin and rudder could with advantage be as great as 0·2 in cases where the height of the fin is such that the greatest possible lateral support is required.

Note.—To avoid confusion, attachments which exactly fulfil these conditions should be referred to as having a reserve factor of 1·0, not 2·0. *See* definitions of reserve factor in Chapter I, para. 5.

17. Over-riding torsional loading from tail plane*

On unusual designs there is always the possibility of the torsion due to the fin and rudder loads (*see* para. 16) being small or zero. As an overruling condition, therefore, the fuselage, tail plane and tail plane attachments of single engined aeroplanes must have a factor of at least 2 under a torsional loading from the tail plane given by the formula.—

$$\text{Torsion} = \cdot 0007\, V_s^2\, c' D^2 \text{ (lb. in.)}$$

where $c' = \text{chord of tail plane (ft.)} = \dfrac{\text{Area}}{\text{Span}}$ (ft.).

D = airscrew diameter (ft.).

V_s = stalling speed of aeroplane (f.p.s.).

This is intended to cover unsymmetrical loading on the tail plane due to such causes as the rotation of the slipstream, spinning, etc. The greatest torsion from the tail plane of a multi-engined aeroplane with outboard engines is likely to occur when one of the outboard engines stops. To cover this case the fuselage, tail plane and tail plane attachments of such aeroplanes must have a factor of at least 2 under a torsion from the tail plane given by the formula.—

$$\text{Torsion} = \cdot 0025\, V_s^2\, c' \left[P\,(D - P) + s'^2 - \frac{D^2}{4} \right] \text{ lb. in.}$$

where $c' = \text{chord of tail plane (ft.)} = \dfrac{\text{Area}}{\text{Span}}$ (ft.).

s' = semi-span of tail plane (ft.).

D = diameter of airscrew of outboard engines (ft.).

p = perpendicular distance from plane of symmetry to thrust line measured at the tail (ft.).

V_s = stalling speed of aeroplane (f.p.s.).

The above formulae are for a monoplane tail. For a biplane tail 1·5 times the torsion given by the formulae should be used.

18. Engine mounting

(i) When the engine mounting forms part of the fuselage the stresses corresponding to the loading conditions described below (with N at least 6) should be followed through as far as the main planes. When the engines are situated in the wings the stresses corresponding to these loading conditions should be combined with those from the air load on the wings and followed through to the attachment of the wings to the fuselage, the value of N being that specified for the wings as a whole. The overriding requirement that N should be at least 6 applies in this case to the engine mounting structure only. Three loading cases are to be considered.—

(*a*) Turning in flight with engine on.

(*b*) Normal flight and landing with engine off.

(*c*) Side load. (*See* para. 19).

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* *Note.*—In some cases it may be advisable to take a more severe unsymmetrical load on the tail plane and elevators. Under these circumstances half the over-riding tail load of para. 7 should be applied to one side of the tail plane and zero load to the other.

(ii) *Turning.*—The structure is to have a factor of at least 1·0 under a combination of the following loads.—

(*a*) Gravity forces in level flight multiplied by N, where N is the specified ultimate factor on the wings in the C.P. forward case. If N is less than 6, the value 6 is to be taken.

(*b*) 2 × airscrew thrust and torque.

$$2 \times \text{thrust} = \frac{2\eta\, 550 \times \text{H.P.}}{Vs\sqrt{\dfrac{N}{2}}}\ \text{lb.}$$

$$2 \times \text{torque} = \frac{2 \times 33{,}000 \times \text{H.P.}}{2\pi \times \text{r.p.m.}}\ \text{lb. ft.}$$

where Vs is the stalling speed in f.p.s. ; N is the specified C.P. forward ultimate factor (the proviso in (*a*) above does not apply) ; H.P. is the brake horse-power of the engine when the aeroplane is flying horizontally at speed $Vs\sqrt{\frac{N}{2}}$; r.p.m. the airscrew revolutions per minute and η the airscrew efficiency, which in the absence of special data may be assumed to be 0·8.

(*c*) 2 × gyroscopic couple due to a banked turn without side-slipping at a speed $Vs\sqrt{\frac{N}{2}}$.

The gyroscopic effect is of importance for engines with large airscrews, and may be calculated as follows. The aeroplane is banked at an angle ϕ to the horizontal, such that

$$\cos\phi = \frac{2}{N}$$

N being the specified ultimate C.P. forward factor (the proviso under (*a*) above does not apply). 2 × the gyroscopic pitching couple is given by

$$2\,C_2 = \frac{2\,\text{I}^*\,\Omega\sin\phi}{Vs\sqrt{\dfrac{N}{2}}}$$

and 2 × the gyroscopic yawing couple by

$$2\,C_1 = \frac{2\,\text{I}^*\,\Omega\tan\phi\sin\phi}{Vs\sqrt{\dfrac{N}{2}}}$$

where Ω is the angular velocity of the airscrew in radians per second, I* is the polar moment of inertia of the rotating parts in lb. ft.^{2}, and Vs the stalling speed in f.p.s.

(iii) *Engine off.*—The structure should have a factor of at least 1 under gravity forces as given in (ii) (*a*) above.

(iv) In applying the above cases the aeroplane is assumed to be in the stalling attitude.

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* In the case of two-blade airscrews I is to be taken as twice the polar moment of inertia of the air screw (*see* Journal of R. Ae. Soc. Dec. 1934).

19. Side load case for engine mounting, front fuselage, seats, bomb racks, etc.

These components must have a factor of at least 1 under unit gravity loads acting alone and sideways, i.e. at right angles to the plane of symmetry of the aeroplane.

20. Control circuits (including tail plane adjusting gear and brake operating gear). *See* also Chapter III, para. 18.

A.—STRENGTH OF CONTROL MECHANISM.

(i) All controls must have a factor of at least $1 \cdot 3\dot{3}$ under each of the following conditions :—

(*a*) A pull or push on the top of the control column of 150 lb., applied to ring handles at the horizontal diameter, to handwheels equally at either end of the horizontal diameter, to a straight type of handle at the centre of the handgrip, i.e. in each case at the place at which the pilot would normally apply the force.

(*b*) A tangential force on the rim of the handwheel of 75 lb.

(*c*) A side load on the top of the control column of 75 lb. applied as in (*a*) above.

(*d*) A push on one side of the rudder bar of 300 lb. at the point at which the pilot would normally apply the force.

(*e*) A simultaneous push on each side of the rudder bar of 180 lb. at the point at which the pilot would normally apply the force.

(ii) *Pilots opposed case* (*dual control*).—When dual control is installed the loads given in (i) above must, in addition, be considered as being applied by both pilots simultaneously in opposite senses. The factor required on the components of the control system between the two pilots is $1\frac{1}{3}$.

(iii) Where mechanism giving a variable gear ratio forms a part of the transmission system, the strength of the system is to be calculated with the mechanism in the most adverse position.

(iv) In calculating the strength of the control system the loads should be carried through from the cockpit to the attachment of the control surface lever to the spar. The control surface itself will be designed for the specified aerodynamic loads.

(v) *Strength requirements for chains used in aeroplane control circuits* (*see* Chapter IV, para. 8.)

(*a*) The (unfactored) load in a chain forming part of the control system of an aeroplane is to be calculated for the conditions specified in the appropriate section of this paragraph. The size of chain fitted must be such that the greatest (unfactored) load determined as above does not exceed one-third of the breaking load of the chain.

(*b*) An ultimate load equal to three times the greatest (unfactored) load determined as above is to be quoted on the drawings for inspection purposes. In the case of chains supplied by approved chain makers, the ultimate loads stated by the chain makers will be accepted as evidence of satisfactory ultimate strength. For other chains it will be necessary to make a test to destruction on a sample cut from each separate length of chain supplied by the chain maker.

(*c*) All chains used in aeroplane control circuits, after attachment of their end fittings, are to be proof loaded with their end fittings to one-third of the ultimate load quoted on the drawings.

(*d*) The use of spring connecting links in roller chains is prohibited for all aeroplane controls. The only roller chains to be used are those which incorporate a positive method of attaching links.

(*e*) Chains will be accepted as complying with the automatic pilot and tail-to-wind requirements, provided their breaking strength is such as to give the ultimate factor specified for these two cases.

B.—Tail plane adjusting gear

(i) If the incidence of the tail plane is adjustable the operating mechanism must be irreversible. For the purpose of this requirement an irreversible tail operating mechanism is one in which movement of the tail plane relative to the fuselage under the influence of air forces is prevented by the gearing of the tail plane operating mechanism itself, and not solely by some locking device under the control of the pilot.

New methods of achieving tail plane adjustment will be considered on their merits, the application of the above requirements being the subject of agreement in each case.

(ii) The control circuit operating the tail adjusting gear is to have an ultimate factor of at least $1\frac{1}{3}$ under a pull and/or push of 75 lb. applied at the centre of the handgrip of the operating lever or tangentially to the rim of the operating wheel. If the operating lever is in such a position as to make it impossible for the pilot to exert this force, some smaller force, representing the greatest effort the pilot is able to exert, will be accepted.

C.—Brake operating gear

The strength requirements for this are as for the control mechanism given under Section A above, in so far as these are applicable.

* **21. Strength of aeroplanes under automatic control**

(i) Aeroplanes intended for use with automatic control are to have a factor of at least 1·33 under the loads which would be produced if at normal top speed (as defined in Chapter I, para. 5) the incidence were suddenly changed to (i) stalling incidence and (ii) a negative incidence corresponding to the first pronounced change in slope of the C_L curve which, for most aerofoils, can be taken to be — 15° (*see* para. 10 (ii)).

(ii) Aeroplanes fitted with automatic control are to have a conspicuous cockpit notice restricting the speed of flight under automatic control to normal top speed.

(iii) The cockpit speed restriction notice is to take the following form.—

"Maximum allowable speed under automatic control with cut-out disengaged or engaged: . . . . m.p.h. I.A.S."

(iv) In some cases (e.g. aeroplanes which have not been designed in the first place for use with automatic control) the factor of $1 \cdot 3\dot{3}$ under the conditions given in (i) above and the factor of 1·5 required in the down gust case given in para. 9, may not be realized. In such cases the speed under automatic control, with the cut-out disengaged, will be restricted to the speed V_1 m.p.h. I.A.S. at which the factor $1 \cdot 3\dot{3}$ is realized. With the cut-out engaged the speed will be restricted to V_2 m.p.h. I.A.S. where V_2 is 1/1·5 or 1/1·3 (whichever is appropriate to the type of aeroplane, *see* paras. 4 (ii) and 6 (ii)) times the speed at which the down gust requirements given in para. 9 are complied with or normal top speed, whichever is the less.

(v) In such cases (as mentioned in (iv) above) the cockpit speed restriction notice will take the following form.—

"Maximum allowable speed under automatic control
With cut-out disengaged................(V_1) m.p.h. I.A.S.
With cut-out engaged..................(V_2) m.p.h. I.A.S.
When under automatic control at speeds greater than V_1 m.p.h. I.A.S., the aeroplane must be in longitudinal trim for the appropriate speed."

* Previously A.D.M. 304.

(vi) The cut-out referred to in (iv) and (v) above is a device coupled to the elevator circuit to disconnect the automatic control if the elevator moves through more than a given angle from its position of trim. The device has to be reset for each position of trim, that is, for each speed of flight. Normally all automatic controls will be fitted with this device.

(vii) If V_2 estimated as above is less than V_1, then the same restricted speed, which will be V_1, will be adopted both with and without the automatic cut-out in operation. The cockpit notice should then read.—

> " Maximum allowable speed under automatic control whether the automatic cut-out is engaged or not.......... V_1 m.p.h. I.A.S."

22. Automatic control mechanism

(i) When automatic control is fitted the strength of the mounting of the instrument to the structure of the aeroplane and of the control system itself is to comply with the following requirement.

(ii) The mounting of the automatic control apparatus, together with any extensions from the normal control system to this apparatus, must have a factor of at least 1·33 under whichever is the more severe of the following two conditions.—

(*a*) The greatest force that the apparatus can apply (specified on S.I.S. for standard instruments; for others, to be obtained from the Royal Aircraft Establishment).

(*b*) The force exerted by the pilot from the cockpit, as specified in para. 20, carried through to the servo motor stops.

23. Wires cut

(i) *Main planes* (applicable to wings with external bracing wires only).—The structure is to have at least the factors scheduled in the following table when the specified wires are assumed removed.

Stressing case	Wires removed	Factor required
C.P. forward and C.P. back	Any one external lift wire or pair of duplicate lift wires attached to same anchorage.	Half the factor specified for the undamaged structure.
Up gust	Any one external lift wire or pair of duplicate lift wires attached to the same anchorage.	1.
Down gust	Any one external front anti-lift wire or rear lift wire, or pair of duplicate wires attached to the same anchorage.	1.
Terminal velocity dive ..	Any one external front anti-lift wire or rear lift wire, or pair of duplicate wires attached to the same anchorage.	1 or half the factor specified for the undamaged structure, whichever is the greater.

(ii) *Tail unit.*—The tail plane and fin structures are to have at least half the specified factors in all relevant loading cases when any one external bracing wire is removed. Duplicate wires will be dealt with as in (i) above.

(iii) *Fuselage bulkhead bracing.*—Fuselage bulkhead bracing is to have at least half the factor specified for the relevant loading case when one adjacent side panel wire, if any, is cut. The relevant loading case will be that which gives the greatest bulkhead load under these conditions.

24. Duplicate wires

When twin wires of the same strength are fitted in place of a single wire, the load is to be assumed to be distributed between the two wires in the ratio 2 : 1, i.e. each wire must be capable of taking two-thirds of the total load. When twin wires of unequal strength are fitted reference should be made to the Airworthiness Department.

25. Relative strength of lift and anti-lift wires

The vertical component of the strength of the anti-lift wires is to be not less than half the vertical component of the strength of the corresponding lift wires, irrespective of the loads in these anti-lift wires as calculated for the routine stressing cases.

26. Aerodynamic loading on long struts

The effect of air loads on external bracing struts is to be considered. Such struts are to comply with all strength requirements when carrying the most adverse probable combination of end load and aerodynamic lateral loading.

27. Windscreens (*see* also Chapter IV, para. 34)

(i) *Aeroplanes for which the terminal velocity dive case is a specification requirement.*—The windscreen is to have a factor of at least 2 under a uniformly distributed loading of

$$\frac{1}{2} \times 1 \cdot 0 \times 0 \cdot 002378\ V^2{}_1 \text{ lb./sq. ft.}$$

where V_1 is to be the lower of the following.—

(*a*) 660 f.p.s. I.A.S. (450 m.p.h.) ; or

(*b*) the estimated terminal velocity in f.p.s. (I.A.S.) assuming the airscrew drag to be zero.

(ii) *All other aeroplanes.*—The windscreen is to have a factor of at least 2 under a uniformly distributed loading of

$$\frac{1}{2} \times 1 \cdot 0 \times 0 \cdot 002378\ V_2^2 \text{ lb./sq. ft.}$$

where V_2 is the speed as defined in para. 6 (ii).

(iii) If satisfactory measurements are available showing that the maximum normal force coefficient on the windscreen in position on the aeroplane is less than 1·0, then this lower normal force coefficient may be taken instead of 1·0 in (i) and (ii) above.

(iv) *Method of checking compliance with the above requirements*

(*a*) *By calculation.*—Calculate the maximum bending stress in the windscreen glass assuming the glass to be a homogeneous simply supported beam of span equal to the shorter mean distance between the supported edges. Neglect any support along the edges parallel to the span of the beam so defined. The stress so calculated, corresponding to the factored load, should not exceed 5 tons per square inch.

(*b*) *By test.*—If the stress calculated as above exceeds 5 tons per square inch two specimens of the largest panel of the windscreen should be tested to the factored load (without 20 per cent. addition) of para. (i) or (ii) and if they carry this load successfully the windscreen may be approved. The edge supports of the test specimens should be fully representative, and care should be taken to see that the load is uniformly distributed. Loading by shot is liable to be inaccurate because of arching effects. The lower failing load of the two test specimens is to be taken. If the lower failing load is less than 80 per cent. of the higher failing load this should be reported to the Airworthiness Department.

(*c*) Curved surfaces will be similarly loaded and other materials will be considered on their merits.

28. Stiffness and strength of mass-balance weight arms and attachments

(i) Since at high speeds the rigid attachment of such weights to their control surfaces may be vital to the safety of an aeroplane, it is important that the stiffness and strength of balance arms and other fittings for the attachments of such weights should be adequate. Parts liable to become loose or to develop fatigue cracks under vibration should be avoided.

(ii) The strength of these fittings will usually be acceptable if they have a reserve factor of not less than 1 under the ultimate loads specified in each of the following stressing cases.—

Case 1.—Control surface in neutral position; mass balance weights and fittings subjected to the inertia forces corresponding to a normal acceleration of *Ng*, in either direction.—

" *N* " is the appropriate C.P.F. factor.

" Normal " acceleration is in the direction perpendicular to the main plane.

Case 2.—Control surface in neutral position; mass-balance weights and fittings subjected to the inertia forces corresponding to a lateral acceleration of 5*g*.

Case 3.—Mass-balance weights and fittings subjected to the inertia forces corresponding to an angular acceleration of the control surface about its hinge line of 500 radians/second^2.

For each case the parts stressed should include the balance arm and/or the local control surface structure to which the mass-balance weight is fixed.

CHAPTER III.—STRENGTH REQUIREMENTS BASED ON OTHER THAN FLIGHT CONDITIONS

1. Tabular summary

The requirements dealt with in this Chapter are summarized in Table I.

TABLE I.—SUMMARY OF STRENGTH REQUIREMENTS BASED ON OTHER THAN FLIGHT CONDITIONS

Note.—The contents of Chapters II and III have been completely rearranged by A.L. No. 3

Loading case	Factor required unless otherwise specified	Components for which the loading case will usually give design loads (subject to Chapter I, para. 4)	For description of case, *see* Chapter III
Energy absorption (landing tail up)—Landplanes.— With specified vertical velocity and vertical ground reaction not exceeding 3 *W*. (*W* = all up weight of aeroplane.)	—	—	Para. 2
Tyre loading : not to exceed *NDb* per wheel	Value of *N* as laid down in specification.	Tyre and wheel	Para. 3
Strength requirements for landing—Landplanes.—			
(i) Landing tail down—ground reaction vertical and equal to 3 *W*	1·33 on under carriage. 1·5 on remainder of structure	Undercarriage, fuselage	Para. 4 (i)
(ii) Side load at axle = *W* ÷ number of main landing wheels.	1 throughout ..	Undercarriage and centre section ..	Para. 4 (ii)
(iii) Landing tail up with vertical ground reaction of 3 *W* and total horizontal drag force of 0·75 *W* at hubs.	As for (i) above ..	Undercarriage and fuselage.. ..	Para. 4 (iii)
(iv) One wheel landing, tail down attitude—ground reaction vertical and equal to 1·5 *W*.	1 throughout ..	Undercarriage, fuselage main planes, and centre section.	Para. 4 (iv)
(v) Combined 4 *W* vertically upwards, 0·4 *W* horizontally sideways acting inwards, 0·35*W* acting outwards and 1 *W* horizontally backwards applied at axle. Both tail up and tail down attitudes are to be taken.	1 on undercarriage. 1·1 on remainder of structure.	Undercarriage, fuselage	Para. 4 (v)
Additional cases when wheel brakes are fitted.—			
(vi) Landing with brakes on and tail skid just clear of ground. Total ground reaction 4 *W* vertical and 1 *W* horizontally backwards.	1 throughout ..	Undercarriage, fuselage	Para. 4 (vi)
(vii) One wheel braked landing with tail down. Total ground reaction 1·5 *W* vertical and 0·375 *W* horizontally backwards.	1 throughout ..	Undercarriage, fuselage, main planes and centre section.	Para. 4 (vii)
(viii) Application of brakes to prevent aeroplane rolling backwards. Unit gravity loads and twice braking torque.	1 throughout ..	Undercarriage, brake mechanism ..	Para. 4 (viii)
(ix) Same as (v) except backwards load is applied at points of type contact	1 on undercarriage. 1·1 on remainder of structure.	Undercarriage, fuselage	Para. 4 (ix)
Strength requirements for landing—Seaplanes.—	—	—	Para. 5
Boat seaplanes.—			
(i) Landing tail up, with water reaction 3·5 *W*	1 throughout structure	Hull, main planes, engine mounting	Para. 6 (i)

Strength requirements for landing—Seaplanes—*contd.*			
Boat seaplanes—*contd.*			
(ii) Two wave landing, with water reaction $3 \cdot 5\ W$ ..	1 throughout structure	Hull	Para. 6 (ii)
(iii) Pressure over planing bottom	1	Hull	Para. 6 (iii)
(iv) Wing tip floats—side load of 150 lb./sq. ft. (not applicable to float structure).	1	Wing tip float attachments, main planes.	Para. 6 (iv)
Float seaplanes.—			
(i) Landing tail up, with water reaction $3 \cdot 5\ W$ ..	1 throughout structure	Floats, float undercarriage, fuselage, main planes.	Para. 7 (i)
(ii) Two wave landing, with water reaction $5\ W$ ..	1 throughout structure	Floats, float undercarriage, fuselage	Para. 7 (ii)
(iii) Side load $= \dfrac{W - f}{n}$ per float .. .. .. $f =$ weight of floats, $n =$ number of floats (not criterion for float structure).	1 throughout ..	Float undercarriage, main planes ..	Para. 7 (iii)
(iv) Pressure over planing bottom	1	Floats	Para. 7 (iv)
(v) Wing tip floats—as for boat seaplanes	1 throughout ..	Wing tip float attachments main planes.	Para. 7 (v)
Amphibians	As for landplanes and seaplanes.	—	Para. 8
Tail skid and tail wheel loads.—			
(i) Vertical reaction with aeroplane at rest on the ground. For tracking tail skid or wheel all possible positions should be considered.	4 on tail skid or wheel. $4 \cdot 5$ on remainder structure.	Tail skid or wheel, fuselage	Para. 9 (i) (*a*)
(ii) Vertical reaction as in (i) above combined with horizontal fore-and-aft drag force equal to half vertical reaction.	4 on tail skid, $4 \cdot 5$ on remainder of structure.	Tail skid, fuselage	Para. 9 (i) (*b*)
(iii) For unbraked tail wheel vertical reaction as in (i) combined with horizontal drag force of one quarter vertical reaction, resultant force passing through hub of wheel.	4 on tail wheel, $4 \cdot 5$ on remainder of structure.	Tail wheel, fuselage	Para. 9 (i) (*c*)
Catapulting.—			
(i) Acceleration, increment of velocity and component of wind plus ship speed as laid down in specification.	Usually 2	Fuselage, main planes, bomb racks..	Para. 10 (i)
(ii) Aeroplane at rest on catapult, engine off, under most adverse combinations of specified head and side winds (*a*) engine off (*b*) engine at full throttle.	2	Fuselage, centre section	Para. 10 (ii)
Arrested landing.—Speed, landing run and deceleration as laid down in specification	2	Fuselage, undercarriage ..	Para. 11
Slinging and handling loads	4	Centre section	Para. 12
Somersault landing.—			
(i) Leading edge of centre section (or pylon in the case of a low wing monoplane) and nose of aeroplane on ground	2	Centre section	Para. 13
(ii) Aeroplane fully over	4	Centre section	Para. 13
Salvage	6	Cables and parts of aeroplane connecting them to large masses in the fuselage such as engines, etc.	Para. 14

TABLE I.—SUMMARY OF STRENGTH REQUIREMENTS BASED ON OTHE N FLIGHT CONDITIONS—*contd*

Note.—The contents of Chapters II and III have been completely [illegible]ed by A.L. No. 3

Loading case	Factor required unless otherwise specified	Components for which the loading case will usually give design loads (subject to Chapter I, para. 4)	For description of case, *see* Chapter III
Jacking loads	3	Main planes, fuselage, centre section	Para. 15
Wings folded (tail on ground)	3	Main planes, centre section, undercarriage.	Para. 16
Static thrust and torque (applicable to both landplanes and seaplanes).	2	Engine mounting, undercarriage ..	Para. 17
Strength of control surfaces and systems under wind loads when the aeroplane is picketed or taxied tail-to-wind.—			
(i) For elevator and aileron surface and circuits. Applied hinge moment tending to depress each control surface.	Proof factor 1·25, ultimate factor 2.	Control surfaces and circuits ..	Para. 18
(ii) For aileron surfaces and circuits. Applied hinge moment tending to raise ailerons on one side and depress them on the other.	Same as (i) ..	Control surfaces and circuits ..	Para. 18
(iii) For rudder surfaces and circuits	Same as (i)	Control surfaces and circuits ..	Para. 18
Fixing of ballast weights and other large masses	—	Fuselage	Para. 19
Safety belts and harness .—			
(i) Pilots	Not less than 1 under $7g$ forwards and $\frac{Ng}{2}$ upwards.	—	Para. 20
(ii) Other members of crew	Not less than 1 under $2g$ forwards and $\frac{Ng}{2}$ upwards.	—	Para. 20
(iii) Seats and their attachments to the aeroplane, when belts and harness are anchored to the seat structure.	Greater than the above required factors.	—	Para. 20
Subsidiary structure (seats, bomb racks, generator attachments, fittings for stretchers, etc.).	As specified for adjacent main structure.	Subsidiary structure	Para. 21
Lateral stability of spars when unsupported over a length of two rib spacings—applicable to both main planes and tail planes.	As specified for all main and tail plane loading cases.	Main planes, tail unit	Para. 22
Ancillary equipment	4	—	Para. 23
Ribs (both main plane and tail unit). Strength by mechanical test.	As for main plane or tail unit.	Ribs	Para. 24
Openings in wing coverings	Loads due to openings to be provided in the design of the ribs.	—	Para. 25
Beaching chassis and tail trolleys	3	—	Para. 26
Fitting of ring cowlings	—	—	Para. 27

2. Energy absorption (landing tail up)—Landplanes

The undercarriage shock absorption capacity is to be such that the resultant ground reaction shall not exceed three times the weight of the aeroplane fully loaded when the aeroplane lands with the specified vertical velocity, thrust line horizontal. In addition to the ground reaction there will be air forces on the wings and tail, distributed as in the normal flight C.P. forward case and equal in total magnitude to 1 W. The vertical forces on the aeroplane resulting from the specified loads are therefore.—

(*a*) The ground reaction NW upwards at the wheels.

(*b*) The lift 1 W upwards at the wings and tail.

(*c*) Inertia and gravity forces $(N + 1)\,W$ downwards, the resultant passing through the C.G.

The angular velocity and angular acceleration of the aeroplane about all axes may be neglected. The airscrew thrust is to be assumed zero. Under these conditions the factor on the undercarriage, must not be less than 1·33 and on the remainder of the structure must not be less than 1·5 (*see* para. 4). In the event of a bad landing it is required that alighting gears shall fail before the main structure, and factors are specified with this end in view. Actually minimum factors are detailed, but the same margin of excess strength of the structure (and the attachment fittings of the alighting gear) over the alighting gear itself must be given whether or not the factor for the alighting gear is kept down to the specified minimum. This point needs special attention during development, when alighting gears may become strengthened up without the attachment fittings and remaining structure being adequately modified.

3. Tyre loading*

The maximum weight of the aeroplane in pounds divided by the number of main landing wheels is not to exceed $N \times$ product of wheel and tyre diameter in inches, where N is as specified (*see* also Chapter IV, para. 15).

4. Strength requirements for landing—Landplanes (*See* also para. 9).

These requirements may need amplification for special types of undercarriage. Aeroplanes whether fitted with undercarriage wheel brakes or not are to have the factors specified in Table II below in loading cases (i) to (v). Aeroplanes fitted with wheel brakes are, in addition, to have the specified factors in loading cases (vi) to (ix). In all cases W = total weight of aeroplane.

(i) *Tail down.*—The aeroplane landing on an even keel so that the tail wheel and undercarriage wheels touch the ground simultaneously. Resultant ground reaction vertical and equal to 3 W and distributed between the wheels of the undercarriage and the tail wheel so as to give zero moment on the aeroplane as a whole. Air loads on main planes and tail plane to be neglected. Airscrew thrust zero.

(ii) *Side load.*—A side load equal to $\frac{W}{m}$ applied in the same direction, to the hub of each of the m main landing wheels. This is the only external force acting on the aeroplane, it being balanced by the inertia forces introduced by the resulting linear and angular accelerations.

(iii) *Tail up.*—Landing tail up as described in para. 2 with vertical ground reaction equal to 3 W and total horizontal drag force, applied at the wheel hubs, equal to 0·75 W.

(iv) *One wheel landing.*—Total ground reaction at point of tyre contact equal to 1·5 W vertically upwards, distributed equally between all the wheels on one side of the centre line of the aeroplane, the ground reaction at the wheels on the other side being

**Note.*—Requirements for tyres and wheels are not given in this handbook. Lists of approved tyres and wheels may be obtained on application to the Airworthiness Department.

zero. The wing span is assumed to be horizontal and the tail skid resting on the ground. This case represents the unsymmetrical load which may arise in a one-wheel landing. This unsymmetrical load will give rise to angular and linear accelerations. It will usually be an acceptable approximation to consider only angular accelerations about the X (body) axis.

* (v) *Combined vertical, backwards, sideways loads: tail up and tail down.*—Total ground reaction of 4 *W* vertically upwards, 0·4 *W* horizontally sideways acting inwards on one side, 0·35 *W* horizontally sideways acting outwards on the other side, applied at points of tyre contact with ground and total drag load of 1 *W* horizontally backwards, applied at the hubs. This case is to be met both when the thrust line is horizontal and when the tail skid or wheel is just clear of the ground, the aeroplane being assumed on an even keel. The vertical and drag loads are equally distributed between the main under-carriage wheels on either side of the aeroplane. The tyres are assumed to be fully compressed and the closure of the oleo leg is to correspond to the position at which maximum reaction is first developed in landing, with the proviso that the leg is not assumed more than half closed without the sanction of the Airworthiness Department.

When wheel brakes (*see* Chapter IV, para. 14) *are fitted, the following additional requirements are to be complied with.—*

(vi) *Tail down, with brakes.*—The aeroplane landing with brakes on and tail skid or wheel just clear of the ground. Total ground reaction at points of tyre contact 4 *W* vertically upwards and 1 *W* horizontally backwards, distributed equally between all the braked wheels of the main undercarriage. When calculating the braking torque corresponding to the force of 1 *W* horizontally backwards a tyre deflection corresponding to normal load (i.e. wheel load with aeroplane at rest on the ground) is to be assumed. A smaller backward force than 1 *W* with appropriate torque may, however, be taken where the type of brake is such that the maximum torque which can be generated is limited to a known value, provided that twice the braking torque corresponding to full use of brakes gives rise to a smaller backward force at the point of tyre contact than 1 *W* when the vertical reaction is 4 *W*. In these circumstances twice the maximum possible braking torque may be used in strength calculations instead of the torque due to a tangential force of 1 *W* at the point of tyre contact. Overall balance of forces will be obtained by introducing horizontal and vertical inertia forces. Balance of moments will be obtained by introducing an aerodynamic load on the tail plane. This tail load will not be a criterion for the strength of any part of the structure.

(vii) *One wheel landing, with wheel brakes.*—Total ground reaction at point of tyre contact equal to 1·5 *W* vertically upwards and 0·375 *W* horizontally backwards, distributed equally between all the braked wheels on one side of the centre line of the aeroplane, the ground reaction at the wheels on the other side being zero. The wing span is to be assumed horizontal and the tail skid or wheel resting on the ground. The yawing moment is to be balanced out by an arbitrary side force applied at the tail end of the fuselage.

(viii) *Backward load, with brakes.*—Application of brakes to check the aeroplane rolling backwards, due to wind or slope of ground. The aeroplane is assumed to be in the tail down attitude and is subjected to unit gravity loads combined with twice the braking torque corresponding to a coefficient of friction between the tyre and the ground of 0·5. This requirement applies both to aeroplanes fitted with tail skids and to those fitted with tail wheels.

* (ix) *Combined vertical, backwards, sideways loads: tail up and tail down, with brakes* (backwards load applied at points of tyre contact instead of at axle).—This is the same as case (v) except that the backward load is applied at the points of tyre contact. A smaller

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* Previously A.D.M. 371.

backward force, at the point of tyre contact, than 1 W may be used in cases where twice the maximum braking torque gives rise to a backward load less than 1 W. In these circumstances the backward force, at the point of tyre contact, corresponding to twice the maximum braking torque, may be used instead of 1 W at the point of tyre contact. The strength requirements given above are applicable to all types of brakes. The maximum possible braking torque referred to in (vi) should, if possible, be measured rather than calculated. An approximate measurement can sometimes be made by jacking up one wheel and lashing a lever to the rim. A spring balance will then give the force at the end of the lever necessary to rotate the wheel when full braking effort is being applied at the cockpit.

TABLE II.—FACTORS REQUIRED FOR THE LANDING CASES OF LANDPLANES

Case	Required factor	
	Alighting gear	Remainder of structure
(i)	$1\cdot 3\dot{3}$	$1\cdot 5$
(ii)	$1\cdot 0$	$1\cdot 0$
(iii)	$1\cdot 3\dot{3}$	$1\cdot 5$
(iv)	}	$1\cdot 0$
(v)	}	$1\cdot 1$
(vi)	}	}
(vii)	} $1\cdot 0$	} $1\cdot 0$
(viii)	}	}
(ix)	}	$1\cdot 1$

5. Strength requirements for loading—Seaplanes

The factors specified in the following loading cases are, in general, applicable to seaplanes with a planing bottom keel angle of from 115° to 140° at the point of application of the water reaction. Evidence at present available is not sufficient to establish a satisfactory ...ion of specified factor with the planing bottom keel angle. Individual consideration wil[l] ...en to hulls and floats the planing bottom keel angles of which differ considerably from t... res given above.

6. Boat seaplanes

(i) *Landing tail up.*—The hull and aero-structure of boat seaplanes are to have a factor of at least 1 under the conditions of loading shown in fig. 1. The seaplane is assumed to be gliding into the water, the flight path being such that the main planes are at stalling incidence with the main plane chord horizontal. The seaplane is subjected to the following forces.

(*a*) A water reaction R equal to N_1 times the total weight of the seaplane, acting as shown in fig. 1. N_1 is to be taken as $3\cdot 5$ unless otherwise specified.

(*b*) Air forces. As para. 2.

(*c*) The weight of the seaplane acting vertically, i.e. normal to the chord of the main planes.

(*d*) Inertia forces which balance the reaction R and the couple G $(= R\,a)$ turning the seaplane about its centre of gravity.

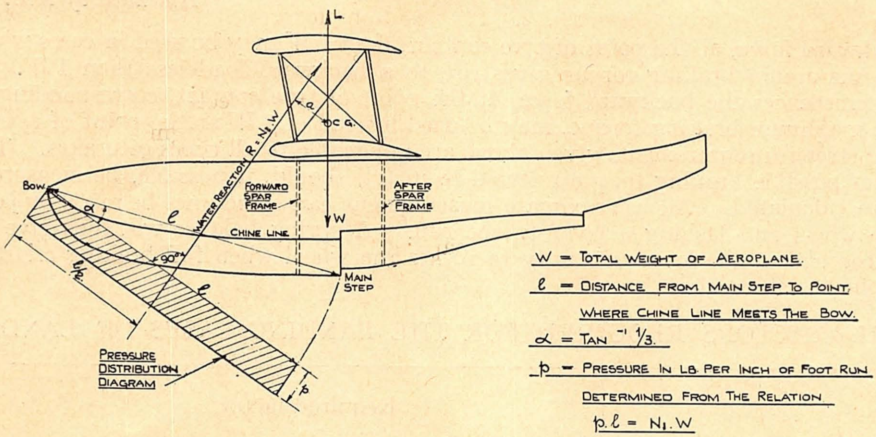


FIG. 1.—CHAP. III. Landing tail up: boat seaplanes.

The stresses in the structure are to be calculated corresponding to the instant of application of the force R. Thus the seaplane will have a linear and an angular acceleration, but the angular acceleration will not have had time to build up an angular velocity. Hence no centrifugal forces will be called into play. As regards the hull structure, this case is intended to check the strength of the hull forward of the forward spar frame in bending and shear, particularly in the neighbourhood of the point of attachment of the hull to the wing structure. In order that the bending moment and shearing force can be obtained at any section of the hull fore body, the water reaction R has been converted into a uniform pressure distribution extending from the bow to the main step as shown in fig. 1. This case is not intended to be any criterion for the strength of the planing bottom under water pressure, nor for any detail component of the hull remote from the point of attachment to the wing structure. The above system of loads will give rise to the forces on any component

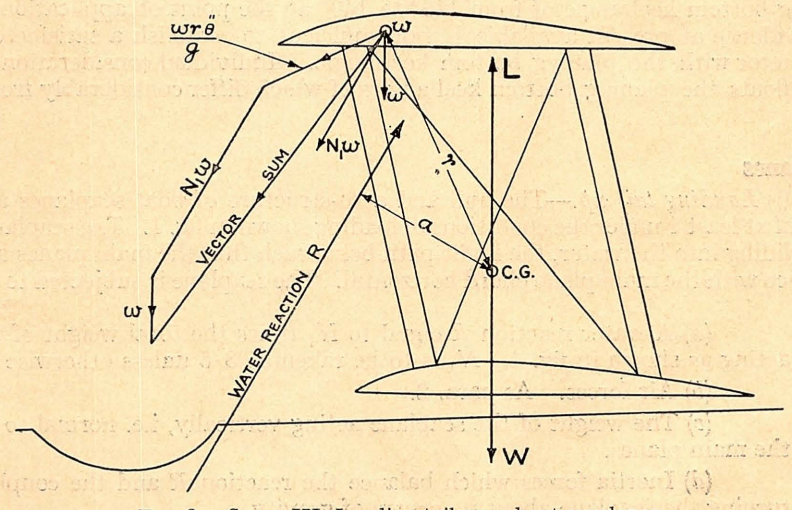
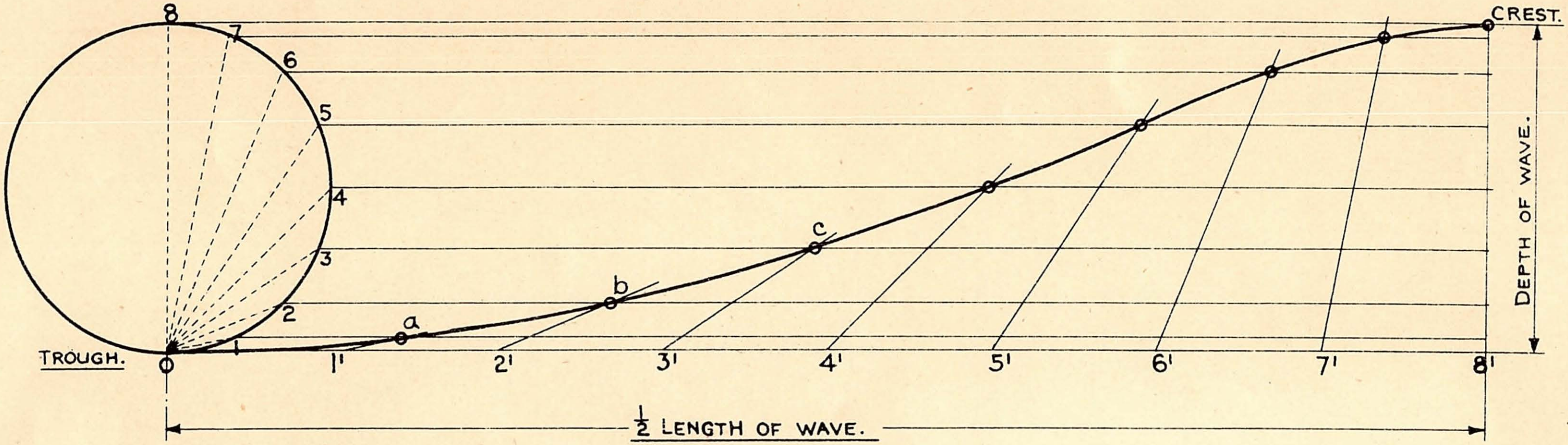


FIG. 2.—CHAP. III. Landing tail up: boat seaplanes.



METHOD:- DIVIDE HALF CIRCLE INTO ANY NUMBER OF EQUAL PARTS AT POINTS 0, 1, 2, 3.----------

JOIN 01, 02, 03 --------------

DIVIDE HALF LENGTH INTO SAME NUMBER OF EQUAL PARTS AT POINTS 0, 1', 2', 3'---------

THRO' POINTS 1', 2', 3'------ DRAW LINES 1'a, 2'b, 3'c ------ PARALLEL TO 01, 02, 03-----TO CUT HORIZONTALS THRO' 1, 2, 3 ------- IN POINTS a, b, c ---------

THEN a, b, c ------- ARE POINTS ON THE WAVE PROFILE.

FIG. 4. CHAP. III

METHOD OF CONSTRUCTING WAVE PROFILE. (TROCHOIDAL) mass of weight w shown in fig. 2. The resultant force on any such component mass is the vector sum of.—

(*a*) A force due to the linear acceleration of magnitude $N_1 w$ acting through the C.G. of the mass, opposite and parallel to the direction of R.

(*b*) A force due to the angular acceleration of magnitude

$$\frac{w\, r\, \ddot{\theta}}{g}$$

where r = distance, in feet, in side elevation, between the C.G. of the mass and the C.G. of the seaplane.

$\ddot{\theta}$ = angular acceleration $= \dfrac{Gg}{B}$

B = the pitching moment of inertia of the whole seaplane in lb. ft.^{2}

(*c*) Gravity force.

(ii) *Two-wave landing.*—The hull and aero-structure are to have a factor of at least 1 under the loading shown in fig. 3, the total water reaction being 3·5 times the total weight of the seaplane (i.e. $N_2 = 3{\cdot}5$).

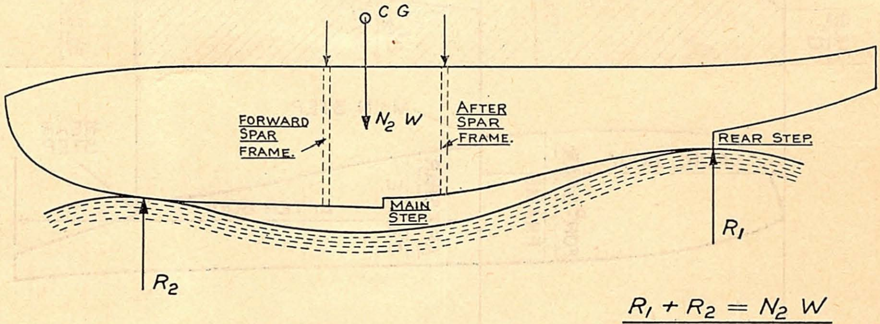


FIG. 3.—CHAP. III.—Two-wave landing : boat seaplanes.

The reaction R_1 is to be assumed to act at the rearmost step or, where only one step is fitted, at the point where the full load water-line at rest cuts the rear portion of the hull in profile. The point of application of the reaction R_2 is determined by drawing in a standard wave (*see* below), with one crest at the point of application of R_1, and rotating the hull about this latter point until the fore part of the hull touches the surface of the water as shown in fig. 3.

The standard wave is trochoidal in form (for method of constructing trochoidal wave, *see* fig. 4), with a length from crest to crest equal to the length of the hull on the still water-line at maximum load, and a depth from crest to trough equal to 1/15th of the length. The intention of this case is to check the strength of the hull forward of the forward spar frame and aft of the after spar frame in bending and shear, the hull being considered as a beam simply supported at the spar frames. For this purpose it is immaterial how the reactions R_1 and R_2 are distributed over the bottom surface of the hull. This case should also be applied to the aero-structure but it will usually be a determining case for the hull only.

(iii) *Pressure over planing bottom*

(*a*) The planing bottom plating and internal bottom structure of the hull are to have a factor of at least 1 under the pressure distribution shown in fig. 5. When no rear step is fitted, the after ordinate of the pressure distribution curve is to be set up at that point where the full load water-line at rest cuts the rear portion of the hull in profile.

(*b*) This type of loading will introduce inertia loads of the same type as those described for the landing tail up case (para. 6 (i)). In general, however, it will not be necessary to go into this complication as this case is intended only as a criterion for the plating, stringers, etc., of the planing bottom, and the inertia effects will be negligible for these components.

(*c*) The pressure is assumed to be normal to the planing bottom surface, and to be uniformly distributed laterally from chine to chine.

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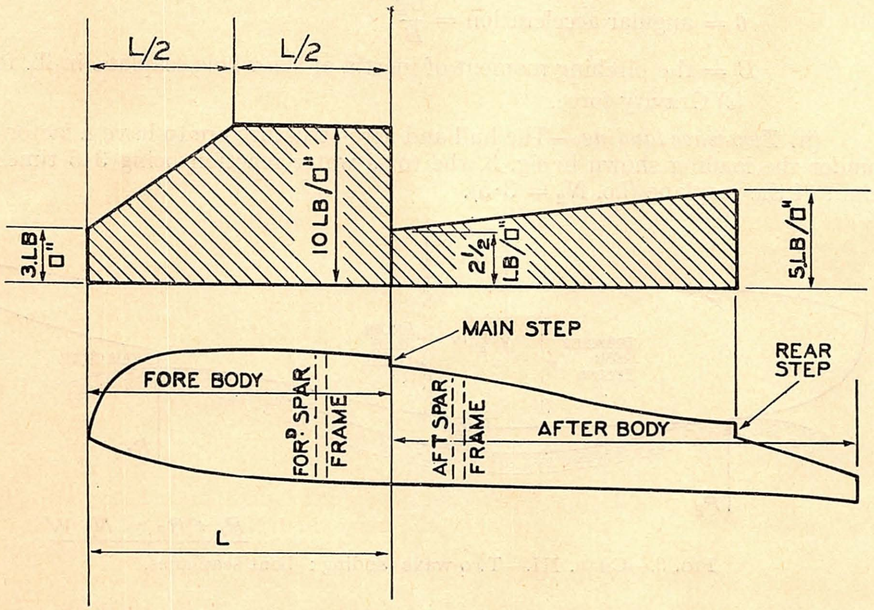

FIG. 5.—CHAP. III. Pressure over planing bottom : boat seaplanes.

(*d*) When applying this case full loading is to be used for checking the strength of the planing bottom plating and half loading for checking the strength of the internal bottom structure, viz. : keel, keelsons, stringers, bottom portion of transverse frames, etc.

(iv) *Wing tip floats, strength requirements under side load*

(*a*) The attachment of the wing tip floats and the adjacent main structure of the aeroplane must be capable of withstanding without failure a side load on the float of at least 150 lb. per sq. ft. of projected area of side elevation, the load acting at the centre of area of the side elevation in a direction parallel to the lower plane, and both towards and away from the centre line of the aeroplane.

(*b*) If a spring is incorporated in the mounting of the float so that the float can move relatively to the lower plane under a side load, the above figure of 150 lb. per sq. ft. may be reduced to 100 lb. per sq. ft.

(*c*) The above requirement does not apply to the structure of the wing tip float itself.

* Previously A.D.M. 337.

7. Float seaplanes

(i) *Landing tail up*

(*a*) The float, or floats, and the remainder of the structure of float seaplanes are to have a factor of at least 1 under the loading shown in fig. 6. This case is identical with the corresponding case for boat seaplanes (fig. 1) with the single exception that the water reaction R is as shown in fig. 6. N_1 is to be taken as $3 \cdot 5$ unless otherwise specified.

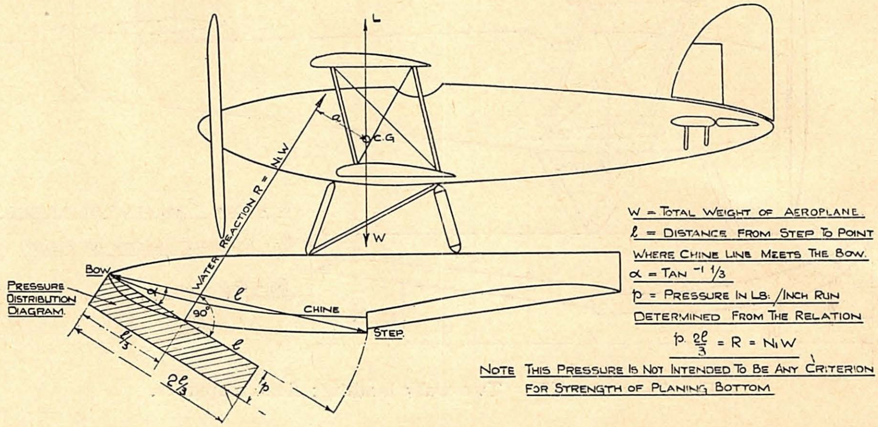

FIG. 6.—CHAP. III. Landing tail-up: float seaplanes.

(*b*) As regards the strength of the float structure, this case is intended to check the strength of the float as a whole in bending and shear, particularly in the neighbourhood of the point of attachment of the float to the forward undercarriage strut. It is not intended to be any criterion for the strength of the planing bottom under water pressure, nor for any detail component of the float remote from its point of attachment to the forward undercarriage strut. The full water reaction $R = N_1W$ must be used when checking the strength of the float structure.

(*c*) When considering the strength of the float undercarriage, it will usually be sufficiently accurate to consider forces (*a*) and (*c*) of para. 6 (i) only, making R equal to $N_1 (W - f)$ where $W =$ total weight of seaplane and $f =$ weight of floats. The weight of the floats acting perpendicularly to the chord of the main plane will relieve the load in the front undercarriage struts. This approximation involves neglecting the angular inertia of the floats themselves, and as this would usually be small and would be a relieving load on the struts in question, the approximation may safely be made.

(ii) *Two-wave landing*

(*a*) The float (or floats) and the remainder of the structure of float seaplanes are to have a factor of at least 1 under the loading shown in fig. 7. N_2 is to be taken as $5 \cdot 0$ unless otherwise specified. The attitude of the seaplane is that which it would take up when floating, fully loaded, at rest, in still water and the water reactions R_1 and R_2 are to be taken at $\frac{l}{6}$ from the bow and stern respectively, l being the overall length of the float.

(*b*) The intention of this case is to check the strength of the structure connecting floats to fuselage, and also the strength of the float(s) both forward and aft of the undercarriage struts, and at intermediate points between the strut attachments, in bending and shear, the float being considered as a beam simply supported at the undercarriage strut connections. For this purpose it is immaterial how the reactions R_1 and R_2 are distributed over the bottom surface of the float.

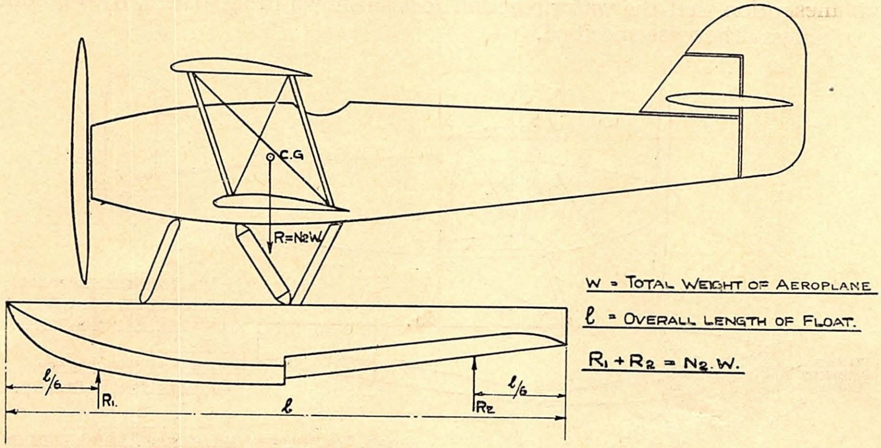

FIG. 7.—CHAP. III. Two-wave landing: float seaplanes.

(*c*) For strength of float structure the full water reaction $R = N_2W$ must be used, but for checking the strength of the undercarriage structure the weight of the floats may be deducted and the reaction taken equal to $N_2\ (W - f)$ where

W = all-up weight of seaplane.

f = weight of float or floats.

This case should also be applied to the aero-structure but it will usually be a determining case for float and structure connecting float to fuselage only.

(iii) *Side landing*

(*a*) The side load on each float is to be $\dfrac{W - f}{n}$,

where W = all-up weight of seaplane.

f = weight of floats.

n = number of main floats.

This force acts through the centroid of the float in side elevation. In the case of twin or multiple float seaplanes the side load is assumed to act simultaneously, and in the same direction, on each float. A factor of at least 1 is required.

(*b*) It should be noted that this side load is the *only* force acting, there being no vertical load.

(*c*) This case is not intended to be a criterion for the strength of the structure of the main float or floats.

(iv) *Pressure over float planing bottom.*—The planing bottom plating and internal bottom structure of each float are to have a factor of at least 1 under the pressure distribution shown in fig. 8. The application of this case is identical with the corresponding case for boat seaplanes, i.e. half the specified loading only need be taken for checking the strength of internal bottom structure.

(v) *Wing tip floats. Strength requirements under side load.*—As specified for boat seaplanes.

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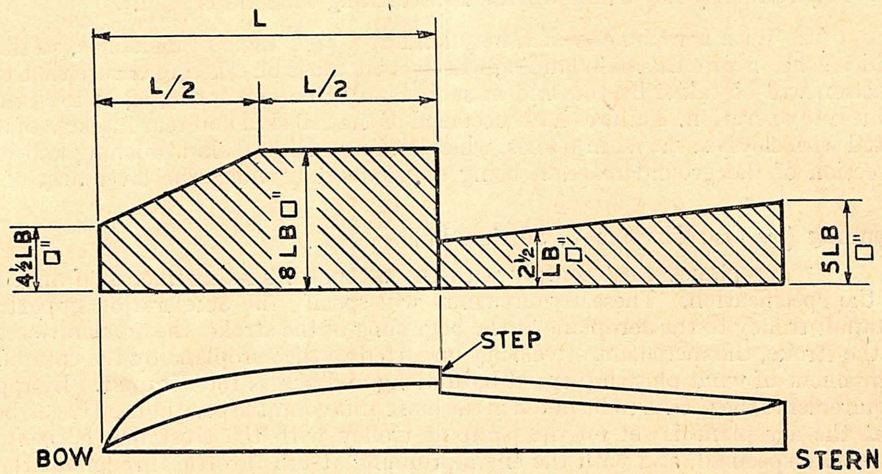

FIG. 8.—CHAP. III. Pressure over planing bottom : float seaplanes.

8. Strength requirements for landing—Amphibians

These are to fulfil the appropriate combination of requirements given in paras. 4, 6 and 7 above.

9. Tail skid and tail wheel loads (*see* also para. 4)

(i) The tail skid or wheel and remainder of the structure must comply with the following requirements.—

(*a*) The tail skid or wheel and its attachment fittings are to have a factor of at least 4, and the remainder of the structure a factor of at least 4·5, when the aeroplane is at rest on the ground. If a tracking tail skid or wheel is fitted this requirement should be fulfilled for all possible positions of the tail skid or wheel.†

(*b*) The tail skid and its attachment fittings are to have a factor of at least 4, and the remainder of the structure a factor of at least 4·5, when the vertical reaction at the tail skid, found as in (*a*) above, is combined with a horizontal fore-and-aft drag force of half the amount of the vertical force. If a tracking tail skid is fitted it is to be taken in its central position.

(*c*) If an unbraked tail wheel is fitted (*b*) above applies except that a horizontal drag force equal to a quarter of the vertical reaction is to be taken, the resultant force passing through the hub of the wheel.

(ii) The tail skid may be called upon to withstand a considerable side force when the aeroplane is turning or taxying over a rutty aerodrome. It is left to the discretion of the designer to make adequate provision for such side forces, no specific requirements being considered warranted at present. When there is any doubt as to the strength of the tail skid and fuselage under side load, the pilot who carries out the approval flight trials will be

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* Previously A.D.M. 337. † *See* footnote to para. 16, Chapter II.

instructed to carry out such taxying manœuvres as will, in his opinion, prove the adequacy or otherwise of the structure under this type of loading, provisional clearance only being given pending such tests.

(iii) *General design considerations for the tail skid.*—It is important that some lateral springing should be provided for the tail skid, in addition to the usual vertical springing. This is particularly important when a non-tracking tail skid is used.

(iv) With aeroplanes which may land at a high angle of incidence, so that the tail skid touches before the main landing wheels, care must be taken to ensure that the ground reaction will be taken by the skid in such a way that the springing of the tail skid will come into operation. Failures have occurred of the tail skid and rear fuselage of aeroplanes fitted with slots on the main planes, when landing with tail skid touching first, due to the direction of the ground reaction being approximately along the lever arm of the skid.

10. Catapulting (*see* also Chapter V, Section IV)

(i) Aeroplanes must comply with the catapulting requirements, if any, laid down in the specification. These requirements will specify the acceleration imparted by the catapult trolley to the aeroplane at the beginning of the stroke, the acceleration at the end of the stroke, the increment of velocity imparted to the aeroplane by the catapult and the component of wind plus ship speed both along and across the catapult. In applying the requirements the C.G. is to be taken in the most unfavourable position and it is to be assumed that the aeroplane is set on the catapult trolley with the trust line horizontal, unless otherwise specified, and with the engine running at full throttle throughout the catapult stroke. Provision is to be made for the most unfavourable combinations of acceleration and front and side winds. A factor of 2 is usually required.

(ii) In addition to those specified above, all aeroplanes which are to be catapulted, must comply with the following requirements.—

(*a*) Aeroplane at rest on the catapult cradle. A factor of at least 2 is required under the most adverse combinations of specified head and side winds with engine off.

(*b*) As in (*a*) above with engine at full throttle.

11. Arrested landing

Certain aeroplanes are required to be designed to withstand the loads induced by a mechanical retardation of the motion after landing on the deck of an aeroplane carrier. The factor throughout the structure in these circumstances must not be less than 2. The speed of the aeroplane at the instant when the arresting hook engages the wire, the length of the landing run and the deceleration caused by the arresting device will be laid down in the specification.

Note.—Details of a side load case can be obtained on application to the Airworthiness Department.

12. Slinging and handling loads

Aeroplanes are required to have a factor of at least 4 under the loads incurred when hanging freely by their slinging gear. The sling when supporting the aeroplane fully loaded is to have the same factor.

13. Somersault landing

The centre section structure is to be sufficiently robust to ensure that the wings will not collapse on to the cockpit and imprison or severely injure the crew should the aeroplane turn over on its back when alighting. Two cases are to be considered. These are illustrated in fig. 9

Amended by A.L. No. 3

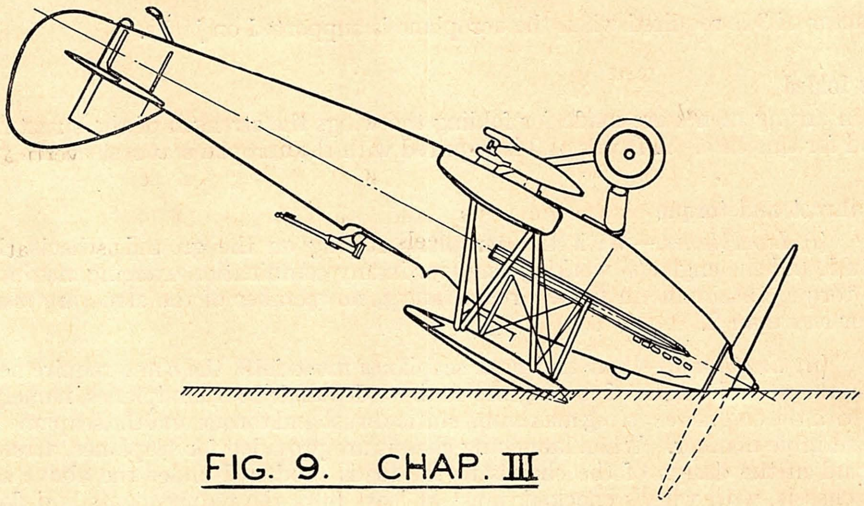

FIG. 9. CHAP. III

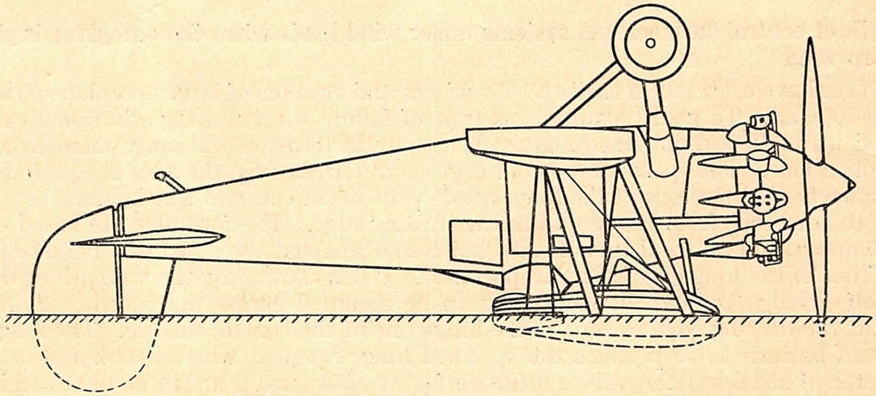

FIG. 10. CHAP. III

Somersault landing.

(leading edge of centre section and nose of aeroplane on ground) and fig. 10 (aeroplane fully over). The loads incurred in these cases are dependent on the precise attitude assumed by the aeroplane when inverted. It does not appear to be possible to specify these attitudes in general terms applicable to all aeroplanes and accordingly each aeroplane must be dealt with on its merits. In the case of a low wing monoplane, a pylon structure must be provided for this purpose in the absence of any alternative protection for the crew. The factors required are not less than 2 for the case shown in fig. 9 and not less than 4 for that shown in fig. 10, based on the fully loaded weight of the aeroplane supported as indicated.

14. Salvage

Salvage cables and those parts of the aeroplane connecting them to large masses in the fuselage such as engines, etc., are required to have a factor of not less than 6 under the loads due to the weight of the aeroplane fully loaded.

15. Jacking loads

A factor of 3 is required when the aeroplane is supported on jacks.

16. Wings folded

When arrangements are made for folding the wings the strength of the structure is to be investigated for this case. A factor of 3 is required with the aeroplane at rest on the ground.

17. Static thrust and torque

(i) *Landplanes.*—With the aeroplane resting on the ground, wheels chocked or braked, and the engine(s) working singly or in any combination exerting maximum thrust and torque obtainable in these circumstances, no member of the structure must have a factor less than 2.

(ii) *Seaplanes.*—Boat and float seaplanes must fulfil the same requirements under static thrust and torque of the engines as those demanded for landplanes, namely, a factor of 2 with the engines exerting maximum static thrust and torque and the seaplane restrained in a suitable manner. When launching chassis are provided for seaplanes, attention must be paid in the design of the chassis to the loads produced under the above conditions. The chassis, with wheels chocked, must at least fulfil the requirements laid down above for the seaplane itself.

18. Strength of control surfaces and systems under wind loads when the aeroplane is picketed or taxied tail-to-wind

Failures have occurred in the control surfaces and systems of large aeroplanes when on the ground tail-to-wind. To guard against this type of failure the following additional cases are to be complied with. The requirements given below are in terms of the hinge moment applied to each control surface. The specified moment is given in terms of S_1, the area in square feet of one control surface behind the axis of the hinges and c_1 the mean chord in feet behind the axis of the hinges, i.e. the distance from the hinges to the trailing edge. The formulae are based on a wind speed of 40 m.p.h. and on wind tunnel tests which indicate that the effect of a balance portion of aileron in front of the hinge is of small importance. When estimating the strength of the control surface itself, triangular load distribution is to be assumed across the chord, varying from a maximum at the trailing edge to zero at the hinge line of the control surface. The magnitude of the load is to be such as to produce the specified hinge moment when distributed in this way. A proof factor of at least 1·25 and an ultimate factor of at least 2 are to be obtained in each of the following three loading cases.—

Case (i).—The aileron and elevator control surfaces and circuits are to have the specified factor when a hinge moment H_1 is applied to each control surface in a direction tending to depress the surface.

$$H_1 = 3c_1S_1 \text{ lb. ft.}$$

For the aileron system the moments on each aileron will balance out through the balance circuit. In the case of the elevator circuit the strength of the system is to be examined both when the reaction is taken by a locked control column and when, with column free, the elevator has moved downwards to the extent allowed by the stops.

Case (ii).—The aileron control surface and circuit are to have the specified factors when a hinge moment H_2 is applied to each aileron in a direction tending to raise the ailerons on one side of the aeroplane and to depress them on the other.

$$H_2 = 2c_1S_1 \text{ lb. ft.}$$

This case is to be investigated both for a locked control column and for a free column displaced to the full amount allowed by the stops.

Case (iii).—The rudder control surface and circuit are to have the specified factors when a hinge moment H_3 is applied to the rudder.

$$H_3 = 4c_1S_1 \text{ lb. ft.}$$

This requirement is to be fulfilled both with the rudder bar locked central in the cockpit and with the rudder bar free to move to the extent allowed by the stops.

* In cases where the control circuits are locked at the surfaces, the above requirements are to be complied with in order to provide for release of the locks by the pilot whilst taxying (*see* Chapter IV, para. 10).

19. Fixing of ballast weights and other large masses (*see* also Chapter IV, para. 38)

(i) *Ballast weights.*—The fastenings of ballast weights and adjacent structure must be designed to withstand a deceleration of 10*g* applied in a direction parallel to the thrust line. This requirement will be waived for ballast weights placed so that no injury to occupants of the aeroplane is likely should the weights break loose in a crash. The attachments of such ballast weights need only be of the same strength as the main structure to which they are anchored.

(ii) *Other large masses.*—Satisfactory anchorages are to be provided for all such items of equipment as would, in the event of a heavy landing or crash, be liable to break from their positions and injure occupants of the aeroplane. The strength of the anchorages is to be at least as great as that of the structure to which they are attached under forward inertia forces acting parallel to the thrust line.

20. Safety belts and harness

(i) (*a*) *Belt or harness for pilot(s), including pupil (if any)*.—The attachments must be capable of carrying the ultimate loads due to the following accelerations acting separately or together on the pilot.—

$$7g \text{ forwards}$$
$$\frac{Ng}{2} \text{ upwards.}$$

(*b*) *Belt or harness* (*see* sub-para. (ii) (*g*) below) *for all other members of the crew.*—The attachments must be capable of carrying the ultimate loads due to the following accelerations acting separately or together on the crew.—

$$2g \text{ forwards}$$
$$\frac{Ng}{2} \text{ upwards.}$$

(ii) *Notes.*

(*a*) N is the specified C.P. forward ultimate factor, or twice the factored acceleration in the specified down gust case, whichever is the greater.

(*b*) The forward direction is defined by the airscrew axis. The upward direction is at right angles to this.

(*c*) The pilot and crew are to be assumed to weigh 200 lb. each if wearing a seat type parachute ; otherwise 180 lb. is to be assumed.

(*d*) Proper provision is to be made for carrying the loads through to the main airframe. It is desirable that the airframe should be stronger than the attachment lugs so that the latter would break first.

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* 653655/37.

(*e*) If the belt or harness is attached to the seat, and if the wearer normally sits upon his parachute, then the strength of the connection between seat and airframe is to be greater than that between belt and seat. This requirement arose out of an accident in which the seat connections broke and the pilot was thrown out still attached to the seat and consequently unable to use his parachute.

(*f*) Requirement (i) (*b*) applies to belt and harness attachments in rotating gun turrets. The loads are to be taken through the rotating mechanism on to the main airframe.

(*g*) Gunners and other members of the crew are not necessarily belted down into their seats the whole time as their duties require them to move about. The restraint when the belt is not used is by a wire or adjustable strap carried down to a strong point in the floor. The requirement of paragraph (i) (*b*) applies also to this strong point.

21. Subsidiary structure

Seats, bomb racks, generator attachments, fittings for stretchers, etc., must have at least the same factor as the main structure in all the appropriate stressing cases. Local loads at points of attachment to the main structure should be carefully considered to ensure that the main structure is not unduly weakened thereby.

22. Lateral stability of spars when unsupported over a length of two rib spacings

When the construction of main planes and tail planes is such that failure of a single rib by accidental damage or otherwise increases the length of the spar without lateral support, then such main planes and tail planes must comply with all specified strength requirements, with full factor, when the spars are entirely unsupported laterally over a length of two rib spacings the unsupported length being in the most adverse position.

23. Ancillary equipment

The ultimate factor for platforms or similar parts included in the aeroplane structure and for any item of ancillary equipment supplied by the contractor for lifting or supporting an engine, freight, personnel, etc., is to be not less than 4. If the item is so constructed that its strength cannot be accurately calculated from existing data it is to be subjected to a proof load equal to twice the maximum static load it is required to withstand. The inscription " " Safe for loads up to (here insert the maximum static load figure) " is to be painted or otherwise marked on each such item at a clearly visible spot.

24. Ribs

(i) The strength of the main plane ribs is to be demonstrated by a mechanical test reproducing with reasonable accuracy the load distribution along the chord corresponding to the various flight conditions considered. If the spars to which the rib is attached are particularly flexible in the plane containing the spar minor axis, the rib test should be arranged to give an end load and/or torque along the inter-spar length of the rib representing the stabilizing force exerted by the rib on the spar.

(ii) A mechanical test for tail plane ribs will only be necessary when these ribs differ considerably from types whose strength characteristics are known.

(iii) These mechanical tests are to conform to the conditions stated in Chapter I, para. 3. Correction down to the standard component conditions will seldom be possible so that the 20 per cent. extra factor expedient will usually have to be adopted.

(iv) Vibration tests on metal ribs will also be required under approved conditions.

25. Openings in wing coverings

The existence of openings in the coverings of a wing may effect the static pressure within the wing to such an extent that the normal rib loads are exceeded very considerably. The making of permanent openings in the coverings of aeroplane wings (or control surfaces) is therefore to be avoided. Should this not be possible, the appropriate loads must be provided for when designing the ribs, so that the factor under such conditions shall not fall below the required standard. Alternatively, the form of construction at points where openings occur is to be such that the passage of air to or from the interior is prevented.

26. Beaching chassis and tail trolley of boat seaplanes (*see* also para. 17)

The main beaching chassis and tail trolley of a boat seaplane are to have an ultimate factor of 3 with the boat seaplane at full normal load under the following conditions.—

(i) when standing on a level concrete apron ;

(ii) when the main wheels of the chassis have just passed over the crest of a $7\frac{1}{2}°$ slipway.

(iii) when standing on a $7\frac{1}{2}°$ slipway with the main wheels chocked.

27. Fitting of ring cowlings

Aerodynamic forces on ring cowlings obtained from wind tunnel experiments do not form a sufficient criterion for calculating the necessary strength of the ring and its attachments. Other loads such as those due to turbulence of the air behind the airscrew, engine vibration, landing shocks, etc., not susceptible to precise calculation will come upon the ring. The ring and its attachments should therefore be examined from a general engineering point of view and should be given ample strength in this respect as well as sufficient strength to take the maximum estimated anti-drag forces arising in terminal velocity conditions. The ring attachments to the engine should make provision for expansion.

CHAPTER IV.—NON-FACTOR REQUIREMENTS

1. Tabular summary

The requirements given in this Chapter are summarized in Table I. All of these requirements are to be complied with unless specifically waived in the aeroplane specification.

TABLE I

Requirement	*See* Chapter IV, para.
Prevention of wing aileron flutter	2
Prevention of undue control circuit stretch and of aileron instability ..	3
Prevention of tail flutter	4
Identification markings on control surfaces	5
Duplication of control circuits	6
Cables in control systems	7
Chains in control systems	8
Bearings in control systems	9
Locking of controls	10
Stability and control of aeroplanes in which automatic controls may be used	11
Tail unit control surfaces :	
(i) Divided elevators	12
(ii) Clearance between fin and rudder	12
Rudder power	13
Wheel brakes	14
Undercarriage wheels and tail wheels	15
Castings	16
Welding of steel parts	17
Streamline wires and tie rods	18
Flattening of ends of tubes for the purposes of attachment	19
Sweating and drilling steel tubing to B.S. Specification T.2	20
Use of 4 B.A. bolts	21
Use of even sizes of B.A. screws	22
Use of tab washers	23
Design of wiring lugs	24
Lugs for external wire bracing	25
Sections of aeroplane metal parts (produced from hot-formed sections or forgings)	26
High-tensile steel fittings	27
Bending of aluminium alloy sheets and strips (applicable to material to B.S. Specification 4 L.3 and L.38 and to D.T.D.270 and 275)	28
Permissible bow in light alloy tubing for use in aeroplane structures ..	29
Corrosion of S.61 and S.62 bolts in wooden members subject to wetting in service	30
Collars for high-tensile steel pins	31
Provision of longitudinal datum marks	32
Safe limit of deterioration of shock absorber legs	33
Securing of windscreens..	34

TABLE I—*contd.*

Requirement	*See* Chapter IV, para.
Fasteners for cowling and inspection doors	35
Compression shakes in spruce rib flanges	36
Use of duralumin tubes and sheet thinner than 22 gauge	37
Provision for ballast	38
Use of parallel pins	39
Attachment of wireless aerials (fixed and trailing)	40
Retractable undercarriages	41
Undercarriage springing characteristics	42
Torsional stiffness of ailerons	43
Torsional stiffness of elevators	44
Repairs to mass-balanced surfaces	45
Ground clearance	46
Hand and foot holes in airframes	47
Buoyancy of engines in estimating buoyancy of aeroplanes	48
Protection of aeroplane cowlings and structure from gun blast	49
" Closed " structures (provision for internal inspection and removal of riveting pin or mandrel heads)	50
Emergency exits in aeroplanes	51

2. Prevention of wing aileron flutter (*see* also Chapter V, Section V)

(i) Careful consideration is to be given to the feasibility of complying with the anti-flutter recommendations of R. & M. 1155, articles 9 and 9a and 1255, articles 51 and 52. In particular, all ailerons must be mass-balanced.

(ii) For an aileron on a given wing, mass-balance is defined as follows.—Let m be an element of aileron mass at a perpendicular distance x, measured in the plane of the wing, from the aileron hinge axis and a perpendicular distance y from a longitudinal axis in the plane of the wing parallel to, and usually* coincident with, the chord at the wing root. Let x be positive when m is behind the hinge axis and y positive when m is outboard of the longitudinal axis. Then the aileron should be regarded as mass-balanced when the following conditions are each separately fulfilled :—

(*a*) The product of inertia Σmxy of the mass-balanced aileron in its " C.G. back " condition is zero or negative both when the aileron is neutral and when it is displaced $\pm$ 10°.

(*b*) The centre of gravity of the mass-balanced aileron in its " C.G. forward " condition is not forward of the hinge line, measured parallel to the wing chord, by more than $0 \cdot 06 c_a$, where c_a is the mean chord of the aileron behind the hinge line, both when the aileron is neutral and when it is displaced $\pm 10°$.

In the above the C.G. back condition of an aileron is to be taken as that obtaining when the aileron is in a condition as regards fabric and doping weights, trimming strips, repairs, etc. corresponding to a most rearward position of the aileron C.G. ; and the C.G. forward condition corresponds to the opposite practicable extreme. For biplanes, the weight of

* When the inboard portion of the wing can be shown to be abnormally stiff, either inherently or by virtue of exceptionally stiff external supports, it may be permissible to locate the longitudinal axis elsewhere than at the wing root.

any inter-aileron member should, for the purpose of calculating Σmxy for the ailerons, be halved and the halves treated as two isolated masses fixed to the upper and lower ailerons respectively at the points of attachment of the inter-aileron member. The finished covering of a control surface supplies an important contribution to the total mass. It is important, therefore, in the above C.G. back case to arrange mass-balance to be adequate when the weight of the finished covering is a maximum. The aileron to be mass-balanced should be weighed both covered and uncovered, and the weight of the finished covering obtained. This latter should then be corrected up to the maximum likely to be encountered during the life of the aircraft and this value used in the above C.G. back case. If the surface is covered with doped fabric a weight of at least 10 ounces per square yard of fabric (i.e. a total of 20 ounces per square yard of control surface area) is to be taken. Ailerons should not be painted (*see* para. 5). If mass-balancing is effected by means of a bob-weight this should be placed well outboard of any point of external support to the wing.

(iii) In exceptional cases compliance with the above obligatory requirement may not be a sufficient safeguard. When this appears probable, compliance with one or more of the other recommendations of R. and Ms. 1155 and 1255 may be required.

* 3. Prevention of undue control circuit stretch and of aileron instability

(i) *Stiffness of aileron control circuits.*—Unless the control and/or circuit is of unconventional type, the aileron control circuit of an aeroplane must be such that the hinge stiffness, h, in lb. ft./radian, is not less than the values given in fig. 1, Chap. V, Sect. VI,

(*a*) by the upper curve for upward movement of the aileron,

(*b*) by the lower curve for downward movement of the aileron, for the value of the control circuit factor, $\frac{S_a C_a V^2}{10^6}$ appropriate to the particular aileron. These requirements are to be met when measurements are made as in Chapter V, Section VI, and are irrespective of any tendency of the aileron to blow up or down under normal flight conditions. In the above.—

S_a = aileron area aft of hinge line (sq. ft.),

C_a = mean aileron chord aft of hinge line (ft.) both S_a and C_a being measured in the plane of the undersurface of the particular aileron, and

V = maximum speed (ft./sec.) to which the aeroplane is required to be dived during contractor's and official flight trials.

For a monoplane, only one aileron need be considered. For a biplane one upper and, if present, one lower aileron must be considered and each must individually satisfy (*a*) and (*b*) above. If the lateral control and/or the circuit is of unconventional type—as, for example, a circuit with irreversible units—reference should be made to the Airworthiness Department so that the case may be dealt with on its merits.

(ii) *Stiffness of elevator and rudder control circuits.*—The elevator and rudder control circuits must—unless the circuit is of unconventional type—be such that the percentage stretch for each of the circuits (S_3 for elevator, S_4 for rudder), when measured as described in Chapter V does not exceed 20 per cent. If an elevator or rudder control circuit is of unconventional type—as, for example, a circuit with an irreversible unit—reference should be made to the Airworthiness Department so that the case may be dealt with on its merits.

* Previously A.D.M. 343.

(iii) *Slack and friction in control circuits.*—Attention is drawn to the need for minimizing slack and friction in control circuits. The fit of all control bearings should be as close as possible consistent with ease of manipulation of the controls, and the areas of the bearings should be adequate to prevent rapid wear leading to the development of slackness. A control circuit should be designed so that its operation when unloaded does not involve stretch, deformation or slackness in any part of the system.

(iv) *Aileron and trimming strip settings.*—Standard settings for the ailerons and their trimming strips (if any) are to be stated in Volume I of the Air Publication relevant to the particular aeroplane. For each aileron the standard setting stated is to correspond to the upper limit of permissible positions for the trailing edge of the unloaded aileron relative to the wing, any unavoidable slack in the control circuit being assumed taken up by gently raising the aileron trailing edge, the controls being centralized. For each trimming strip the standard setting stated is to correspond to the lower limit of permissible positions for its trailing edge relative to the ailerons.

It must be demonstrated during the contractor's and official flight trials of the type aeroplane, that the ailerons and trimming strip (if any) are such that, when in their standard setting positions, no overbalance or other instability is observable in the aileron system for all conditions of flight to which the type aeroplane may be subjected. This applies to aeroplanes with no trimming strips, to aeroplanes with trimming strips on the ailerons on both sides of the aeroplane and to aeroplanes with trimming strips on the ailerons on one side of the aeroplane only. In the last case the contractor must make any special provisions that may be necessary for correcting the lateral trim during the flying trials.

It is further required that no adjustment of a trimming strip to a position below its standard setting shall be possible. For this purpose a positive stop must be fitted.

(v) *Construction and adjustment of aileron trimming devices.*—All trimming devices are to be constructed so that they are sufficiently stiff and securely held to withstand the loads occurring in flight and during normal handling and maintenance operations.

It is required that the aeroplane drawings and Volume I of the Air Publication relevant to the particular aeroplane shall include as much information as is practicable concerning the method of adjustment of aileron and trimming strip settings. The information given in Volume I of the relevant Air Publication should include the guidance necessary for the satisfactory replacement of a damaged aileron by a new or repaired one.

4. Prevention of tail flutter (*see* also para. 12 (i))

(i) *Rudder.*—Careful consideration is to be given to the feasibility of complying with the anti-flutter recommendations of R. & M. 1255, articles 85 and 85a. In particular the rudders of all aeroplanes with the possible exception of rudders incorporating auxiliary surfaces, are to be mass-balanced.

(ii) For the purpose of this requirement rudder mass balance is defined as follows : Let m be an element of rudder mass whose perpendicular distances from the rudder hinge axis and above the fuselage axis of torsion are x and z respectively. Then the rudder is mass-balanced if Σmxz for the complete rudder is small (i.e. between about $\frac{1}{8}$th and $\frac{1}{12}$th of the magnitude of the product of inertia of that part of the rudder aft of the hinge axis) and negative (*see* Chapter V, Section V).

(iii) x is to be considered positive for elements of mass aft of the rudder hinge axis. On conventional types of rudder, i.e. rudders the greater part of which is above the fuselage axis of torsion, z is to be considered positive for elements of mass above the fuselage axis of torsion. Other types of rudders will be dealt with individually, and particulars of such rudders for which approval is desired should be forwarded to the Airworthiness Department.

(iv) Present evidence indicates that if rudders incorporating auxiliary surfaces are mass-balanced in the following way they will be free from flutter. Hence this method of mass-balancing main and auxiliary surfaces is to be adopted unless permission is obtained to waive the requirement or to adopt a different method.

(*a*) Arrange for $\Sigma m_a x_a z$ for the auxiliary surface alone to be small and negative, where x_a is the distance of an element m_a of auxiliary surface mass aft of the auxiliary surface hinge line ;

(*b*) Arrange for $\Sigma m_a x_a (d + x_a)$ to be small and negative, where d is the distance between the hinge axes of main and auxiliary surfaces ;

(*c*) Arrange for $\Sigma m_a x_a{}^2$ to be as small as possible ;

(*d*) Arrange for Σmxz of the main and auxiliary surfaces combined to be small and negative, x being measured aft of the main rudder hinge ;

(*e*) Arrange for Σmx^2 for the main and auxiliary surfaces combined to be as small as practicable.

(v) In fulfilling the above requirement all masses present in the finished rudder system, such as tail lamps and their fittings, fabric covering, dope and paint, are to be taken into account. The weight of dope and paint is variable and it is important that the maximum probable weight of the covering should be assumed in each case. For unpainted rudders the lowest weight of doped fabric to be assumed in complying with the above requirement is 10 ounces per square yard (i.e. a total of 20 ounces per square yard of control surface area). For a painted surface 25 ounces per square yard (50 ounces per square yard of control surface area) is the minimum. If the figure appropriate to the doped surface is used steps must be taken to ensure that the rudder will not subsequently be painted (*see* para. 5).

(vi) *Tail structure.*—It is important that the tail plane and fin should be very stiff. In particular the tail plane should be very stiff in torsion and flexure and its attachment to the fuselage, through the trimming gear or otherwise, should be as rigid as possible.

(vii) In order to prevent tail flutter due to relative movement between an elevator and its trimming tab, the following conditions must be fulfilled.—

(*a*) The trimming tab control circuit must be irreversible ; the irreversible unit or units in the circuit should be placed as close to the tabs as possible.

(*b*) The trimming tab control circuit, particularly those parts of it between the irreversible units and the tabs, must be stiff and as free from backlash as possible. On the latter point special care should be taken in the design of the tab hinges.

(*c*) Each trimming tab must be stiff in torsion and flexure and the number of hinges ample to prevent any material bending of the tab between the hinges under air loads.

(*d*) The design of the trimming tab control circuit must be such as to prevent the development of any differential motion between the tabs on the port and starboard elevators.

(viii) Consideration should be given to the mass-balancing of elevator trimming tabs as an addition to the preventive measures enumerated above. Such mass-balancing would appear particularly desirable when any doubt is felt as to the adequacy with which these other measures may have been met. Where mass-balancing is decided upon, this should be arranged so that the C.G. of the trimming tab is on, or slightly ahead of, its hinge line when the tab is in its neutral position.

5. Identification markings on control surfaces

(i) It has been the practice hitherto to paint identification markings on the rudders of aeroplanes and on the wings in accordance with S.I.S. No. 4. The " targets " on the wings have been painted in such a manner that an appreciable proportion of them lies on the ailerons.

(ii) The weight of paint thus carried on control surfaces is in some instances of material importance when the mass-balancing of these control surfaces is under consideration.

(iii) It has therefore been decided that the identification marks on rudders are to be entirely eliminated, and the identification marks on the wings are to be of smaller diameter, so that the ailerons and slats are no longer involved. Other identification markings will remain as before.

(iv) Action is being taken to ensure that the spaces so left vacant are not employed for squadron markings after the aeroplanes are delivered.

(v) In designing control surfaces for mass-balance, therefore, it is to be assumed that the weight of paint hitherto involved by identification markings and for squadron markings need no longer be taken into consideration.

(vi) It is pointed out that the aeroplane number will still be painted on the rudder in the location given in S.I.S. No. 4.

* **6. Duplication of control circuits**

When a servo control surface is fitted the control system must be arranged so that in the event of failure of the servo surface or its attachments, it will be possible to retain some measure of control over the aeroplane by direct operation of the control surface concerned.

7. Cables in control systems

(i) Cables in control systems are to be so arranged that they are easy to adjust and easy to replace. Turnbuckles, other adjustment points, fairleads and other points in the circuit which might give difficulty when replacing a cable should be placed in positions where they are readily accessible and visible, an inspection panel being fitted if necessary.

(ii) Where a cable is led round a pulley, joints in the run of the cable may be inserted to facilitate replacement of the portion which travels over the pulley.

(iii) Cables should run as straight as possible. If fairleads have to be used these should be of dry red fibre type.

(iv) The layout of the control system should not involve the splicing of cables in position.

(v) All flying control cables should be proof loaded, after all splicing operations are completed to 50 per cent. of the nominal strength of the cable. Save in exceptional circumstances this should be done before the assembly of the cable in the control system. Where a splice *in situ* cannot be avoided the proof load test must still be applied if practicable.

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* Previously A.D.M. 368.

(vi) The effective length of cables should not change during the operation of controls.

(vii) The standard approved method of splicing cables is as shown in S.I.S. No. 3 Alternative methods of splicing may be approved provided that the alternative type of splice is found to be of equal strength.

(viii) Provision is to be made to render mechanically impossible the assembly of control systems with reversed connections. Any suitable and effective method may be used. The following are possible methods.—

(*a*) Attachment fittings provided with diffierent sizes of pins.

(*b*) Attachment fittings designed to assemble in one position only.

(*c*) Lengths of cable and positions of fairleads arranged to make incorrect assembly impossible.

(*d*) Use of levers and rods instead of cables.

8. Chains in control systems (*see* also Chapter II, para. 20 A (v))

When chains and sprockets are used in the control system, steps are to be taken to en that the chain is guided on to the sprocket and guarded in such manner that it is impossible for the chain to jam or over-ride its sprocket, even when the chain is completely slack. The guards must be so arranged that loose articles such as screws and small nuts inadvertently left in the neighbourhood of the sprocket cannot enter the guard or cause the chain to jam. The guards must be readily detachable for inspection purposes.

9. Bearings in control systems

Precautions are to be taken to eliminate slack in control systems. To this end the fit of all bearings must be as close as possible consistent with ease of manipulation of the controls, and the bearing surfaces must be large enough to prevent rapid wear.

10. Locking of controls

* If provision is made for locking the controls of an aeroplane to prevent the control surfaces flapping when the aeroplane is unattended on an aerodrome or at moorings, the locking device is to be arranged so that the pilot cannot sit in his seat and attempt to take off so long as the elevator, rudder and aileron controls are locked. (This requirement does not apply to aeroplanes of the Fleet Air Arm type on which the control surfaces must be locked in accordance with S.I.S.369 when the aeroplane is stowed on deck or on a catapult.) Locking devices must be capable of being stowed away when not in use so that they do not restrict the pilot in any way, and so that they cannot accidentally obstruct the use of the controls. Should it be found necessary to introduce locking devices in the control system to prevent the control surfaces from flapping about during taxying, it is essential that such locking devices should be kept in operation only by continuous physical effort on the part of the pilot, e.g. by means of a push button on the control column.

11. Stability and control of aeroplanes in which automatic controls may be used

(i) In the specifications of certain types of aeroplane provision for fitting automatic controls is called for. In what follows an attempt is made to define as generally as possible the characteristics required of an aeroplane in order to obtain the best results under automatic control compatible with satisfactory manual control. Any such definition must of necessity be tentative until further experience has been obtained of automatic controls in various aeroplanes.

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* Previously A.D.M. 368.

(ii) The requirements may be summarized as.—

(*a*) an approximation to neutral stability both longitudinally and laterally with controls fixed;

(*b*) the dominance of self-centring moments over the friction of the control system.

With regard to (*a*), an aeroplane possessing these characteristics will be subjected to a minimum initial angular disturbance when the direction of the relative wind is altered either by bumps or by the use of the controls, for example, in turning "flat." Thus the displacement acquired before the automatic controls can apply the necessary corrections will be reduced to a minimum. As regards (*b*), when an automatic control is working satisfactorily in calm air, minute movements of the servo-motors are taking place. These movements could be accommodated by the stretch of the control cables and the slack in the system without any corresponding movement of the control surfaces if the friction of the system were sufficient to cause the control surfaces to "stick" when displaced. The result would be that the deviation of the aeroplane would increase until the servo-motor movement were sufficient to take up the slack and strain, when the control movement would be excessively large. In this way the aeroplane would be caused to "hunt" instead of settling down in equilibrium with the automatic controls. Satisfactory working depends upon a minute movement of the servo-motor being sufficient to cause a minute movement of the control surface.

Design requirements

(iii) The aeroplane should be so designed as to obtain the maximum advantage from the use of automatic controls consistent with safe and easy manual control.

(iv) *Feel.*—All controls should be self-centring when disturbed in flight. This requires that the hinge moment resulting from a displacement of the control surface from its trimming position by not more than $\frac{1}{4}°$ should be sufficient to overcome the friction in the control system when the normal amount of vibration is present. If this condition is satisfied, the control will have satisfactory "feel" for small movements under manual control. For large control movements the amount of balance required will be determined by the consideration that the aeroplane must be controllable by manual effort.

(v) *Longitudinal stability.*—The aeroplane should be as nearly "neutral" as possible for a condition of normal loading but should not be longitudinally unstable when the C.G. is on the aft permissible limit. These conditions refer to the state of the aeroplane when the elevator control is held fixed. Stability with controls free is not required for automatic control, but the fact that the aeroplane must also be flown under manual control makes it essential that the aeroplane should not be dangerously unstable with the elevators free.

(vi) *Lateral stability.*—The rolling moment due to side-slip should be as small as possible consistent with the consideration that the aeroplane may have to be landed at night under manual control. (Of course from general considerations the aeroplane must not be actually unstable in this respect.) The characteristic is desirable for precision stabilization whether the control is manual or automatic. It is to be expected that reduction of rolling moment due to side-slip will require closer limits to the permissible value of yawing moment due to side-slip with rudder fixed. As for manual control, the yawing moment due to side-slip should be slightly stable (i.e. N_V positive and small). For automatic control this condition is required only when the rudder is fixed. For manual control it is desirable either that it should hold with the rudder free or alternatively that some means of applying a spring constraint to keep the rudder in the trimming position should be available when the automatic control is not in use. This device must not produce any interference with the smooth operation of the rudder when precision flying is required.

12. Tail unit control surfaces

(i) *Divided elevators.*—Sections of elevators in the same plane must be rigidly connected together. No form of indirect linking will be accepted (*see* also para. 4 (vi)).

(ii) *Clearance between rudder and fin.*—The clearance between the rudder and the fin should be sufficient to preclude fouling even though the structure becomes slightly distorted under Service conditions, or as the result of delayed tautening effect of the dope used.

13. Rudder power

The rudder power of all landplanes must be at least 10 unless spinning model tests have shown that a particular design is satisfactory with a rudder power smaller than this. Rudder power is defined as.—

$$\text{Rudder power} = \frac{1000\,N}{0{\cdot}002378\,S\,s\,V^2}$$

where N = maximum yawing moment (lb. ft.) about Z (body) axis due to the rudder at aircraft incidence of 25° and at speed V (lb. ft.). N is to be measured on a complete model at 25° incidence from zero lift, at zero yaw.

S = area of the wings (sq. ft.).

s = semi-span of the wings (ft.).

V = wind tunnel speed at which N is measured (ft./sec.).

Certain features of design may make it necessary to provide a greater rudder power than 10. Such features are.—

Wings.—Thick section ; monoplane ; stagger less than 15° forward ; gap abnormally small.

Rudder.—Where liable to excessive shielding when the aeroplane is stalled.

14. Wheel brakes (*see* Chapter III, para. 4)

Wheel brake installations are to comply as far as possible with the following requirements :—

(i) The controls for operating the brakes are to be so designed as to enable the pilot to apply the brakes on the port and starboard sides of the aeroplane in unison and with equal power, or independently of one another for the purpose of steering.

(ii) In cases where aeroplanes are fitted with a tail wheel it is essential that some form of " parking " brake be provided of sufficient strength to hold the aeroplane without the use of wheel chocks for the purpose of running up the engine. On aeroplanes provided with a tail skid a " parking " brake is not regarded as absolutely essential, but it is considered very desirable that it should be provided. The use of tail wheels in lieu of tail skids is generally desirable from the point of view of minimizing damage to aerodrome surfaces, and it is hoped that the general introduction of wheel brakes will enable the use of tail wheels to become more general.

(iii) The action of the brakes must be progressive, that is to say, must increase and decrease in propotion to the force applied by the pilot to the brake control. In particular, the brakes must cease to act instantly with removal of force from the brake control. The sensitivity of the brake control should be such that the pilot is enabled to judge by the force which he is applying how much braking effect he is using, and he should be enabled to arrive at this judgment before the wheel actually touches the ground during the landing manœuvre.

(iv) The system employed for operating the brakes must be easy to adjust and maintain and must not require frequent attention. Appropriate access must be provided to all parts which require adjustment, maintenance and inspection. From the maintenance point of view the order of preference ascertained in the course of comparative trials was as follows :—

(*a*) Oil operation ;

(*b*) Pneumatic operation ;

(*c*) Cable operation.

(v) The operation of both brakes acting in unison may be controlled by hand or by the feet. If the lay-out is such that operation is by hand, then the time during which the hand has to be removed from the throttle and/or control column to apply or release the brakes must be the minimum possible. In the case of aeroplanes whose aerodynamic controls are so heavy that both hands are needed normally to hold the control column back after landing, any form of hand control used for applying the brakes in unison on landing must be fitted on the control column or wheel. Hand levers used for the purpose of the " parking " brake may be separate from the normal braking control, and the above comments in regard to hand operation need not apply to the control for the " parking " brake.

(vi) For the differential operation of the brakes for the purpose of steering, it is preferred that control should be by the use of the feet direct on the rudder bar. If the controls for the brakes move with the rudder bar, they must be adjustable with it on the general lines in which rudder bars are adjustable to suit pilots of different stature. The use of pedals for controlling the brakes is regarded as generally undesirable, but if they are used it is essential that they should be of ample size and provided with a form of surface which will prevent pilots' feet from slipping. The use of pedals operated by the heel is to be avoided.

(vii) The above guidance relates to the system of control employed for operating the brakes. In addition to this attention is drawn to the fact that arrangements are now in hand for the standardization of brake drums, both as regards diameter and location relative to the centre plane of the wheel. It is, therefore, essential that where standards have already been fixed for brake drums they should be adhered to, and where sizes of wheel are being used for which standard brake drums do not yet exist, consultation with the Air Ministry should be arranged with a view to ascertaining whether a standard is about to be fixed.

15. Undercarriage wheels and tail wheels (*see* also Chapter III, para. 3)

(i) *Main undercarriage wheels.*—Lists of approved wheels together with allowable loads are to be obtained on application to the Airworthiness Department. The lists of approved wheels include tail wheels. The operational load is defined as the weight of the aeroplane fully loaded divided by the number of main undercarriage wheels.

(ii) *Tail wheels.*—The operational load is defined as the load on the tail wheel when the fully loaded aeroplane is at rest on a level surface with the C.G. at the aft limit.

16. Castings

Castings may be used in aeroplanes subject to compliance with the following conditions.—

(i) *Class 1 castings.*—Castings used in such a position that a single failure might cause collapse of the structure or loss of control, either in flight or when landing or taking off. The specified ultimate factor for such castings is twice the ultimate factor specified

* for the aeroplane as a whole, and every casting must be radiologically examined, as laid down in A.I.D. Inspection Instruction No. M.429 except as may be agreed under paragraph (vi) below.

(*Note*.—To avoid confusion, a casting which exactly fulfils these conditions should be referred to as having a reserve factor of 1 not 2. *See* definitions of reserve factor in Chapter I, para. 5.)

(ii) *Class 2 castings*.—Stressed castings used in such a position that a single failure would not cause collapse of the structure or loss of control, either in flight or when landing or taking off. The first castings of a given part from each source of supply are to be radiologically examined, as laid down in A.I.D. Inspection Instruction No. M.429, to prove the suitability of the foundry methods, pattern and alloy. If this radiological examination shows all the castings to be satisfactory, thereafter only 2 per cent. of the castings need be radiologically examined, the balance being subjected to individual visual examination for general condition, as laid down in the Inspection Instruction.

(iii) *Class 3 castings*.—Castings, the strength of which does not contribute to the strength of the primary structure or control system. Such castings will be subjected to individual visual examination only.

(iv) Drawings of castings are to be endorsed to indicate the classification of each casting and the method of inspection agreed, so as to enable the appropriate inspection procedure to be followed.

(v) Subject to the agreement of the Director of Aeronautical Inspection first being obtained, when the castings are such that complete radiological examination presents great difficulty, the " break up " procedure, as detailed in A.I.D. Inspection Instruction No. M.429 may be adopted as an alternative to radiological examination under Class 1 or Class 2 above.

17. Welding of steel parts

The following requirements refer to oxy-acetylene welding but proposals to use electric welding will be considered if they are put forward.

(i) *General requirements*

(*a*) The welding processes will be carried out to the satisfaction of the Director of Aeronautical Inspection, in accordance with the methods laid down by him.

(*b*) Welded joints will be permitted only in positions where failure of any one welded joint will not involve the collapse of the structure or cause the pilot to lose control of the aeroplane. This requirement may, subject to the concurrence of the Director of Technical Development, be waived in certain cases of repairs to damaged members, when the nature of the repair provides a considerable margin of strength over the strength of the original undamaged member. Special consideration will also be given to cases in which the area of the weld is large in relation to the load to be carried and the form of the joint is such as to ensure a good weld.

(*c*) Strength calculations on welded joints are to be based on the minimum strength in the welded and (if applicable) heat-treated condition. If this strength is not given in the specification it must be obtained from material tests. In no case, however, must the strength taken for calculation exceed the minimum specified strength in the unwelded condition.

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* Previously A.D.M. 336, issue 2.

(*d*) Local application of heat is not permissible for heat-treatment purposes. All structures and fittings, etc., shall be heat-treated after welding except in the following three cases.—

Case 1.—Where the drawings state that heat-treatment can be omitted, provided that the drawings have been approved by the Director of Technical Development. Such approval will only be granted when both.

1. It is considered that the design of the part renders heat-treatment impracticable, and

2. The part is made from steel to one of the following specifications.—

Sheets and strips ..	S.3, D.T.D.12, D.T.D.124, D.T.D.141.
Tubes	T.26, T.35, T.45, D.T.D.41, D.T.D.178.
Bars	S.21, D.T.D.126.

Case 2.—Where the parts have been made from non-corrodible steels to Specifications D.T.D.171, 176 and 207.

Case 3.—Parts made from non-corrodible steel to Specification D.T.D.166 which have been edge or spot-welded (the strength of the steel in the vicinity of the weld will be reduced to that of steel to Specification D.T.D.171).

(*e*) Where the part is to be heat-treated after welding, any aircraft steel to a British Standard or D.T.D. Specification may be used, provided that the approval of the Director of Technical Development is obtained if the steel is not to one of the specifications quoted in sub-para. (*d*) above.

(ii) *Welding rods, etc.*

(*a*) All the steels mentioned in sub-para. (ii) (*d*) *Case* 1 must be welded with iron or mild steel wire to Specification D.T.D.82 or with a rod or strip of approximately the same composition as the material being welded.

(*b*) Non-corrodible steels must be welded with rods or strips of approximately the same composition as the material being welded, or with rods to Specification No. D.T.D.61.

18. Streamline wires and tie rods

(i) *Precautions to minimize breakage.*—Compliance with the following recommendations, which will reduce the probability of these wires breaking in extended service, is to be satisfied.—

(*a*) Avoid values of length/diameter ratios in streamline wires greater than 400. Length is defined as the "A" length in B.S. specifications, irrespective of intermediate support such as by means of an acorn. Diameter is the diameter of the rod used for making the streamline wire, referred to in B.S. Specification 5.W.3 as " size of wire " or " size ".

(*b*) Provide centre anchorages for streamline wires wherever possible. The anchorage should be such as to prevent both torsional and flexural movement.

(*c*) Avoid designs of end fitting which give fixation greater than that of the standard fork end.

(*d*) Avoid structural arrangements in which normal deformations may impose bending loads near the ends of a streamline wire or tie rod, e.g., ample clearance should be provided if the wire passes through a hole in a bulkhead near the end of the wire.

* (ii) In order to reduce the number of different sizes of fork ends which have to be stocked, only the following sizes of streamline wires and tie rods to B.S. Specifications 5.W.3 and 5.W.8 are to be used in future in new aeroplane designs.—

4 B.A., 2 B.A., 8/32, 10/32, 12/32......
(even sizes expressed as thirty-seconds).

(iii) *Lock-nuts*

(*a*) For streamline wires and tie rods not made of stainless steel, brass or cast-iron lock-nuts only are to be used. On seaplanes, ship-planes and amphibians brass lock-nuts only (A.G.S. 672 and 673) are to be used.

(*b*) For stainless steel streamline wires and tie rods stainless steel standard thin lock-nuts to B.S.I. Specification 4 A.1 are to be used.

19. Flattening of ends of tubes for the purpose of attachment

When the ends of tubes are flattened a liner tube must first be inserted, the liner having a serrated inner end and a length of three diameters of the containing tube. The gauge of the liner must be less than that of the original tube. The complete flattened end should be made solid by approved methods (brazing, sweating, etc.).

20. Sweating and drilling steel tubing to B.S. Specification T.2

The strength of tubes to this specification is considerably reduced by sweating and by drilling. If sweated-on fittings cannot be avoided the temperature during sweating operations is to be controlled pyrometrically, and must not be allowed to exceed 220° C. This requirement is to be complied with both when the tubes are used as received and when they are heat-treated to give a lower tensile strength.

21. Use of 4 B.A. bolts

(i) Attention is drawn to the risk involved by the use of 4 B.A. bolts where the removal of the nuts is likely to be necessary for maintenance operations, in that they are liable to be stripped or over-strained in tightening up.

(ii) In this connection it should be borne in mind that maintenance work must often be done by the service in unfavourable conditions, particularly when on active service.

(iii) The use of 4 B.A. bolts and nuts must therefore be avoided where failure due to stripping or overstraining would lead to serious results.

22. Use of even sizes of B.A. screws

With a view to keeping down superfluous stocks as much as possible, only even sizes of B.A. screws, as listed in the relevant A.G.S. sheets should be employed.

† **23. Use of tab washers**

The use of tab washers is to be avoided in the construction of aeroplanes in any place where their failure to lock a bolt securely would lead to the development of structural failure, or to the controls failing to function in the air.

24. Design of wiring lugs

(i) Experience has shown that wiring lugs as in fig. 1 are unsatisfactory. The thickness of the sheet and the stiffness of the bolt are generally insufficient to prevent concentration of the load at the ends of the lug as at *aa*, with the result that a crack is

* Previously A.D.M. 334. † Previously A.D.M. 237.

started which gradually travels from end to end. It is preferable that such lugs should be made from forgings as in fig. 2.

(ii) It is also important in all fittings that sharp corners and rough tool marks should be avoided, as cracks are likely to start from stress concentration at such places.

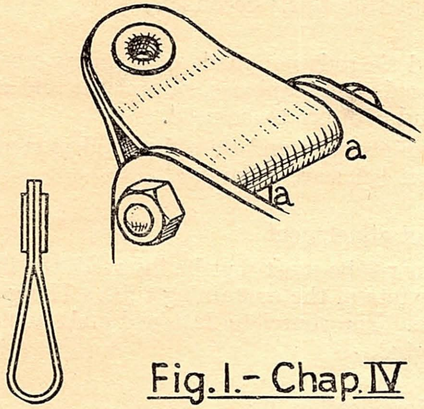

Fig. 1.- Chap. IV

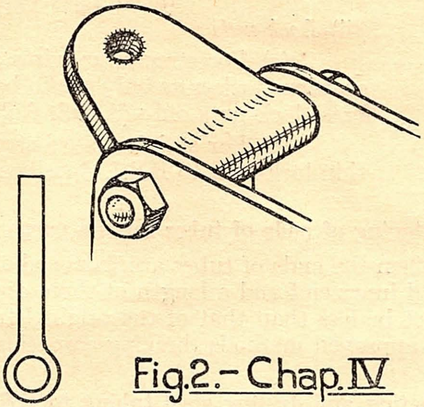

Fig. 2.- Chap. IV

*** 25. Lugs for external wire bracing**

To reduce the risk of fatigue failures of external bracing wires or attachments which are under tension during the greater proportion of the time in flight, the thickness of the wiring lug is to be such that there is a clearance between the lug and the jaws of the fork-end fitting and the hole in the lug is to provide a loose fit for the pin.

It is not possible to state the minimum clearances necessary to eliminate such failures but, in general, the requisite slight freedom of the lug in the fork will be obtained if the clearance in the jaws is not less than 2 per cent. of the width between them and if the clearance for the pin in the hole is not less than 2 per cent. of the pin diameter.

For external bracing wires which are slack during the greater proportion of the time in flight, serious elongation of the hole in the wiring lug may occur if too much clearance is provided for the pin. The hole, therefore, is to be no larger than that required for assembly purposes and excessive vibration of these wires is to be prevented by the use of spreaders or by other suitable means.

† 26. Sections of aeroplane metal parts (produced from hot-formed sections or forgings)

Sharp-cornered sections are unsuitable in aeroplane structural parts (whether rolled, drawn extruded or machined) which have to carry any appreciable bending or torsional stress. All such parts should have fillets of a reasonable size. In general, the inner radius of a fillet should not be less than one-fifth of the thickness of the flange or web (whichever is the thicker) at the corner.

27. High-tensile steel fittings

(i) The edges of all holes should be given a small radius, and buffing all over is recommended as a final finish.

(ii) Although certain high-tensile steels may be welded, great care is required and complicated fittings involving difficult welds should be avoided in these steels.

(iii) The position for identification markings and inspector's stamps must be indicated in the drawings of structural parts.

* 666102/37. † Previously A.D.M. 347.

* **28. Bending of aluminium alloy sheets and strips** (applicable to material to B.S. Specifications 4.L.3. and L.38 and D.T.D.270 and 275)

Attention is directed to the necessity for the employment of suitable radii in bending and forming operations on aluminium alloy sheets and strips, and it is notified that the radii specified below are the minima that can be accepted for the manufacture of aeroplane parts and fittings.

*

Condition	Minimum radius			
	Angle of bend. 18 s.w.g. and thinner		Angle of bend. Thicker than 18 s.w.g.	
	Through 120° or less	Over 120°	Through 120° or less	Over 120°
Fully annealed	0·5T	1T	0·5T	1T
Finally heat treated and within one hour of quenching	0·5T	1·5T	1·5T	2T
Finally heat treated and aged	2T	3T	2·5T	3T

In the above table T = thickness of sheet or strip.

(*Note.*—The minimum radii specified above apply to the tools over which the light alloy strips or sheets are bent, and not to the dimensions of the finished plating.)

† **29. Permissible bow in light alloy tubing for use in aeroplane structures**

(i) When light alloy tubing is used in compression in the primary structure or control system of aeroplanes the departure from straightness is not to exceed $\frac{L}{600}$.

(ii) Two classes of straightness will be included in future issues of specifications for light alloy tubing of $\frac{3}{4}$ in. outside diameter and over. These will be.—

Class "A" with a departure from straightness not exceeding $\frac{L}{600}$ and

Class " B " with limits of straightness to be agreed between the purchaser and the manufacturer.

(iii) For parts under para. (i) above, made of tubing of $\frac{3}{4}$ in. outside diameter or over, the drawing schedule should call for material to Class "A" straightness. For parts made of tubing less than $\frac{3}{4}$ in. outside diameter, the limit of departure from straightness, $\frac{L}{600}$, should be given in the detail drawing.

(iv) When Class "A" straightness is required for tubing to existing specifications the order should be marked accordingly until such time as the specifications are revised to include the straightness clause.

* Previously A.D.M. 338. † Previously A.D.M. 344.

*\ **30. Corrosion of bolts to B.S. Specifications S.61 and S.62 in wooden members subject to wetting in service**

It has been found in practice that bolts made of material to the above Specifications have been seriously weakened by corrosion in places where they pass through wooden members which are liable to become wet in use, as for example, the spars of the lower main plane and the tail plane spars of flying boats and seaplanes. For this reason, S.61 and S.62 material should not be used for bolts and other parts in contact with wood on flying boats and seaplanes. S.80 is a suitable material for such use. A.G.S. parts 868 and 869 (bolts) and A.G.S. parts 863–877 inclusive (nuts) are the current parts which should be called for on aeroplane drawings.

31. Collars for high-tensile steel pins

Where high-tensile steel pins (A.G.S. 383 and 384) are used to connect actuating levers and control rods or cables, the rotary movement sometimes causes excessive wear on the small diameter split pin securing the high-tensile pin in position. Provision is to be made to prevent this and an approved method is to fit a small duralumin collar (A.G.S.674) over the end of the high-tensile pin and insert the split pin through the hole already drilled in the collar.

32. Provision of longitudinal datum marks

Aeroplanes are to be provided with datum marks complying with S.I.S. No. 36 for checking the longitudinal rigging position.

33. Safe limit of deterioration of shock absorber legs

Shock absorber legs are to be provided with a gauge or mark to indicate when the safe limit of low air pressure and/or deterioration of the rubbers has been reached.

34. Securing of windscreens (*see* also Chapter II, para. 27)

(i) Adequate precautions are to be taken to secure the glass firmly in windscreen frames by positive locking methods.

(ii) In windows or windscreens of transparent material such as cellastoid, rhodoid, etc., secured by riveting in a metal frame, it is very important to ensure that sufficient clearance is provided to permit the material to contract freely at the low temperatures reached in flight. This clearance should be provided both at the rivet and bolt holes and also between the edges of the transparent panel and its frame, or alternatively between adjacent panels. The provision for securing the edges of the panels should be such that when contraction or expansion due to changes of temperature takes place the panels are free to adjust themselves. The clearance allowed for holes, edges, etc., should be of the order of $\frac{1}{2}$ per cent. of the linear dimensions of the panel.

(iii) If it is essential that the joints between the panel and its frame should be water-tight, it may be necessary to provide some form of flexible packing at the joints.

(iv) It is pointed out that failures have occurred during flight due to clearance not being provided at the point of attachment and it is also to be anticipated that failure to allow for temperature effects on the material will result in the surface deforming with consequent impairment of visibility. The drawings of such parts should be dimensioned and noted so as to ensure that the necessary clearances are provided.

35. Fasteners for cowling and inspection doors

These fasteners must give clear indication whether they are in the locked or free position. As far as possible all fasteners should be arranged to lock with the slot in the direction of flight.

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* Previously A.D.M. 206.

36. Compression shakes in spruce rib flanges

Compression shakes which have not passed the neutral axis of the flange and which occur in that part of the flange in the forward two-thirds of the distance between the leading edge and the front spar, may be allowed in ribs where the flange is reinforced by a three-ply or solid spruce web. In cases where the flange is not suitably supported, compression shakes cannot be allowed.

37. Use of duralumin tubes and sheet thinner than 22 gauge (·025 in. and ·028 in.—*see* Chapter VIII, Section I, para. 4)

Main structural members of duralumin such as struts and spars are not to be made of material thinner than 22 gauge. This does not apply to.—

(*a*) seamless circular tubes less than 2½ in. outer diameter used in positions where they are protected from handling loads (e.g. inside a main plane) ;

(*b*) stiffeners, diaphragms and webs of spars ;

(*c*) the skin and stiffeners of monocoque construction ;

(*d*) ribs, trailing edge tubes, etc. ;

(*e*) wing slats.

38. Provision for ballast (*see* Chapter III, para. 19)

(i) Provision for ballast should be made in the early stages of design.

(ii) Provision is to be made for carrying sufficient ballast to trim the aeroplane if necessary when any items of removable equipment or members of the crew are omitted from the load.

(iii) On aeroplanes destined primarily for overseas, provision is to be made for carrying ballast in a neutral position where it does not affect the trim so that it may be available for trim if required at a later stage of the flight.

(iv) If the ballast is carried internally the stowage is to be at the extreme rear of the fuselage or forward of the foremost member of the crew actually carried when the ballast is in position.

39. Use of parallel pins

Solid parallel rivets which are unsupported for the major portion of their length, such as through thin tubes, are not to be used where it is possible to use taper pins in reamed holes. When solid parallel pins are essential to the design, then the length/diameter of the unsupported portion should not exceed 4. This restriction is necessary because the strength of the joint depends upon the pin tightly filling the hole and it is not always possible to ensure by inspection that the riveting of parallel pins has produced the required result.

40. Attachment of wireless aerials (fixed and trailing)

(i) To determine the necessary strength for those parts of an aeroplane structure which support a fixed or trailing aerial, tensions will be assumed in the aerial wires producing an aerial load of such magnitude that the breaking strength of the weakest part of the aerial system is just reached. Parts of the airframe which are affected by any of the standard stressing cases must be so designed that the superposition of the aerial load upon the full factored aerodynamic load for the relevant stressing case does not bring the structure below the minimum requirements.

(ii) If welding is used for securing aerial attachments to the airframe, the attachments and the form of the weld are to be such as will give an ample margin of safety against vibration fractures.

41. Retractable undercarriages

A.—General requirements

(i) *Indicators*

(*a*) A visible indicator is to be provided showing the pilot when the undercarriage is in the safe landing position and when it is in the safe retracted position. It need not indicate intermediate positions.

(*b*) If the indicator is electrical, the making, not the breaking, of a circuit is to indicate that the undercarriage is in position for landing. Preferably a green light should show all the time the undercarriage is in the safe landing position*, and a red light when the undercarriage is in the safe retracted position.

(*c*) The indicator is to show whether the locks (if any) which hold the undercarriage in the safe landing position have gone home. When locks are fitted the indicator must be operated by the locks themselves. The locks must not operate the indicator until they are fully home.

(*d*) An audible or specially prominent visible indicator must be fitted in addition to and independent of the visible indicator referred to in sub-paras. (*a*), (*b*) and (*c*) above. This additional warning device is to be arranged to indicate danger to the pilot if he attempts to land with the undercarriage in an unsafe position for landing. It may be arranged to be brought into action by any of the following—(i) air-speed indicator, (ii) air speed, (iii) incidence, (iv) landing flaps, (v) engine r.p.m. The arrangement should give the maximum possible certainty of the warning device operating in time to prevent a landing with the undercarriage in an unsafe position, while at the same time the device should not be brought into action during ordinary flight manœuvres. The provisions of this sub-para. (*d*) do not apply to amphibians.

(*e*) If the indicator of sub-paras. (*a*), (*b*) and (*c*) and the emergency indicator of sub-para. (*d*) are both electrically operated, the two systems are to have separate and independent fuses, or alternatively some equivalent safeguard is to be provided to minimize the possibility of both warning systems breaking down together.

Note.—An earth return through the frame or bonding system is not permitted. Insulated cable must be used for both positive and negative leads.

(ii) *Operating gear*

(*a*) Provision is to be made on hydraulically operated gears for lowering the undercarriage and locking it in the landing position in the event of a burst pipe or failure of a mechanically-operated pump.

(*b*) Provision is to be made on electrically-operated gears for lowering the undercarriage and locking it in the landing position in the event of a breakdown of the electrical motor or electricity supply.

(*c*) Precautions are to be taken to prevent damage, due to overwinding at both ends of the travel. If hand operated, it must be impossible for the operator to damage any portion of the mechanism if he exerts his full strength to overwind the gear.

(*d*) Unequal lowering of the two sides must be capable of being corrected by continuing to operate the lowering gear until both sides are fully down.

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* The switch needed to prevent the green light burning all the time the aeroplane is on the ground should be interconnected to the engine switch so that switching *on* the engine also switches on the green light while switching *off* the engine does not break the green light circuit. Equivalent alternative arrangements will be considered.

(iii) *Locking*

(*a*) Positive locks are to be provided to hold the undercarriage in the landing position. A dead centre mechanism will not, in general, be accepted without locks as an additional safeguard. An irreversible worm gear will in general be accepted as complying with this sub-paragraph.

(*b*) If locks are provided to fix the undercarriage in the up position, precautions are to be taken to prevent the lock seizing up (e.g. due to mud, rust or flight deflections) and so making it impossible to lower the undercarriage. In addition to the normal means provided for operating such locks, a second emergency method of opening or breaking the locks must be provided. This emergency method must be capable of applying to the lock mechanism a force much greater than is necessary for normal operation of the locks.

* (*c*) The locking mechanisms securing the undercarriage in the up position and in the down position are to be such that effective inspection of their mechanical condition can be carried out by service personnel without a large amount of dismantling.

(iv) *Strength*

(*a*) When locked ready for landing, a retractable undercarriage is to comply with all the requirements applicable to a fixed undercarriage.

(*b*) The shock absorber leg must be capable of functioning satisfactorily if lowered quickly immediately before the aeroplane makes contact with the ground.

(*c*) In both the open and the retracted positions the undercarriage and mechanism are to have the factors specified for the aeroplane under all loads (e.g. acceleration loads) that may arise in flight. In any intermediate position the factors should be at least one-half those specified for flight loads.

B.—Tests

New designs of retractable undercarriages and/or operating gears fitted on new experimental aeroplanes or on existing types are to be tested as follows.—

(i) *Static tests prior to taxying trials*

(*a*) Jack the aeroplane off the ground with the jacks so placed as to deflect that part of the wing affecting the operation of the undercarriage to an extent comparable to the deflection in steady level flight at cruising speed when fully loaded. Raise and lower the undercarriage twenty-five times, locking it in the extended and retracted positions; if locks are fitted in the retracted position these are to be released by the emergency method twice during the course of the test. The emergency device may be reset after each operation.

(*b*) Repeat test (*a*) with weight of wheels reduced to 80 per cent. of their normal weight.

† (*c*) Repeat tests (*a*) with the wheels loaded so that

1. for 90 per cent. of the full movement of the retractable undercarriage (starting from the extended position) the wheels weigh 1·33 times their normal weight and

2. for the last 10 per cent. movement the wheels weigh twice their normal weight.

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* Previously amendment No. 1 to A.D.M. 321.
† Previously corrigendum No. 2 to A.D.M. 322.

(*d*) Repeat test (*a*) with a load representing twice the air drag on wheels and undercarriage appropriate to flight at a speed equal to twice the stalling speed with flaps normal. If it is found to be impracticable to arrange the test rig to permit full retraction of the undercarriage with the load in place, partial retraction is permissible, but the loading system should be devised to allow the greatest possible range of retracting movement.

(*e*) If the retracting gear is hydraulically or pneumatically operated it is to be proved by test that in the event of a pump or pipe failure the undercarriage may be lowered to the extended position and locked there.

(*f*) If operated by electric power a similar demonstration to that required under (*e*) is required to prove satisfactory operation in the event of breakdown in the electrical system.

(*g*) *Notes*

1. During tests (*a*) to (*d*) inclusive the operator should ensure that the undercarriage is hard against its stops at the limiting positions, using the full available force to effect this. The locking device in the retracted position should be examined under each test condition to ascertain the adequacy of the locking, taking into account possible structural deflections in accelerated flight.

2. It is of great importance that special care should be taken to support the aeroplane in such a way that excessive loads are not imposed on the aeroplane structure. If the presence of the skin is not essential during the tests, it is preferable that the tests should be made before the covering is applied. This would enable a thorough inspection to be made after the tests.

3. If the prime mover for the undercarriage is the engine, some other prime mover (e.g. an electric motor) is to be provided for the static tests of sub-para. (i) to avoid having to run the engine with the aeroplane jacked up.

(ii) *Taxying trials.*—Taxying trials, as follows, are to be made prior to actual flight.

(*a*) *Aeroplane at light load* (tare weight plus reasonable fuel, oil, ballast and crew).—

1. Run engine(s) against chocks, brakes off, to full ground r.p.m.

2. Run engine(s) against brakes, no chocks, to full ground r.p.m. or until brakes or wheels slip.

3. Make straight runs tail up and tail down, three in each attitude, to $\frac{2}{3}$ stalling speed (I.A.S., flaps not in use). Bring aeroplane to rest by progressively harsh use of brakes on successive runs.

4. Turn aeroplane port and starboard with one wheel locked by the brakes, three complete turns, each way, raising the speed in the turn progressively, to a reasonable maximum in the final turn.

(*b*) *Aeroplane fully loaded.*—

Repeat tests under 3 and 4 above.

Note.—Flight trials at light load are permitted between the series of taxying trials under (*a*) and (*b*) above.

* 42. Undercarriage springing characteristics

Comparative tests show that undercarriages vary considerably in their landing and taxying qualities. It is not possible to specify quantitatively all the features which make for a satisfactory undercarriage but tests on service aeroplanes indicate that particular attention should be paid to the following.—

(i) *Track.*—The undercarriage track should be as wide as possible.

(ii) *Taxying.*—To obtain good springing when taxying the travel of the oleo from the static position (i.e. the position when the aeroplane is at rest on the ground) to that corresponding to $3\,W$ should be large. This, however, will make the aeroplane roll badly unless the undercarriage track is wide, hence the importance of (i).

(iii) *Rebound.*—In order to reduce the tendency to rebound on landing, rebound damping must be provided. A good working rule is to provide sufficient damping to reduce the velocity of opening to not more than half the velocity of closing when the oleo is tested under conditions representing the specified vertical velocity of descent.

43. Torsional stiffness of ailerons

(i) *Ailerons with distributed mass-balance* (*see* definition in sub-para. (iii)).—The value of the torsional criterion

$$\frac{1}{VS_a}\sqrt{Ts'}$$

is not to be less than 0·010.

(ii) *All other ailerons.*—The value of the torsional criterion

$$\frac{1}{VS_a}\sqrt{Ts'}$$

is not to be less than 0·018, and in addition special care is to be taken to avoid material changes of aileron torsional stiffness along the aileron span.

(iii) *Definitions*

T = aileron torsional stiffness estimated as described in (iv) or (v) below (pounds-feet per radian twist).

S_a = area of that portion of the aileron which is aft of the hinge line (square feet).

s' = overall span of aileron measured parallel to the aileron hinge line (feet).

V = maximum indicated air speed to which the aeroplane is required to be dived during contractor's trials (feet per second).

The strict theoretical implication of " distributed mass-balance " is that each fore-and-aft cross section of the aileron is mass-balanced in that same cross section. While this is the thing to aim at it will sometimes be inconvenient to provide mass-balance in this way. For the present purpose, therefore, an aileron will be deemed to have distributed mass-balance provided that if the aileron were divided into two parts by a cut through its mid-span at right angles to its hinge line, each half would itself be separately mass-balanced when in its correct relative position on the aeroplane. This approximate rule only applies provided that mass-balance is not effected by means of a concentrated weight at the mid-span of the complete aileron, nor by two concentrated masses each at the inner end of its respective half aileron.

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* Previously A.D.M. 246.

(iv) Let *aa* and *bb* be two cross sections at right angles to the hinge line of the aileron distant respectively $0 \cdot 1\ s'$ from the inner and outer ends of the aileron. Hold the aileron rigidly at *aa* and apply a couple at *bb*. Then T is the torque required to produce one radian twist, assuming a linear stress-strain relationship. If the aileron is fabric covered, T is to be measured or estimated with the fabric removed. It will usually be necessary to estimate T by a mechanical test. For this purpose the aileron should be detached from the aeroplane and wooden frames fitted round its section at positions *aa* and *bb*. The aileron should then be slung from its hinges in a vertical plane with its hinge line horizontal and one frame fixed whilst the other is left free to rotate with as little restraint as possible from the slinging cables or cords. By applying incrementally increasing twisting couples C_1, C_2 .... to the free frame and measuring the resulting twists θ_1, θ_2..... between the sections *aa* and *bb*, and plotting, the average value of $C/\theta = T$ may be obtained. In applying these couples care should be taken to avoid causing any damage or permanent strain to the aileron. Owing to the presence of a control lever or a mass-balance arm, or for some structural reason, it may not in some cases be practicable to apply the couples and measure the twists at the exact positions *aa* and *bb*. In such cases the couples should be applied outside rather than inside the ideal sections and an additional twist measurement made at a section near the mid-span of the aileron. The measurements should then be corrected to give the value of T by plotting twist against aileron span and interpolating for the twist between the ideal sections.

(v) A mechanical test will be unnecessary on an aileron whose torsional stiffness is primarily given by an aileron spar of known torsional properties (e.g. a circular metal tube), provided that T is taken as the calculated torsional stiffness of this spar itself without any additional allowance for ribs, leading and trailing edges, etc.

44. Torsional stiffness of elevators

(i) The overall torsional stiffness of an elevator shall be such that the value of the criterion

$$\frac{1}{V S_e}\sqrt{T s''}$$

is not less than $0 \cdot 010$.

(ii) In the above criterion

T = elevator torsional stiffness (pounds-feet per radian twist).

S_e = area of that portion of the elevator (comprising all sections in the same plane) which is aft of the hinge line (square feet).

s'' = overall span of elevator, including all sections in the same plane (feet).

V = maximum indicated air speed to which the aeroplane is required to be dived during contractor's trials (feet per second).

(iii) T should be obtained in a manner similar to that described for ailerons in para. 43 (iv) above, the two sections *aa* and *bb* being taken at distances $0 \cdot 1\ s''$ from the extreme port and starboard tips. Calculation may be used if the conditions of para. 43 (v) apply and if any interconnection fittings present in the elevator unit have well-known torsional properties.

* **45. Repairs to mass-balanced surfaces (ailerons and rudders)**

(i) In all future designs allowance must be made for the effect of repairs on mass-balance. This will in general necessitate a slight increase of the balance weight above that previously required, but care should be taken to see that repairs are so designed as to involve as little increase of weight as possible.

(ii) On all future designs the aileron and rudder mass-balance requirements of this chapter, paras. 2 and 4 are to be complied with when *three* typical repair patches have been applied to the control surface so placed as to have the most adverse effect upon mass-balance. In fabric covered surfaces the additional weight due to the patched fabric is to be included. Alternatively, a simple means of correcting the mass-balance is to be provided in the repair scheme as a part of the instructions for repair of the control surfaces.

(iii) If provision is made for trimming an aileron by doping cord along its trailing edge, or by some equivalent method, the most adverse arrangement of such trimming device is to be taken into account when complying with the aileron mass-balance requirement.

† **46. Ground clearance**

The ground clearance for airscrews, elevators, fins and rudders must not be less than the following.—

(i) *Airscrews*

(*a*) Aeroplane resting on the ground (tail down), tyres normally inflated and the undercarriage and tyres compressed corresponding to 3*g*. .. .. 9 in.

(*b*) Aeroplane resting on the ground, tyres normally inflated with the thrust line horizontal and under carriage and tyres compressed corresponding to 3*g*. .. .. $3\frac{1}{2}$ in.

When the thrust line is abnormally tilted, allowance for this will be made in determining the attitude of the aeroplane at which this airscrew clearance is to be measured.

(ii) *Elevators* (*fully down*), *fins and rudders*

(*c*) Tail at rest on the ground with tail wheel and its shock absorber compressed corresponding to 3*g*. .. .. 6 in.

‡ **47. Hand and foot holes in airframes**

In designing hand and foot holes in airframes it is necessary to ensure that the flap or similar device to prevent ingress of foreign matter is arranged to operate in such a way that there is no possibility of the hand or foot being grazed or jammed on withdrawal in any position. Metal flaps with a spring hinge should have sufficient overhang to ensure that the outer edge is never in contact with the fingers, and this edge should be beaded or otherwise arranged so as to run smoothly over a boot.

§ **48. Buoyancy of engines in estimating buoyancy of aeroplanes**

In estimating the buoyancy to be provided by emergency flotation gear, the buoyancy of the engine complete with airscrew hub, exhaust system and accessories is to be based on a mean specific gravity of 5·25, no allowance being made for air spaces within the engine, exhaust system or accessories.

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* Previously A.D.M. 305.

† Previously A.D.M. 362.

§ Previously A.D.M. 265.

‡ Previously A.D.M. 272.

* **49. Protection of aeroplane cowlings and structure from gun blast**

Experience has shown that the blast from fixed guns causes damage of various kinds to those parts of the aeroplane adjacent to the gun muzzle and immediately in front of it. For example, the gun blast tends to lift up the cowling joints in the gun tunnel when these face aft. Damage occurs consequent on the panting of insufficiently supported surfaces, rivets working loose and rivet heads being removed by wads at breaks or joins in the surfaces ahead of the gun muzzle.

DESIGN RECOMMENDATIONS

(i) Where guns are mounted in tunnels or grooves, the plating should be sufficiently stiff to avoid panting, and joints should be avoided between a point aft of the muzzle attachment to a point 2 ft. ahead of the muzzle attachment. The top and bottom edges should be lapped over the edges of the adjacent cowling. Grooves and tunnels of 22 s.w.g. steel to B.S. Specification S.3 have in general been found satisfactory.

(ii) If lap joints are used in the cowling joints of the gun tunnel these joints should face forward. Consideration should be given to the use of butt joints.

(iii) Where the gun muzzle is not housed in a tunnel or groove the surface adjacent to and immediately forward of the muzzle should be adequately supported to avoid panting.

(iv) Use of rivets should be avoided as far as possible immediately in front of the gun muzzle. Where rivets are essential they should be flush with the surface exposed to blast.

† **50. Closed structures** (provision for internal inspection and removal of riveting pin or mandrel heads)

(i) In closed structures such as stressed skin wings, large box spars, etc., in which the form of construction is such that the satisfactory completion of assembly processes cannot be seen by external inspection alone, the design must provide for internal inspection both during manufacture and in the final inspection, and must provide for the removal of dropped rivets, rivet mandrel heads, clips, bolts and nuts, tools and other loose articles which may have been left in the component during assembly.

(ii) In any portion of the structure containing moving mechanism, such as parts of the control system, a means of ensuring the complete absence of loose articles such as those mentioned above is to be provided, and means of inspection for the presence and removal of any items such as rivet mandrel heads, which may become free during subsequent use, must be arranged.

(iii) Any closed part of the structure to which ready access is not provided and which is liable to contain the broken off heads of rivet mandrels used in its construction or final closing, must be sealed to prevent these heads from gaining access to parts containing controls, unless the pin heads are of the type specially formed to remain secure in the rivets, and approved as such by D.T.D.

‡ **51. Emergency exits in aeroplanes**

(i) All aeroplanes are to have easy means of exit for all occupants under the following conditions.—

(*a*) Emergency exit by parachute in the air.

(*b*) Emergency exit on ground from an overturned aeroplane.

(*c*) With the aeroplane floating in the water after an emergency landing at sea (this only applies to aeroplanes with flotation gear).

(*d*) Emergency exit on ground from an aeroplane which has landed with the undercarriage retracted.

* Previously A.D.M. 179 and 300. † Previously A.D.M. 348. ‡ Previously A.D.M. 361.

(ii) It is appreciated that standardized requirements cannot be expected to meet every case, but the following detailed requirements should be complied with as far as possible.—

(*a*) All enclosed aeroplanes are to have one or more easily breakable or opening windows, large enough for a man wearing a parachute to get through easily. Sliding roofs or opening windows which may make satisfactory parachute exits must not be counted as emergency crash exits in the number required by (*g*) below unless their design is of such a type that, when opened, satisfactory exit is possible from an aeroplane which is overturned on the ground.
In such cases the sliding roof may be assumed to have been opened before the aeroplane overturned, if the roof when opened is secured from sliding forward.

(*b*) All aeroplanes must be provided with conveniently placed handgrips, supports, etc., to assist the occupants to extricate themselves from the machine whilst experiencing accelerations (e.g. in a spin).

(*c*) Doors counted as parachute exits, except where they are of the type which can be completely jettisoned, should open inwards.

(*d*) Exit from enclosed turrets should be either direct or through an exit accessible and adjacent to the turret.

(*e*) All doors are to be fitted with positive locks and independent safety catches.

(*f*) Emergency parachute exits must be situated so as to minimize the risk of the persons using them colliding with the tail or other parts of the aeroplane.

(*g*) Enclosed compartments are to be provided with exits from each compartment at the rate of at least one for every four occupants, to meet all cases of emergency on the ground and water as shown in sub-para. (i) above.

No. of occupants in the compartments	No. of exits to meet each emergency under (*a*), (*b*), (*c*) and (*d*) of sub-para. (i) above
4 or less	1
5 to 8 inclusive	2
9 to 12 inclusive	3
Over 12	4

(iii) Where hoods, for blind-flying practice, or sunblinds are provided in cabin tops that have exits for use in emergency, the operating mechanism of the blind or hood must be such that, when the cabin top is opened, the sunblind or hood is automatically furled or folded out of the way so that it cannot prevent easy exit with parachute.

COPY OF R.A.E. CIRCULAR LETTER NO. 677.

6th, September, 1939.

CLASS I CASTINGS.

The following remarks concern the interpretation of the requirements for Class I Castings as given in A.P. 970, Chapter IV, Paragraph 16 (1).

Proof Strength.

The requirement that a Class I Casting must be capable of withstanding twice the fully factored loads applied to the aeroplane as a whole concerns ultimate failure and is not applicable to proof failure.

It is, however, considered essential that a Class I Casting should be capable of withstanding the fully factored loads applied to the aeroplane as a whole (as distinct from three quarters of them), without appreciable yield.

Test Factor.

Subject to the above conditions being satisfied it will not be necessary to increase the loads for test purposes merely to cover variation from standard dimensions and material specification, i.e., the 20% margin for test is not required.

Extended Definition of Class I Castings.

Class I Castings will in future include not only those castings whose failure might lead to collapse of the structure or loss of control, but also those castings whose failure might lead to release of a bomb.

(Please amend your copy of A.P. 970 as ~~[illegible]~~ above)

CHAPTER VI.—THE STRENGTH OF SPARS

1. Introductory.—This chapter describes the methods usually adopted for estimating the strength of metal spars. Special consideration will be given to cases where the applicability of these methods is doubtful.

In this chapter the assumption is made (which is in general approximately true) that the minor principal axis of the spar section lies in the plane of the drag bracing system. Where for convenience of strut fitting the spar section is "skewed" or twisted over (generally to conform to the angle of stagger), the consequent interaction of vertical and horizontal loads and deflections necessitates considerable modifications. Where the angle of "skew" is considerable, special methods of calculation may have to be employed. Such cases should be referred to the Airworthiness Department.

In general a spar will be loaded both laterally and axially, the unit lateral load w being obtained as described in chapter V and the unit end load P from the stress diagram for the truss considered. It is required to find the factor N such that under loads Nw and NP the stress at the weakest point on the spar is equal to the maximum allowable stress.

2. Maximum allowable stress.—The maximum allowable stress should be obtained from an *ad hoc* strength test on a specimen representing a portion of spar concerned. Usually the test specimen is chosen to represent a portion of one of the spar bays of length equal to the length between the points of contraflexure, determined from the spar bending moment diagram. When there are no points of contraflexure the length of the specimen required will not normally exceed fifteen times the maximum depth of the spar. The spar is mounted* as illustrated in fig. 1.

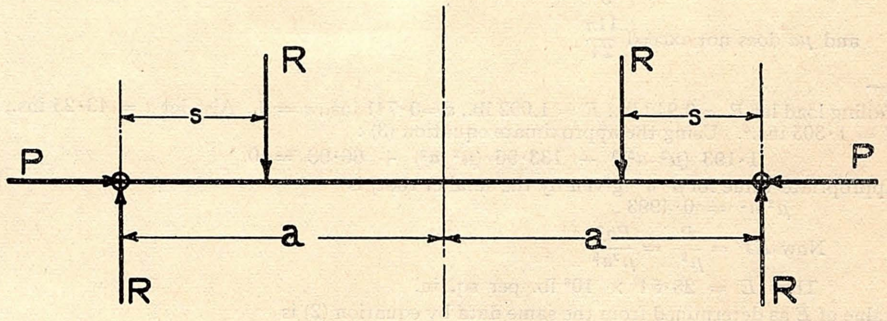

FIG. 1.—CHAP. VI.

A standard form of roller bearing pin end fitting is employed to give zero bending moment at the spar extremities and loading boxes are required at the points of application of the lateral load. The spar is supported in the drag plane at points of rib attachment, one such support being omitted to represent conditions arising when one rib fails or is shot away (chapter III, para. 21). The magnitudes of the couple Rs and the end load P are arranged to give approximately the same ratio of bending moment stress to direct stress as obtains at some critical portion of the spar when built into the aeroplane. The length of the bending arm s is chosen so that the load R gives a shear about equal to the greatest shear over that portion of the spar in the aeroplane represented by the test specimen. If the shear at any other point along the spar is considerably greater than this, and if the spar design is such that its strength in shear is in

* Other methods of tests are acceptable. If it is desired to use any method not hitherto approved prior concurrence should be obtained.

doubt, a second test specimen is sometimes necessary for a special shear test. On test R and P are increased until the spar breaks, the ratio R/P being kept constant.* Then, if R and P are the greatest loads that the spar will carry, the maximum allowable stress is

$$\frac{P}{A} + \frac{R \sin \mu s \sec \mu a}{\mu Z} \quad \ldots \quad (1)$$

where $\mu^2 = P/EI$, and A and Z are the cross section area and modulus of section obtained from the measured dimensions of the test specimen. The value of E used is such that the calculated and measured deflections of the test specimen at mid span relative to the ends of the specimen coincide at three-quarters of the failing load, *i.e.* if d be this measured deflection, then E is chosen to satisfy the equation

$$d = \frac{R \sin \mu s \sec \mu a}{\mu P} - \frac{Rs}{P} \quad \ldots \quad (2)$$

when the values of R and P are three-quarters of the maximum values, and d is positive downwards for the loading system of fig. 1.

If the P's are applied eccentrically at a distance e (measured positive upwards) from the neutral axis then $\frac{Pe\sec\mu a}{Z}$ should be added to expression (1) and e $(\sec\mu a - 1)$ to the right-hand side of equation (2).

A convenient approximation to equation (2) for finding μ^2 when the other quantities are known is

$$\frac{s^3}{6a^2}(\mu^2 a^2)^2 - \left[\left\{5s + (5e + 4d)\,p\right\} - \frac{\pi^2 s^3}{6a^2}\right]\mu^2 a^2 + pd\pi^2 = 0 \quad \ldots \quad (3)$$

where $p = \frac{P}{R}$ and the eccentricity e is taken into account.

This approximation is sufficiently accurate if

μs does not exceed $\frac{\pi}{3}$

and μa does not exceed $\frac{11\pi}{24}$

Example :—

At $\frac{3}{4}$ failing load let $P = 9{,}940$ lb., $R = 1{,}092$ lb., $d = 0{\cdot}741$ ins., $e = 0$. Also let $a = 43{\cdot}25$ ins., $s = 23{\cdot}75$ ins., and $I = 1{\cdot}305$ ins.^{4}. Using the approximate equation (3) :

$$1{\cdot}193\ (\mu^2 a^2)^2 - 133{\cdot}96\ (\mu^2 a^2) + 66{\cdot}98 = 0.$$

The appropriate value for $\mu^2 a^2$ (given by the smaller root) is

$$\mu^2 a^2 = 0{\cdot}4993$$

$$\text{Now } EI = \frac{P}{\mu^2} = \frac{Pa^2}{\mu^2 a^2}$$

$$\text{Thus } E = 28{\cdot}54 \times 10^6 \text{ lb. per sq. in.}$$

The value of E as determined from the same data by equation (2) is

$$E = 28{\cdot}50 \times 10^6 \text{ lb. per sq. in.}$$

The moment of inertia, I, is to correspond to the measured dimensions of the test specimen.

A convenient method of calculating I is as follows. Draw out the section to a conveniently large scale, and mark off the flanges and webs into small elements of equal length (say $0{\cdot}1$ in. long) as shown in fig. 2.

Let t_1 = thickness of flange.
t_2 = thickness of web.
d_1 = distance of any one element of the flange from the neutral axis of the spar.
d_2 = distance of any one element of the web from the neutral axis of the spar.
n_1 = number of elements in flange.
n_2 = number of elements in web.

Then area $= 0{\cdot}1(n_1 t_1 + n_2 t_2)$.

$I = 0{\cdot}1 t_1 \Sigma\,(d_1^2) + 0{\cdot}1 t_2 \Sigma\,(d_2^2)$.

It is advisable to tabulate the calculation.

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* A convenient method of plotting the test results is given in R. & M. 1537.

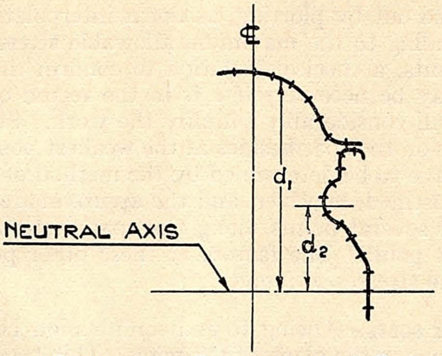

FIG. 2.—CHAP. VI.

Allowance for variation from minimum specification values in the material properties of the test specimen is made as follows. The 0·1 per cent. proof stress is obtained from a control tension test on a specimen of the material of which the spar is made,* and if this is greater than the minimum specification value the spar stress at failure estimated as above is reduced in the ratio

$$\frac{\text{Minimum specification } 0\cdot1 \text{ per cent. proof stress}}{0\cdot1 \text{ per cent. proof stress given by control test}}$$

If, however, the 0·1 per cent. proof stress given by the control test is *less* than the minimum specification value, the above correction is not applied. The reason for this is that this method of correction over-estimates the effect of variations in 0·1 per cent. proof stress, and hence would lead to an unsafe estimate of spar failing stress if applied to correct upwards as well as downwards. The question of correcting upwards should not arise, however, as it implies that the material of the test spar should not have been accepted. If the material specification quotes the 0·2 per cent. proof stress instead of the 0·1 per cent. proof stress, the 0·2 per cent. proof stress is to be used in making the correction described above. If ultimate stress only is quoted in the specification the scheduled 0·1 per cent. proof stress, obtained from chapter VIII, section VI, is to be taken. The maximum allowable stress, the value of E determined as above, and the moment of inertia and area corresponding to the minimum scantlings allowed by the drawings, are to be used in the calculations which follow.

3. Calculation of the stresses produced by given applied loads.—The generalised three moment theorem summarised in para. 6 is to be used for calculating the spar bending moments. The minimum realized factor N is ascertained by replacing w and P in equation (1) of para. 6, section A by Nw and NP, or, more generally, by multiplying all terms representing externally applied loads by N, and determining what value of N corresponds to a maximum spar stress equal to the maximum allowable spar stress. A direct solution for N is not readily obtainable. A convenient procedure is to calculate the maximum bending moment, and hence the maximum stress, corresponding to three trial values of N, these values being the specified ultimate factor and factors 10 per cent. greater and 10 per cent. less than this. The correct value of N which produces the maximum allowable stress at the weakest point in the spar can then be obtained

* This control specimen is to be cut from the same flat untreated strip from which the spar specimen was made, and not from the spar itself, (i) when the spar is formed by cold working with no subsequent heat-treatment, and (ii) when any heat-treatment during or subsequent to forming is not in accordance with the conditions for heat-treatment (if any) laid down in the specification. When the heat-treatment during or subsequent to forming is exactly in conformity with that laid down in the material specification, the control specimen is to be cut from the spar itself.

by interpolation, best carried out by plotting, as linear interpolation may be seriously in error. If the value of N corresponding to the maximum allowable stress falls appreciably outside the assumed $\pm$ 10 per cent. limits, a check calculation to confirm the value is desirable. Smaller limits than 10 per cent. may be necessary if α is in the region of $\pi/2$. A tabular method of arranging the calculation will considerably simplify the work. Strictly speaking, if factors are to be quoted at other points on the spar besides at the weakest point, the value of N appropriate to each such point would have to be determined by the method of trial and error just described. The work involved is not justified, however, and the approximate procedure usually adopted is to calculate the stresses at several points along the spar corresponding to the value of N as determined for the weakest point. The factors at these other points are then assumed to be inversely proportional to the stresses so found.

4. Secondary failure of spars.—Owing to end compression both spars of a plane may fail by buckling together sideways in the plane of the wing. This type of failure is called secondary failure and the strength of the spars to resist it may be calculated as follows :—

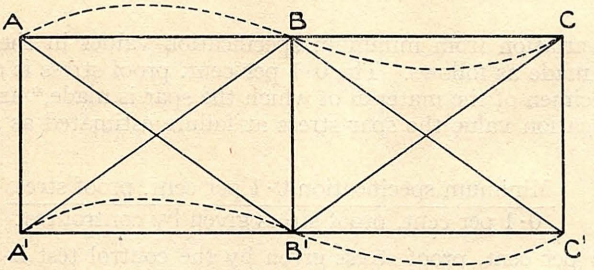

FIG. 3.—CHAP. VI.

Let fig. 3 represent the structure of a main plane where AC, $A'C'$ are the spars and AA', BB', CC' the drag struts. Then the failure in question will occur in the plane $ACC'A'$. Owing to the presence of the ribs (which act as ties between the two spars, and also, owing to the stabilising effect of the fabric, as struts), spars AB, $A'B'$ must deflect together in the same direction and by the same amounts if the lengths of the ribs between the spars are supposed invariable. The same applies to BC and $B'C'$. The only case which need be considered is that in which AB and BC deflect in opposite directions as shown in fig. 3. Hence we may regard AB as a strut pin-jointed at A and B and similarly for BC. It may be assumed that the ribs between the spars introduce no fixing moments. Referring to fig. 4, let

l = length of spar FF' = length of spar RR'.
A_F = cross sectional area of spar FF'.
I_F = moment of inertia of spar FF' for failure in the plane under consideration.
P_F = compressive end load in spar FF' when the wings are carrying unit load.
Q_F = Euler failing load for spar $FF' = \frac{\pi^2 EI_F}{l^2}$.
y_F = deflection at any point on spar FF' distant x from F.
x_L = distance of any rib from FR.
δ_F = equivalent eccentricity of spar FF'.
h_F = distance from neutral axis of spar FF' of most highly stressed fibre.
A_R, I_R, etc. = similar quantities for spar RR'.
L = load in any rib.
p_2 = 0·2 per cent. proof stress.
N = realized factor when failure occurs in one or both spars.

The spars are constrained to deflect equally by the ribs, so

$$y_F = y_R = y \text{ say} \quad \ldots \quad (1)$$

For the spar FF', when deflected under the factored loads

$$EI_F \frac{d^2y}{dx^2} = -NP_F y - NP_F \delta_F - \sum_0^x NL\,(x - x_L) \quad \ldots \quad (2)$$

For spar RR'

$$EI_R \frac{d^2y}{dx^2} = -NP_R y - NP_R \delta_R + \sum_0^x NL(x - x_L) \quad \ldots \quad (3)$$

Adding (2) and (3)

$$E(I_F + I_R)\frac{d^2y}{dx^2} = -N(P_F + P_R)y - NP_F\delta_F - NP_R\delta_R \quad \ldots \quad (4)$$

Generally

$$\delta_F = \delta_R = \delta \text{ say} \quad \ldots \quad (5)$$

If δ_F, δ_R, are unequal then the assumption is made that each is equal to δ where δ is the greater of the two

$$\therefore E(I_F + I_R)\frac{d^2y}{dx^2} = -N(P_F + P_R)y - N(P_F + P_R)\delta \quad \ldots \quad (6)$$

The signs of δ_F, δ_R are ignored so that the case considered will be the most severe for any particular numerical values of δ_F, δ_R.

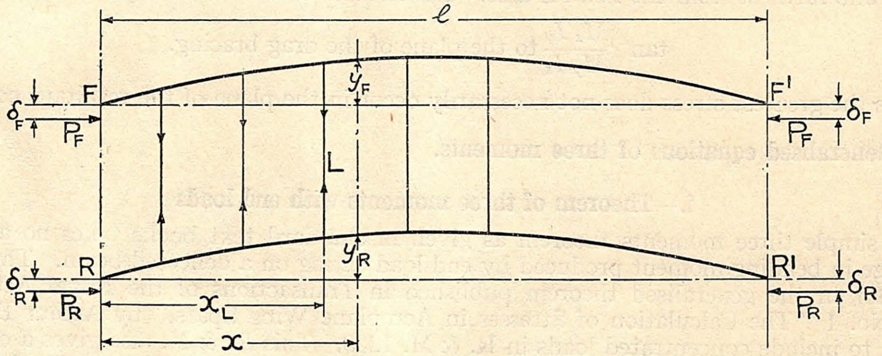

FIG. 4.—CHAP. VI.

Now (6) is the differential equation which would be obtained for a strut with moment of inertia $(I_F + I_R)$ and equivalent eccentricity δ, having a compressive end load $N(P_F + P_R)$.

The solution of the differential equation, when we put $I = I_F + I_R$ and $P = P_F + P_R$ is

$$y = G \sin \mu x + H \cos \mu x - \delta \quad \ldots \quad (7)$$

where G and H are constants to be determined and

$$\mu^2 = \frac{NP}{EI}$$

when $\quad x = 0,\ y = 0$ and so $H = \delta \quad \ldots \quad (8)$

when $\quad x = l,\ y = 0$ and so $0 = G \sin \mu l + \delta \cos \mu l - \delta$

Therefore $\quad G = \delta \dfrac{(1 - \cos \mu l)}{\sin \mu l} \quad \ldots \quad (9)$

By differentiation of (6) above it is clear that $EI \dfrac{d^2y}{dx^2}$ is maximum when $\dfrac{dy}{dx} = 0$.

By symmetry, this is when $x = \dfrac{l}{2}$.

Therefore maximum moment is

$$M_{max.} = -NP\delta\left\{\frac{(1 - \cos \mu l)}{\sin \mu l}\sin\frac{\mu l}{2} + \cos\frac{\mu l}{2}\right\} = -NP\delta \sec\frac{\mu l}{2}$$

$$= -NP\delta \sec\frac{\pi}{2}\sqrt{\frac{NP}{Q_F + Q_R}} \quad \ldots \quad (10)$$

It follows that for the spar FF' the equation

$$p_2 = \frac{NP_F}{A_F} + \frac{N(P_F + P_R)}{I_F + I_R}\delta h_F \sec\frac{\pi}{2}\sqrt{\frac{N(P_F + P_R)}{Q_F + Q_R}} \quad \ldots \quad (11)$$

gives the realized factor N at failure.

If the spar RR' fails first, the appropriate equation is

$$p_2 = \frac{NP_R}{A_R} + \frac{N\,(P_F + P_R)}{I_F + I_R}\,\delta\, h_R \sec\frac{\pi}{2}\sqrt{\frac{N\,(P_F + P_R)}{Q_F + Q_R}} \quad \ldots \quad \ldots \quad \ldots \quad (12)$$

In practice, all the terms in equations (11) and (12) are known except N. Both equations should be solved by trial and error and the lower of the two values of N obtained is the appropriate realized factor for the particular case of loading under consideration. It is difficult to specify any general value for δ applicable to all types of spars. Unless there are particular reasons for adopting a value greater than $l/600$ this value should be taken. It is clear that this type of failure is resisted to a certain extent by the leading edge to which the ribs are attached and to a still greater extent by the fabric when it is attached to the spars. For this type of failure, therefore, the minimum factor calculated as described above is to be multiplied by an arbitrary correcting factor which may be taken as 1·5 for a fabric covered wing and 3 for a wing with 3-ply or metal covering forward of and firmly attached to the front spar.

5. Unsymmetrical bending.—When it is necessary to consider the stresses due both to a lift bending moment M_y and to a drag moment M_z the greatest stresses will in general occur at the points furthest from the neutral axis. The neutral axis is at an angle :—

$$\tan^{-1}\frac{M_z\, I_y}{M_y\, I_z} \text{ to the plane of the drag bracing.}$$

Thus the greatest stress does not necessarily occur in the plane of the resultant couple.

6. Generalised equations of three moments.

i.—Theorem of three moments with end loads

The simple three moments theorem as given in standard text books takes no account of the change in bending moment produced by end load acting on a deflected beam. This effect is allowed for in the generalised theorem published in Transactions of the Royal Aeronautical Society No. 1 "The Calculation of Stresses in Aeroplane Wing Spars," by Arthur Berry, and extended to include concentrated loads in R. & M. 1233. This R. & M. also gives a convenient polar diagram method of drawing the bending moment diagram. *See* also R. & M. 1507, Appendix III, and the Journal of the Royal Aeronautical Society, June, 1934. The above publications should be referred to for the derivation of the formulae given below.

(i) *Notation and three moment equation.*—Referring to fig. 5 the points of support are denoted by the letters A, B, C, in order, beginning with the support nearest the tip; the bending moments at these points by M_A, M_B, M_C . . . which are counted positive when they tend to bend the beam concave upwards, thus $\smile$; the length of the members AB, BC . . . by $2a_1$, $2a_2$ . . . the loadings per inch run (taken positive upwards) by w_1, w_2 . . . that of the overhang being w_0; and the compressions (or tensions) by P_1, P_2 . . .

These end loads P_1 and P_2 are obtained by solving the truss shown in fig. 5 after applying the simple three moments theorem without end loads to the top and bottom spars A, B, C,

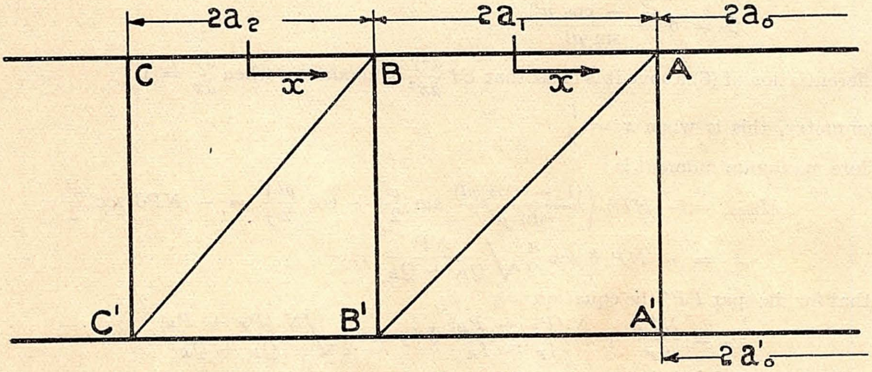

FIG. 5.—CHAP. VI.

and A', B', C' and adding the loads from the drag bracing. This is a first approximation to the end loads in the bays AB, BC and, usually, when there are no offsets, these end loads are sufficiently accurate for a single application only of equation (1) below to be necessary. When the lift wires are offset, however, more than one application of equation (6) below and hence more than one solution of the truss may be necessary for an accurate estimate of the end loads.

The deflection at any point is denoted by y, counted positive when upwards; the distance of any point in a member from the middle point of the member is denoted by x counted positively in the direction from B to A, so that for the member AB, $x = a_1$ at A, $x = -a_1$ at B.

Then the generalised theorem of three moments for a continuous beam with compressive end load is

$$\frac{a_1}{I_1} M_A f(\alpha_1) + \frac{a_2}{I_2} M_C f(\alpha_2) + 2 M_B \left\{\frac{a_1}{I_1}\phi(\alpha_1) + \frac{a_2}{I_2}\phi(\alpha_2)\right\}$$

$$= \frac{w_1 a_1^3}{I_1}\psi(\alpha_1) + \frac{w_2 a_2^3}{I_2}\psi(\alpha_2) \qquad \ldots \quad (1)$$

where

$$\mu_1 = \sqrt{\frac{P_1}{EI_1}}, \quad \mu_2 = \sqrt{\frac{P_2}{EI_2}},$$

$$\alpha_1 = a_1\mu_1, \; \alpha_2 = a_2\mu_2,$$

$$f(\alpha) = \frac{3}{2}\left[\frac{2\alpha \operatorname{cosec} 2\alpha - 1}{\alpha^2}\right] = 2\phi(\alpha) - \phi\left(\frac{\alpha}{2}\right)$$

$$\phi(\alpha) = \frac{3}{4}\left[\frac{1 - 2\alpha \cot 2\alpha}{\alpha^2}\right]$$

$$\psi(\alpha) = 3\left[\frac{\tan\alpha - \alpha}{\alpha^3}\right] = \frac{1}{3}\phi\left(\frac{\alpha}{2}\right)\left\{4\phi(\alpha) - \phi\left(\frac{\alpha}{2}\right)\right\}$$

If the supports are deflected by known amounts δ_A, δ_B and δ_C, measured positive upwards, then equation (1) becomes

$$\frac{a_1}{I_1} M_A f(\alpha_1) + \frac{a_2}{I_2} M_C f(\alpha_2) + 2 M_B \left\{\frac{a_1}{I_1}\phi(\alpha_1) + \frac{a_2}{I_2}\phi(\alpha_2)\right\} + \frac{3E}{2a_1}(\delta_B - \delta_A)$$

$$+ \frac{3E}{2a_2}(\delta_B - \delta_C) = \frac{w_1 a_1^3}{I_1}\psi(\alpha_1) + \frac{w_2 a_2^3}{I_2}\psi(\alpha_2) \qquad \ldots \quad (2)$$

If both members are alike, so that $I_1 = I_2 = I$, say, I divides out from equation (1) and can be omitted.

If the loadings are the same in the two members, so that $w_1 = w_2$, the right-hand side of equations (1) and (2) can be slightly simplified.

It is important to notice that when once the three functions which occur in (1) and (2) have been tabulated, their mathematical expression can be completely ignored; all that is wanted in any numerical case is to take the values directly from the tables. Tables for $f(\alpha)$, $\phi(\alpha)$ and $\psi(\alpha)$ have been carried up to $2{\cdot}36$ radians (*see* table I), the values of the functions between $1{\cdot}52$ and $1{\cdot}62$ radians being omitted as they are varying very rapidly in this region and linear interpolation is no longer sufficiently accurate. Approximate formulae are given at the appropriate place in table I from which the functions over this range can easily be calculated. The calculations should be carried out with special care when α in any bay approaches or exceeds $\pi/2$ radians. This point is dealt with later.

A.L.2.

If both members are in tension the generalised equation of three moments becomes

$$\frac{a_1}{I_1} M_A\, F(\alpha_1) + \frac{a_2}{I_2} M_C\, F(\alpha_2) + 2M_B \left\{ \frac{a_1}{I_1} \Phi(\alpha_1) + \frac{a_2}{I_2} \Phi(\alpha_2) \right\} + \frac{3E}{2a_1}(\delta_B - \delta_A)$$

$$+ \frac{3E}{2a_2}(\delta_B - \delta_C) = \frac{w_1 a_1^{\,3}}{I_1}\, \Psi(\alpha_1) + \frac{w_2 a_2^{\,3}}{I_2}\, \Psi(\alpha_2) \quad \ldots \quad \ldots \quad \ldots \quad (3)$$

where $F(\alpha) = \frac{3}{2}\left\{\frac{1 - 2\,\alpha \operatorname{cosech} 2\,\alpha}{\alpha^2}\right\} = 2\,\Phi(\alpha) - \Phi\left(\frac{\alpha}{2}\right)$

$$\Phi(\alpha) = \frac{3}{4}\left\{\frac{2\,\alpha \coth 2\,\alpha - 1}{\alpha^2}\right\}$$

$$\Psi(\alpha) = 3\left\{\frac{\alpha - \tanh \alpha}{\alpha^3}\right\} = \frac{1}{3}\,\Phi\left(\frac{\alpha}{2}\right)\left\{4\,\Phi(\alpha) - \Phi\left(\frac{\alpha}{2}\right)\right\}$$

Tables for these hyperbolic functions and for $\tanh \alpha$ are given in tables II and III respectively.

If the end load, P, in one member, say AB, is zero the corresponding functions all become unity.

If one member, say AB, is in tension and the other BC, in compression, the three moments equation becomes

$$\frac{a_1}{I_1} M_A\, F(\alpha_1) + \frac{a_2}{I_2} M_C\, f(\alpha_2) + 2M_B \left\{ \frac{a_1}{I_1} \Phi(\alpha_1) + \frac{a_2}{I_2} \phi(\alpha_2) \right\} + \frac{3E}{2a_1}(\delta_B - \delta_A)$$

$$+ \frac{3E}{2a_2}(\delta_B - \delta_C) = \frac{w_1 a_1^{\,3}}{I_1}\, \Psi(\alpha_1) + \frac{w_2 a_2^{\,3}}{I_2}\, \psi(\alpha_2) \quad \ldots \quad \ldots \quad \ldots \quad (4)$$

(ii) *Maximum bending moment in bays.*—The bending moment at any point in a member *AB under compression* is given by

$$M = \frac{1}{2}(M_A - M_B)\,\frac{\sin \mu x}{\sin \alpha} + \frac{1}{2}(M_A + M_B)\,\frac{\cos \mu x}{\cos \alpha} + \frac{w}{\mu^2}\left(1 - \frac{\cos \mu\, x}{\cos \alpha}\right) \quad \ldots \quad (5)$$

$$\text{or } M = \frac{wa^2}{\alpha^2} - \left(\frac{wa^2}{\alpha^2} - \frac{M_A + M_B}{2}\right)\frac{\cos \mu x}{\cos \alpha} + \frac{M_A - M_B}{2}\,\frac{\sin \mu x}{\sin \alpha} \quad \ldots \quad \ldots \quad \ldots \quad (6)$$

For the bending moment M to be a maximum

$$\tan \mu x = -\frac{\frac{1}{2}(M_A + M_B)\cot \alpha}{\frac{wa^2}{\alpha^2} - \frac{1}{2}(M_A - M_B)} \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (7)$$

in which x is measured from the middle point of AB.

This equation is for a mathematical maximum or minimum of M. It should be noted therefore that although μx (which may be positive or negative) given by equation (7) may be numerically less than α it does not necessarily follow that the bending moment for the corresponding point is the greatest (ignoring sign) in the span as one or both of the fixing moments may be numerically greater than this value.

If μx given by equation (7) is numerically greater than α then there is no true maximum or minimum for the bending moment in the bay AB and in these circumstances M increases steadily from A to B or vice versa, the worst stresses in the bay occurring either at A or B.

The maximum value of M is given by :—

$$M_{max} = \frac{wa^2}{\alpha^2} - \frac{\dfrac{wa^2}{\alpha^2} - \dfrac{1}{2}(M_A + M_B)}{\cos \mu x \cos \alpha} \quad \ldots \quad (8)$$

μx being given by equation (7).

Taking M_{max} positively, the maximum fibre stress in the bay is given by the usual formula

$$f_s = \frac{M_{max}}{Z} + \frac{P}{A} \quad \ldots \quad (9)$$

where Z is the modulus of section and A is the area.

The bending moment at any point in a member AB *in tension* is

$$M = \tfrac{1}{2}(M_A - M_B)\frac{\sinh \mu x}{\sinh \alpha} + \tfrac{1}{2}(M_A + M_B)\frac{\cosh \mu x}{\cosh \alpha} - \frac{w}{\mu^2}\left(1 - \frac{\cosh \mu x}{\cosh \alpha}\right) \quad \ldots \quad (10)$$

and the maximum bending moment is

$$M_{max} = \frac{\tfrac{1}{2}(M_A + M_B) + \dfrac{wa^2}{\alpha^2}}{\cosh \mu x \cosh \alpha} - \frac{wa^2}{\alpha^2} \quad \ldots \quad (11)$$

where μx is given by

$$\tanh \mu x = -\frac{\tfrac{1}{2}(M_A - M_B)\coth \alpha}{\tfrac{1}{2}(M_A + M_B) + \dfrac{wa^2}{\alpha^2}} \quad \ldots \quad (12)$$

(iii) *Offsets*.—In the case of a spar with the lift-wires offset the usual modifications must be made in the above equations.

For example, if there is an offset at B, equation (1) becomes

$$\frac{a_1}{I_1} M_{AL} f(\alpha_1) + \frac{a_2}{I_2} M_{CR} f(\alpha_2) + 2\left\{\frac{a_1}{I_1} M_{BR}\, \phi(\alpha_1) + \frac{a_2}{I_2} M_{BL}\, \phi(\alpha_2)\right\} = \frac{w_1 a_1^3}{I_1}\psi(\alpha_1) + \frac{w_2 a_2^3}{I_3}\psi(\alpha_2) \quad \ldots \quad (13)$$

In the formulae for the bay AB under compression M_{AL} should be taken for M_A, and M_{BR} for M_B.

(iv) When the beam carries concentrated as well as lateral loads (*e.g.* the front spar of a slotted plane) or when the lateral loading changes at a point between two points of support, equation (1) becomes

$$\begin{aligned}\frac{a_1}{I_1} M_A f(\alpha_1) &+ 2M_B\left\{\frac{a_1}{I_1}\phi(\alpha_1) + \frac{a_2}{I_1}\phi(\alpha_2)\right\} + \frac{a_2}{I_2} M_C f(\alpha_2) \\ &= \frac{3a_1^2}{I_1 \alpha_1^2}\Bigg[R'_{BR} - \frac{a_1}{2\alpha_1}(F_1 \sec \alpha_1 + H_1 \operatorname{cosec} \alpha_1) + \tfrac{1}{2}(G_1 \sec \alpha_1 - K_1 \operatorname{cosec} \alpha_1) \\ &\quad + \frac{a_1}{2\alpha_1}\left\{(w_n - w_o)\cot \alpha_1 + (w_n + w_o)\tan \alpha_1\right\}\Bigg] \\ &+ \frac{3a_2^2}{I_2 \alpha_2^2}\Bigg[R'_{BL} - \frac{a_2}{2\alpha_2}(F_2 \sec \alpha_2 - H_2 \operatorname{cosec} \alpha_2) + \tfrac{1}{2}(G_2 \sec \alpha_2 + K_2 \operatorname{cosec} \alpha_2) \\ &\quad - \frac{a_2}{2\alpha_2}\left\{(w_n - w_o)\cot \alpha_2 - (w_n + w_o)\tan \alpha_2\right\}\Bigg] \quad \ldots \quad (14)\end{aligned}$$

Where (*see* fig. 6) :—

$$\theta_r = \mu x_r$$

$$F = \sum_{r=1}^{r=n} \left\{ (w_r - w_{r-1}) \sin \theta_r \right\}$$

$$H = \sum_{r=1}^{r=n} \left\{ (w_r - w_{r-1}) \cos \theta_r \right\}$$

$$G = \sum_{r=1}^{r=n} (W_r \cos \theta_r)$$

$$K = \sum_{r=1}^{r=n} (W_r \sin \theta_r)$$

R'_A and R'_B are the reactions at A and B due to lateral loads when the continuity of the beam is neglected. The sign convention is that loads and reactions are positive upwards, so that R'_A and R'_B will usually be negative. It should be noted that the expression on the right-hand side referring to Bay 1 is not identical with that referring to Bay 2. If the loading is symmetrical about the centre line in any bay H, K and $(w_n - w_o)$ vanish. The application of this equation to consecutive bays will give sufficient equations for the calculation of the bending moments at the points of support. When the bending moments at the points of support have been calculated as above, the bending moment diagram can be drawn as described in R. & M. 1233.

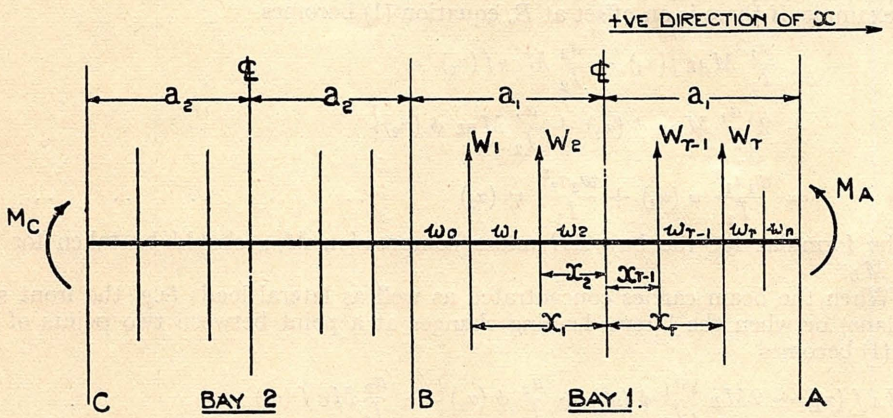

FIG. 6.—CHAP. VI.

TABLE I

$$f(\alpha) = \frac{3\,(2\,\alpha \operatorname{cosec} 2\alpha - 1)}{2\,\alpha^2} \qquad \phi(\alpha) = \frac{3\,(1 - 2\,\alpha \cot 2\,\alpha)}{4\,\alpha^2} \qquad \psi(\alpha) = \frac{3\,(\tan \alpha - \alpha)}{\alpha^3}$$

α	$f(\alpha)$	Diff.	$\phi(\alpha)$	Diff.	$\psi(\alpha)$	Diff.
Radians.						
0·00	1·000	·001	1·000	·000	1·000	·001
0·04	1·001	·002	1·000	·002	1·001	·002
0·08	1·003	·004	1·002	·002	1·003	·003
0·12	1·007	·005	1·004	·003	1·006	·004
0·16	1·012	·007	1·007	·004	1·010	·006
0·20	1·019	·009	1·011	·005	1·016	·008
0·24	1·028	·010	1·016	·006	1·024	·008
0·28	1·038		1·022		1·032	
0·28	1·038	·006	1·022	·003	1·032	·005
0·30	1·044	·006	1·025	·003	1·037	·006
0·32	1·050	·007	1·028	·004	1·043	·005
0·34	1·057	·007	1·032	·004	1·048	·006
0·36	1·064	·008	1·036	·005	1·054	·007
0·38	1·072	·008	1·041	·004	1·061	·007
0·40	1·080	·009	1·045	·005	1·068	·008
0·42	1·089	·009	1·050	·006	1·076	·008
0·44	1·098	·010	1·056	·005	1·084	·009
0·46	1·108	·011	1·061	·006	1·093	·009
0·48	1·119	·011	1·067	·007	1·102	·009
0·50	1·130	·012	1·074	·007	1·111	·010
0·52	1·142	·013	1·081	·007	1·121	·011
0·54	1·155	·014	1·088	·007	1·132	·012
0·56	1·169	·014	1·095	·008	1·144	·012
0·58	1·183	·015	1·103	·008	1·156	·012
0·60	1·198	·016	1·111	·009	1·168	·014
0·62	1·214	·017	1·120	·010	1·182	·014
0·64	1·231	·018	1·130	·010	1·196	·016
0·66	1·249	·019	1·140	·010	1·212	·016
0·68	1·268	·020	1·150	·011	1·228	·017
0·70	1·288	·022	1·161	·012	1·245	·017
0·72	1·310	·022	1·173	·012	1·262	·020
0·74	1·332	·023	1·185	·013	1·282	·020
0·76	1·355	·025	1·198	·014	1·302	·021
0·78	1·380	·028	1·212	·015	1·323	·023

TABLE I—*cont.*

α	$f(\alpha)$	Diff.	$\phi(\alpha)$	Diff.	$\psi(\alpha)$	Diff.
·078		28		15		23
Radians.						
0·80	1·408	·029	1·227	·016	1·346	·024
0·82	1·437	·030	1·243	·016	1·370	·025
0·84	1·467	·032	1·259	·017	1·395	·027
0·86	1·499	·034	1·276	·019	1·422	·029
0·88	1·533	·037	1·295	·020	1·451	·031
0·90	1·570	·039	1·315	·021	1·482	·033
0·92	1·609	·043	1·336	·022	1·515	·034
0·94	1·652	·046	1·358	·025	1·549	·039
0·96	1·698	·049	1·383	·026	1·588	·042
0·98	1·747	·053	1·409	·027	1·630	·042
1·00	1·800	·056	1·436	·030	1·672	·047
1·02	1·856	·061	1·466	·033	1·719	·050
1·04	1·917	·067	1·499	·034	1·769	·052
1·06	1·984	·072	1·533	·037	1·821	·058
1·08	2·056	·078	1·570	·042	1·879	·069
1·10	2·134	·085	1·612	·045	1·948	·071
1·12	2·219	·093	1·657	·049	2·019	·080
1·14	2·312	·103	1·706	·054	2·099	·087
1·16	2·415	·116	1·760	·059	2·186	·090
1·18	2·531	·128	1·819	·066	2·276	·102
1·20	2·659		1·885		2·378	
1·200	2·659	·034	1·885	·017	2·378	·028
1·205	2·693	·035	1·902	·018	2·406	·028
1·210	2·728	·036	1·920	·018	2·434	·028
1·215	2·764	·037	1·938	·020	2·462	·032
1·220	2·801	·039	1·958	·020	2·494	·033
1·225	2·840	·039	1·978	·020	2·527	·033
1·230	2·879	·040	1·998	·021	2·560	·033
1·235	2·919	·043	2·019	·022	2·593	·034
1·240	2·962	·044	2·041	·022	2·627	·036
1·245	3·006	·044	2·063	·023	2·663	·038
1·250	3·050	·046	2·086	·024	2·701	·040
1·255	3·096	·047	2·110	·024	2·741	·040
1·260	3·143	·048	2·134	·025	2·781	·040
1·265	3·191	·050	2·159	·026	2·821	·041
1·270	3·241	·052	2·185	·027	2·862	·041
1·275	3·293	·055	2·212	·027	2·903	·043
1·280	3·348	·057	2·239	·029	2·946	·046
1·285	3·405	·059	2·268	·030	2·992	·050
1·290	3·464	·061	2·298	·031	3·042	·050
1·295	3·525	·063	2·329	·032	3·092	·051

TABLE I—*cont.*

α	f (α)	Diff.	ϕ (α)	Diff.	ψ (α)	Diff.
Radians.						
1·300	3·588	·065	2·361	·033	3·143	·053
1·305	3·653	·066	2·394	·034	3·196	·055
1·310	3·719	·070	2·428	·036	3·251	·056
1·315	3·789	·073	2·464	·037	3·307	·059
1·320	3·862	·077	2·501	·039	3·366	·062
1·325	3·939	·080	2·540	·040	3·428	·066
1·330	4·019	·084	2·580	·043	3·494	·071
1·335	4·103	·087	2·623	·045	3·565	·073
1·340	4·190	·091	2·668	·046	3·638	·074
1·345	4·281	·095	2·714	·048	3·712	·075
1·350	4·376	·099	2·762	·050	3·787	·076
1·355	4·475	·102	2·812	·051	3·863	·081
1·360	4·577	·107	2·863	·054	3·944	·089
1·365	4·684	·113	2·917	·058	4·033	·097
1·370	4·797		2·975		4·130	
1·3700	4·797	·062	2·975	·031	4·130	·049
1·3725	4·859	·063	3·006	·031	4·179	·049
1·3750	4·922	·064	3·037	·032	4·228	·050
1·3775	4·986	·065	3·069	·033	4·278	·050
1·3800	5·051	·065	3·102	·033	4·328	·052
1·3825	5·116	·066	3·135	·034	4·380	·057
1·3850	5·182	·068	3·169	·035	4·437	·060
1·3875	5·250	·071	3·204	·036	4·497	·062
1·3900	5·321	·073	3·240	·037	4·559	·062
1·3925	5·394	·077	3·277	·039	4·621	·062
1·3950	5·471	·078	3·316	·039	4·683	·062
1·3975	5·549	·080	3·355	·040	4·745	·064
1·4000	5·629	·082	3·395	·041	4·809	·064
1·4025	5·711	·085	3·436	·043	4·873	·067
1·4050	5·796	·089	3·479	·045	4·940	·073
1·4075	5·885	·090	3·524	·046	5·013	·078
1·4100	5·975	·093	3·570	·047	5·091	·079
1·4125	6·068	·096	3·617	·048	5·170	·081
1·4150	6·164	·103	3·665	·052	5·251	·082
1·4175	6·267	·104	3·717	·052	5·333	·083
1·4200	6·371	·105	3·769	·053	5·416	·084
1·4225	6·476	·107	3·822	·054	5·500	·086
1·4250	6·583	·112	3·876	·056	5·586	·094
1·4275	6·695	·119	3·932	·061	5·680	·103

TABLE I—*cont.*

α	$f(\alpha)$	Diff.	$\phi(\alpha)$	Diff.	$\psi(\alpha)$	Diff.
Radians.						
1·4300	6·814	·122	3·993	·062	5·783	·105
1·4325	6·936	·127	4·055	·064	5·888	·106
1·4350	7·063	·131	4·119	·066	5·994	·107
1·4375	7·194	·136	4·185	·068	6·101	·109
1·4400	7·330	·143	4·253	·071	6·210	·110
1·4425	7·473	·147	4·324	·074	6·320	·115
1·4450	7·620	·153	4·398	·077	6·435	·122
1·4475	7·773	·160	4·475	·080	6·557	·130
1·4500	7·933	·167	4·555	·084	6·687	·136
1·4525	8·100	·174	4·639	·087	6·823	·140
1·4550	8·274	·182	4·726	·091	6·963	·140
1·4575	8·456	·190	4·817	·095	7·103	·150
1·4600	8·646	·198	4·912	·099	7·253	·161
1·4625	8·844	·208	5·011	·105	7·414	·171
1·4650	9·052	·217	5·116	·109	7·585	·178
1·4675	9·269	·229	5·225	·115	7·763	·187
1·4700	9·498	·240	5·340	·120	7·950	·193
1·4725	9·738	·253	5·460	·127	8·143	·203
1·4750	9·991	·267	5·587	·134	8·346	·218
1·4775	10·258		5·721		8·564	
1·4775	10·26	·28	5·721	·142	8·564	·236
1·4800	10·54	·30	5·863	·149	8·800	·245
1·4825	10·84	·31	6·012	·157	9·045	·256
1·4850	11·15	·33	6·169	·166	9·301	·263
1·4875	11·48	·36	6·335	·177	9·564	·280
1·4900	11·84		6·512		9·844	
1·4900	11·84	·37	6·512	·189	9·84	·30
1·4925	12·21	·41	6·701	·202	10·14	·33
1·4950	12·62	·43	6·903	·214	10·47	·36
1·4975	13·05	·46	7·117	·230	10·83	·37
1·5000	13·51	·49	7·347	·246	11·20	·40
1·5025	14·00	·53	7·593	·266	11·60	·42
1·5050	14·53	·57	7·859	·287	12·02	·45
1·5075	15·10	·62	8·146	·310	12·47	·50
1·5100	15·72	·67	8·456	·337	12·97	·55
1·5125	16·39	·74	8·793	·367	13·52	·60
1·5150	17·13	·80	9·160	·402	14·12	·66
1·5175	17·93	·88	9·562	·442	14·78	·72
1·5200	18·81		10·004		15·50	

For values of α between 1·52 and 1·62 the functions $f(\alpha)$, $\phi(\alpha)$ and $\psi(\alpha)$ can be calculated with sufficient accuracy from the approximations

$$f(\alpha) = \frac{6}{\pi(\pi - 2\alpha)} = \frac{1{\cdot}910}{\pi - 2\alpha}$$

$$\phi(\alpha) = \frac{3}{\pi(\pi - 2\alpha)} + \frac{6}{\pi^2} = \frac{{\cdot}955}{\pi - 2\alpha} + {\cdot}608$$

$$\psi(\alpha) = \frac{48}{\pi^3(\pi - 2\alpha)} + \frac{144}{\pi^4} - \frac{12}{\pi^2} = \frac{1{\cdot}547}{\pi - 2\alpha} + {\cdot}262$$

TABLE I—*cont.*

α	$f(\alpha)$	Diff.	$\phi(\alpha)$	Diff.	$\psi(\alpha)$	Diff.
Radians.						
1·6200	−19·42	·94	−9·092	·478	−15·48	·76
1·6225	−18·48	·85	−8·624	·426	−14·72	·69
1·6250	−17·63	·77	−8·198	·389	−14·03	·63
1·6275	−16·86	·71	−7·809	·355	−13·40	·57
1·6300	−16·15	·66	−7·454	·327	−12·83	·53
1·6325	−15·49	·60	−7·127	·305	−12·30	·49
1·6350	−14·89	·57	−6·825	·285	−11·81	·47
1·6375	−14·32	·50	−6·540	·252	−11·34	·40
1·6400	−13·82		−6·288		−10·94	
1·640	−13·82	·93	−6·288	·466	−10·936	·755
1·645	−12·89	·81	−5·822	·405	−10·181	·655
1·650	−12·08	·72	−5·417	·360	−9·526	·584
1·655	−11·36		−5·057		−8·942	
1·655	−11·364	·635	−5·057	·316	−8·942	·515
1·660	−10·729	·567	−4·739	·285	−8·427	·462
1·665	−10·162	·509	−4·454	·255	−7·965	·414
1·670	−9·653	·461	−4·199	·231	−7·551	·375
1·675	−9·192	·418	−3·968	·211	−7·176	·340
1·680	−8·774	·382	−3·757	·191	−6·836	·311
1·685	−8·392	·350	−3·566	·177	−6·525	·281
1·690	−8·042	·320	−3·389	·161	−6·244	·263
1·695	−7·722	·298	−3·228	·149	−5·981	·243
1·700	−7·424		−3·079		−5·738	
1·70	−7·424	·986	−3·079	·496	−5·732	·812
1·72	−6·438	·749	−2·583	·381	−4·920	·596
1·74	−5·689	·589	−2·202	·300	−4·324	·481
1·76	−5·100	·478	−1·902	·243	−3·843	·390
1·78	−4·622	·393	−1·659	·202	−3·453	·322
1·80	−4·229	·328	−1·457	·169	−3·131	·270

TABLE I—*cont.*

α	$f\ (\alpha)$	Diff.	$\phi\ (\alpha)$	Diff.	$\psi\ (\alpha)$	Diff.
Radians.						
1·82	−3·901		−1·288		−2·861	
		·278		·145		·226
1·84	−3·623		−1·143		−2·635	
		·239		·125		·200
1·86	−3·384		−1·018		−2·435	
		·208		·109		·174
1·88	−3·176		−0·909		−2·261	
		·180		·096		·150
1·90	−2·996		−0·813		−2·111	
		·159		·086		·133
1·92	−2·837		−·727		−1·978	
		·141		·077		·119
1·94	−2·696		−·650		−1·859	
		·126		·069		·108
1·96	−2·570		−·581		−1·751	
		·111		·063		·093
1·98	−2·459		−·518		−1·658	
		·102		·058		·088
2·00	−2·357		−·460		−1·570	
		·092		·053		·081
2·02	−2·265		−·407		−1·489	
		·081		·048		·070
2·04	−2·184		−·359		−1·419	
		·075		·046		·066
2·06	−2·109		−·313		−1·353	
		·069		·041		·064
2·08	−2·040		−·272		−1·289	
		·061		·040		·055
2·10	−1·979		−·232		−1·234	
		·056		·037		·052
2·12	−1·923		−·195		−1·182	
		·052		·035		·049
2·14	−1·871		−·160		−1·133	
		·047		·033		·046
2·16	−1·824		−·127		−1·087	
		·042		·032		·043
2·18	−1·782		−·095		−1·044	
		·040		·030		·040
2·20	−1·742		−·065		−1·004	
		·035		·028		·035
2·22	−1·707		−·037		−·969	
		·032		·028		·034
2·24	−1·675		−·009		−·935	
		·029		·026		·032
2·26	−1·646		+·017		−·903	
		·026		·026		·031
2·28	−1·620		+·043		−·872	
		·023		·025		·029
2·30	−1·597		+·068		−·843	
		·021		·025		·028
2·32	−1·576		+·093		−·815	
		·019		·025		·025
2·34	−1·557		+·118		−·790	
		·016		·024		·025
2·36	−1·541		+·142		−·765	

TABLE II

$$F(\alpha) = \frac{3\,(1 - 2\alpha \operatorname{cosech} 2\alpha)}{2\alpha^2} \qquad \Phi(\alpha) = \frac{3\,(2\alpha \coth 2\alpha - 1)}{4\alpha^2} \qquad \Psi(\alpha) = \frac{3(\alpha - \tanh \alpha)}{\alpha^3}$$

α	$F(\alpha)$	$\Phi(\alpha)$	$\Psi(\alpha)$	α	$F(\alpha)$	$\Phi(\alpha)$	$\Psi(\alpha)$
Radians.				Radians.			
·00	1·0000	1·0000	1·0000	·40	·9301	·9598	·9399
·01	0·9999	1·0000	0·9999	·41	·9268	·9579	·9371
·02	·9998	0·9999	·9998	·42	·9234	·9559	·9342
·03	·9996	·9998	·9996	·43	·9200	·9539	·9312
·04	·9992	·9996	·9994	·44	·9165	·9519	·9282
·05	·9988	·9993	·9990	·45	·9129	·9499	·9252
·06	·9983	·9990	·9986	·46	·9094	·9478	·9221
·07	·9977	·9987	·9981	·47	·9057	·9456	·9189
·08	·9970	·9983	·9975	·48	·9020	·9435	·9157
·09	·9962	·9979	·9968	·49	·8983	·9413	·9125
·10	·9953	·9973	·9960	·50	·8945	·9391	·9092
·11	·9944	·9968	·9952	·51	·8906	·9369	·9059
·12	·9933	·9962	·9943	·52	·8868	·9346	·9026
·13	·9922	·9955	·9933	·53	·8828	·9323	·8992
·14	·9909	·9948	·9922	·54	·8789	·9300	·8957
·15	·9896	·9940	·9910	·55	·8748	·9276	·8922
·16	·9882	·9932	·9898	·56	·8708	·9253	·8887
·17	·9867	·9924	·9886	·57	·8667	·9229	·8851
·18	·9851	·9915	·9872	·58	·8626	·9204	·8815
·19	·9834	·9905	·9857	·59	·8584	·9180	·8779
·20	·9816	·9895	·9842	·60	·8542	·9155	·8743
·21	·9798	·9884	·9826	·61	·8500	·9130	·8706
·22	·9779	·9873	·9810	·62	·8457	·9105	·8669
·23	·9758	·9862	·9793	·63	·8414	·9080	·8632
·24	·9738	·9850	·9775	·64	·8371	·9054	·8595
·25	·9716	·9837	·9756	·65	·8328	·9028	·8557
·26	·9693	·9824	·9736	·66	·8284	·9003	·8519
·27	·9670	·9811	·9717	·67	·8240	·8977	·8481
·28	·9646	·9797	·9696	·68	·8196	·8950	·8442
·29	·9621	·9783	·9675	·69	·8151	·8924	·8403
·30	·9595	·9768	·9653	·70	·8107	·8897	·8364
·31	·9569	·9753	·9630	·71	·8062	·8871	·8325
·32	·9542	·9737	·9607	·72	·8017	·8844	·8286
·33	·9514	·9721	·9583	·73	·7972	·8817	·8247
·34	·9486	·9705	·9558	·74	·7927	·8790	·8207
·35	·9457	·9688	·9533	·75	·7881	·8762	·8167
·36	·9427	·9671	·9507	·76	·7835	·8735	·8127
·37	·9396	·9653	·9481	·77	·7790	·8708	·8087
·38	·9365	·9635	·9454	·78	·7744	·8680	·8047
·39	·9333	·9617	·9427	·79	·7698	·8653	·8007

TABLE II—*cont.*

α	$F(\alpha)$	$\Phi(\alpha)$	$\Psi(\alpha)$	α	$F(\alpha)$	$\Phi(\alpha)$	$\Psi(\alpha)$
Radians.				Radians.			
·80	·7652	·8625	·7967	1·20	·5843	·7499	·6360
·81	·7606	·8597	·7927	1·21	·5801	·7472	·6322
·82	·7560	·8569	·7887	1·22	·5759	·7444	·6284
·83	·7513	·8541	·7847	1·23	·5717	·7417	·6246
·84	·7467	·8513	·7807	1·24	·5675	·7390	·6208
·85	·7421	·8485	·7766	1·25	·5633	·7363	·6170
·86	·7374	·8457	·7725	1·26	·5592	·7336	·6133
·87	·7328	·8429	·7684	1·27	·5551	·7309	·6096
·88	·7282	·8400	·7643	1·28	·5510	·7282	·6059
·89	·7235	·8372	·7601	1·29	·5469	·7255	·6022
·90	·7189	·8344	·7560	1·30	·5429	·7229	·5985
·91	·7143	·8315	·7519	1·31	·5389	·7202	·5948
·92	·7096	·8287	·7478	1·32	·5349	·7175	·5912
·93	·7050	·8259	·7437	1·33	·5309	·7149	·5875
·94	·7004	·8230	·7396	1·34	·5269	·7123	·5839
·95	·6958	·8202	·7355	1·35	·5230	·7097	·5803
·96	·6912	·8173	·7314	1·36	·5191	·7071	·5767
·97	·6866	·8145	·7273	1·37	·5152	·7045	·5732
·98	·6820	·8117	·7232	1·38	·5114	·7019	·5697
·99	·6774	·8088	·7192	1·39	·5075	·6993	·5662
1·00	·6728	·8060	·7152	1·40	·5037	·6967	·5627
1·01	·6683	·8031	·7112	1·41	·4999	·6942	·5593
1·02	·6637	·8003	·7072	1·42	·4962	·6916	·5559
1·03	·6592	·7975	·7031	1·43	·4924	·6891	·5525
1·04	·6547	·7946	·6991	1·44	·4887	·6866	·5491
1·05	·6501	·7918	·6950	1·45	·4851	·6840	·5457
1·06	·6456	·7890	·6910	1·46	·4814	·6815	·5423
1·07	·6411	·7861	·6870	1·47	·4778	·6790	·5389
1·08	·6367	·7833	·6830	1·48	·4742	·6766	·5355
1·09	·6322	·7805	·6790	1·49	·4706	·6741	·5321
1·10	·6278	·7777	·6750	1·50	·4670	·6716	·5288
1·11	·6233	·7749	·6711	1·51	·4635	·6692	·5255
1·12	·6189	·7721	·6672	1·52	·4600	·6668	·5222
1·13	·6145	·7693	·6633	1·53	·4565	·6643	·5189
1·14	·6102	·7665	·6594	1·54	·4530	·6619	·5157
1·15	·6058	·7637	·6555	1·55	·4496	·6595	·5125
1·16	·6015	·7609	·6516	1·56	·4462	·6571	·5093
1·17	·5972	·7582	·6477	1·57	·4428	·6547	·5061
1·18	·5929	·7554	·6438	1·58	·4395	·6524	·5030
1·19	·5886	·7527	·6399	1·59	·4361	·6500	·4999
				1·60	·4328	·6476	·4968

TABLE III

Tanh α

α	Tanh α	α	Tanh α	α	Tanh α	α	Tanh α
Radians.		Radians.		Radians.		Radians.	
0·00	·00000	0·40	·37995	0·80	·66404	1·20	·83365
·01	·01000	·41	·38847	·81	·66959	1·21	·83668
·02	·02000	·42	·39693	·82	·67507	1·22	·83965
·03	·02999	·43	·40532	·83	·68048	1·23	·84258
·04	·03998	·44	·41364	·84	·68581	1·24	·84546
·05	·04996	·45	·42190	·85	·69107	1·25	·84828
·06	·05993	·46	·43008	·86	·69626	1·26	·85106
·07	·06989	·47	·43820	·87	·70137	1·27	·85380
·08	·07983	·48	·44624	·88	·70642	1·28	·85648
·09	·08976	·49	·45422	·89	·71139	1·29	·85913
0·10	·09967	0·50	·46212	0·90	·71630	1·30	·86172
·11	·10956	·51	·46995	·91	·72113	1·31	·86427
·12	·11943	·52	·47770	·92	·72590	1·32	·86678
·13	·12928	·53	·48538	·93	·73059	1·33	·86925
·14	·13909	·54	·49299	·94	·73522	1·34	·87167
·15	·14888	·55	·50052	·95	·73978	1·35	·87405
·16	·15865	·56	·50798	·96	·74428	1·36	·87639
·17	·16838	·57	·51536	·97	·74870	1·37	·87869
·18	·17808	·58	·52267	·98	·75307	1·38	·88095
·19	·18775	·59	·52990	·99	·75736	1·39	·88317
0·20	·19738	0·60	·53705	1·00	·76159	1·40	·88535
·21	·20697	·61	·54413	1·01	·76576	1·41	·88749
·22	·21652	·62	·55113	1·02	·76987	1·42	·88960
·23	·22603	·63	·55805	1·03	·77391	1·43	·89167
·24	·23550	·64	·56490	1·04	·77789	1·44	·89370
·25	·24492	·65	·57167	1·05	·78181	1·45	·89569
·26	·25430	·66	·57836	1·06	·78566	1·46	·89765
·27	·26362	·67	·58598	1·07	·78946	1·47	·89958
·28	·27291	·68	·59152	1·08	·79320	1·48	·90147
·29	·28213	·69	·59798	1·09	·79688	1·49	·90332
0·30	·29131	0·70	·60437	1·10	·80050	1·50	·90515
·31	·30044	·71	·61068	1·11	·80406	1·51	·90694
·32	·30951	·72	·61691	1·12	·80757	1·52	·90870
·33	·31852	·73	·62307	1·13	·81102	1·53	·91043
·34	·32748	·74	·62915	1·14	·81441	1·54	·91212
·35	·33638	·75	·63515	1·15	·81775	1·55	·91379
·36	·34521	·76	·64108	1·16	·82104	1·56	·91542
·37	·35399	·77	·64693	1·17	·82427	1·57	·91703
·38	·36271	·78	·65271	1·18	·82745	1·58	·91860
·39	·37136	·79	·65841	1·19	·83058	1·59	·92015

TABLE III—*cont.*

α	Tanh α	α	Tanh α	α	Tanh α	α	Tanh α
Radians.		Radians.		Radians.		Radians.	
1·60	·92167	1·70	·93541	1·80	·94681	1·90	·95624
1·61	·92316	1·71	·93665	1·81	·94783	1·91	·95709
1·62	·82462	1·72	·93786	1·82	·94884	1·92	·95792
1·63	·92606	1·73	·93906	1·83	·94983	1·93	·95873
1·64	·92747	1·74	·94023	1·84	·95080	1·94	·95953
1·65	·92886	1·75	·94138	1·85	·95175	1·95	·96032
1·66	·93022	1·76	·94250	1·86	·95268	1·96	·96109
1·67	·93155	1·77	·94361	1·87	·95359	1·97	·96185
1·68	·93286	1·78	·94470	1·88	·95449	1·98	·96259
1·69	·93415	1·79	·94576	1·89	·95537	1·99	·96331
						2·00	·96403

ii.—Generalised Theorem of Three Moments extended to include shear deflection

The generalised theorem of three moments given above takes account of the deflection due to bending moment only. Some cases have arisen where there is a considerable discrepancy between the calculated deflection of a spar and the deflection actually measured on test. If this discrepancy is due to shear deflection it can be taken into account as follows :—

In general it will not be possible to calculate the shear deflection of such metal spars as are met with in aircraft without first ascertaining the value of a constant by means of tests on the particular spar section in question. For simple solid rectangular sections an approximate method of calculating the shear deflection is given in " The Strength of Materials ", by John Case, chapter XIV, but at the present time no simple method has been evolved for dealing with the complicated sections usually employed in metal spars.

In the work that follows it will be assumed that tests have been carried out to determine a constant " r " such that :—

$$\frac{dy_s}{dx} = -rS$$

where S is the shear and y_s is the deflection due to shear. Suitable tests for this purpose are described later.

(i) *Notation.*—The notation is the same as that used in section A with the following additions :—

r = see above.

y_b = deflection of the spar at any point due to bending.

y_s = deflection of the spar at any point due to shear.

$y = y_b + y_s$.

S = shear force at any point.

$$\mu^2 = \frac{P}{EI\,(1 - rP)}$$

$$\alpha^2 = \frac{Pa^2}{EI\,(1 - rP)}$$

(ii) *Mathematical theory.*

$$\frac{dy_s}{dx} = -rS = -r\frac{dM}{dx}$$

$$\frac{d^2y}{dx^2} = -\frac{rd^2M}{dx^2} + \frac{M}{EI}$$

i.e. $$EI\frac{d^2y}{dx^2} = M - r\,EI\frac{d^2M}{dx^2}$$

This is the fundamental equation, reducing to the ordinary form when $r = 0$, *i.e.* when the shear deflection is zero.

By taking moments about any point of a member AB (*see* fig. 5) under compression, we have

$M = -Py + \frac{1}{2}w\,(a-x)^2 + S_A\,(x-a) + M_A$, where S_A is the shear to the left of A.

Hence $$\frac{d^2M}{dx^2} = -\frac{P\,d^2y}{dx^2} + w$$

or $$EI\frac{d^2y}{dx^2} = -\frac{EI}{P}\frac{d^2M}{dx^2} + \frac{EIw}{P}$$

Thus $$EI\,(1-rP)\frac{d^2M}{dx^2} + PM = EIw$$

or $$\frac{d^2M}{dx^2} + \mu^2M = \frac{w}{1-rP}$$

This equation is identical with that obtained neglecting shear, except for the modified definition of μ^2, and that w is replaced by $\frac{w}{1-rP}$.

When forming the three moment equation $\frac{dy_b}{dx}$ should be equated for the two bays and not $\frac{dy}{dx}$. This leads to the following expression for the three moment equation :—

$$\left\{\frac{a_1}{I_1}f\,(\alpha_1) - \frac{3Er_1}{2a_1}\right\}M_A + \left\{\frac{a_2}{I_2}f\,(\alpha_2) - \frac{3Er_2}{2a_2}\right\}M_C + 2M_B\left\{\frac{a_1}{I_1}\phi\,(\alpha_1)\right.$$
$$\left. + \frac{a_2}{I_2}\phi\,(\alpha_2) + \frac{3E}{4}\left(\frac{r_1}{a_1} + \frac{r_2}{a_2}\right)\right\} = \frac{w_1\,a_1^{\,3}}{I_1\,(1-rP)}\;\psi\,(\alpha_1) + \frac{w_2\,a_2^{\,3}}{I_2\,(1-rP)}\;\psi\,(\alpha_2)$$

The expression for the bending moment at any point and for the maximum bending moment can be obtained from equations (6) and (8), section A, by using the above definitions of μ^2 and α^2 and replacing w by w' where

$$w' = \frac{w}{1-rP}$$

(iii) *Experimental determination of the shear constant* r.—The following two tests have been found to give a satisfactory determination of this constant. In order that the deflection may be easily measurable, the length of the specimen should be about 20 times its depth.

(*a*) *Pure bending.*—The object of this test is to obtain the correct value of EI for the specimen, so that any error in calculating I or any variation in E from the specified value will not have any influence upon the value of r.

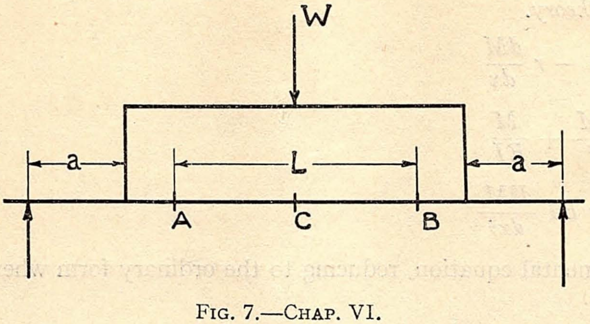

FIG. 7.—CHAP. VI.

The test is shown diagrammatically in fig. 7. The deflection of the centre point C relative to A and B is noted for a range of values of W and hence, by plotting, the ratio $\frac{W}{y_b}$ is found.

Then $$EI = \frac{aL^2W}{16y_b}$$

(*b*) *Bending and shear.*—This test is shown diagrammatically in fig. 8.

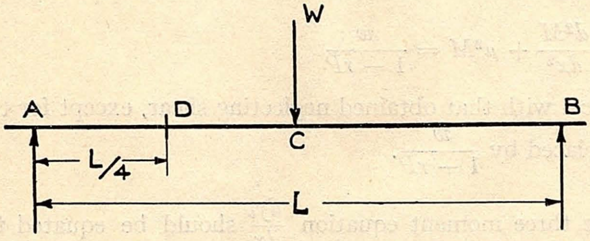

FIG. 8.—CHAP. VI.

The deflection of C relative to A and B is noted as before for a range of values of W.

Hence, by plotting, $\frac{y}{W}$ is obtained.

Then $y_s = y - y_b$

$$= \left\{\frac{y}{W} - \frac{L^3}{48\ EI}\right\} W$$

But $y_s = \frac{rLW}{4}$.

Hence $$r = \frac{4}{L}\left\{\frac{y}{W} - \frac{L^3}{48\,EI}\right\}$$

It is a useful check to take readings at another point D. If D be distant $\frac{L}{4}$ from one support the total deflection of D should be :—

$$\frac{11\ WL^3}{768\ EI} + \frac{r\,WL}{8}$$

iii.—Miscellaneous applications of the Generalised Theorem of Three Moments

(i) *Case 1. Generalised three moment equations where the compressive end load in one bay approximates to the Euler failing load for that bay.*—When the value of α in any bay approaches $\pi/2$ (corresponding to the Euler failing load for that bay considered as a pin-jointed strut) the Berry equations become unworkable because of the rapid variation of the Berry functions with α. The case is dealt with in the "Transactions of the Royal Aeronautical Society," No. 1, by writing $\theta = \pi/2 - \alpha$ and replacing the Berry functions by approximate expressions in terms of θ and π.

The three moment equation for the spars AB and BC, where α_1, corresponding to the bay AB, approximates to $\frac{\pi}{2}$, is then as follows :—

$$a_1 M_A + M_B \left[a_1 + 1{\cdot}047\ \theta_1 \left\{ 1{\cdot}216\ a_1 + 2a_2\ \phi\ (\alpha_2) \right\} \right] + 1{\cdot}047\ \theta_1\ a_2 f\,(\alpha_2)\ M_C$$
$$= {\cdot}811\ w_1 a_1^{\,3}\ (1 + 1{\cdot}273\ \theta_1) + 1{\cdot}047\ \theta_1 \left\{ w_2'\ a_2^{\,3} \psi\ (\alpha_2) - {\cdot}723\ w_1\ a_1^{\,3} \right\}$$

where $\theta_1 = \frac{\pi}{2} - \alpha_1$, θ_1 and α_1 being in radians.

The bending moment at any point in the bay AB distant x from its midpoint is given by the following expression :—

$$M = \frac{w\ a_1^{\,2}}{\alpha_1^{\,2}} + S \cos \mu x + \frac{M_A - M_B}{2}\ \frac{\sin \mu x}{\sin \alpha_1}$$

where

$$S = {\cdot}524 \left[\frac{w_2\ a_2^{\,3}\ \psi\ (\alpha_2)}{a_1} - {\cdot}723\ w_1 a_1^{\,2} - \left\{ 1{\cdot}216 + \frac{2a_2\ \phi\ (\alpha_2)}{a_1} \right\} M_B - \frac{a_2 f\,(\alpha_2)\ M_C}{a_1} \right]$$

The maximum bending moment occurs at x_1 where

$$\tan \mu x_1 = \frac{M_A - M_B}{2S}$$

and is given by

$$M_{max} = \frac{w a_1^{\,2}}{\alpha_1^{\,2}} + S \sec \mu x_1$$

It will be seen that in the particular case when $\alpha_1 = \frac{\pi}{2}$, *i.e.* when $\theta_1 = 0$, the term in M_C disappears from the three moment equation so that this equation only connects the moments at either end of the weaker span. This does not mean that M_C is indeterminate.

(ii) *Case 2. Instability criteria for continuous spars with end load.*—As will be seen from the previous case, a continuous spar subjected to end load will not, in general, fail when the end load in any one bay is equal to the Euler failing load for that bay considered as a pin-jointed strut.

The indication of instability given by the Berry equations is that the fixing moment between two bays becomes infinite or indeterminate as shown by the denominator of the expression for the fixing moment being zero.

For a spar with only one unknown fixing moment, as occurs, for instance, in a two bay aeroplane with a pin joint at the centre, the three moment equation is :—

$$\frac{a_1}{I_1} f\,(\alpha_1)\ M_A + \frac{a_2}{I_2} f\,(\alpha_2)\ M_C + 2M_B \left\{ \frac{a_1}{I_1}\ \phi\ (\alpha_1) + \frac{a_2}{I_2}\ \phi\ (\alpha_2) \right\}$$
$$= \frac{w_1\ a_1^{\,3}}{I_1}\ \psi\ (\alpha_1) + \frac{w_2\ a_2^{\,3}}{I_2}\ \psi\ (\alpha_2)$$

where M_A and M_C are known, M_A being the bending moment at the overhang and M_C a pin joint (or vice versa). This equation, therefore, gives M_B directly, the denominator of the expression for M_B being :—

$$2\left\{\frac{a_1}{I_1}\phi(\alpha_1) + \frac{a_2}{I_2}\phi(\alpha_2)\right\}$$

Thus, M_B will become infinite or indeterminate when :—

$$\frac{a_1}{I_1}\phi(\alpha_1) + \frac{a_2}{I_2}\phi(\alpha_2) = 0.$$

This relation between α_1 and α_2 is shown graphically in fig. 9. For any given value of α_1 the value of α_2 is given, corresponding to any particular value of $\frac{a_2/I_2}{a_1/I_1}$ that will make the bending moment at B (and hence at every other point in the spans ABC) indeterminate or infinite. Hence, if α_2 has a value less than this the spar will be stable, the moment at any point being calculable in the ordinary way, even though in one bay α may exceed 90°. The form of the usual equations being unsuitable for values of α between $1\cdot52$ and $1\cdot62$ radians the modified form given above (Case 1) should be used.

In a similar way it can be shown that in a single bay aircraft with the spar continuous over the centre section, the condition for instability is :—

$$2\left\{\frac{a_1}{I_1}\phi(\alpha_1) + \frac{a_2}{I_2}\phi(\alpha_2)\right\} + \frac{a_2}{I_2}f(\alpha_2) = 0$$

This relation is shown graphically in fig. 10.

In more complicated cases where there are two or more unknown fixing moments, the conditions for instability become too cumbersome to be conveniently represented on a graph. In any particular case the required condition is easily found by solving for one fixing moment and equating the denominator of the expression to zero.

The presence of offsets does not affect the conditions of instability. This is easily seen by replacing in equation (6)

M_{BL} by M_B
and M_{BR} by $M_B + Ph$

h being the amount by which the wire is offset. Then the coefficient of M_B is as given above.

Instability criteria for a continuous strut divided into a number of bays could be obtained on these lines as the instability is not affected by the magnitude of the lateral load.

(iii) *Case 3. An aeroplane spar in which the dihedral starts from a point between two supports.* —A convenient method of dealing with this case is to treat the point at which the dihedral starts as a support which has deflected below the line joining the supports on either side, and inserting the condition that the reaction at this support is zero. At each undeflected support the end load is resolved along the line joining that point to the next undeflected support, the other component being taken by the reaction at the support.

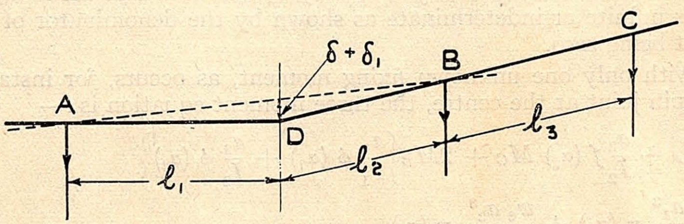

FIG. 11.—CHAP. VI.

INSTABILITY CRITERIA FOR SPARS ACCORDING TO THE GENERALISED THEOREM OF THREE MOMENTS.

For any given value of M and $\left\{ \begin{smallmatrix} \alpha_1 \\ \alpha_2 \end{smallmatrix} \right\}$ the graph shows the value of $\left\{ \begin{smallmatrix} \alpha_2 \\ \alpha_1 \end{smallmatrix} \right\}$ which corresponds to instability. It is necessary to check that this value has not been exceeded, as it may not be immediately apparent from the three-moment equations that the instability point has been passed.

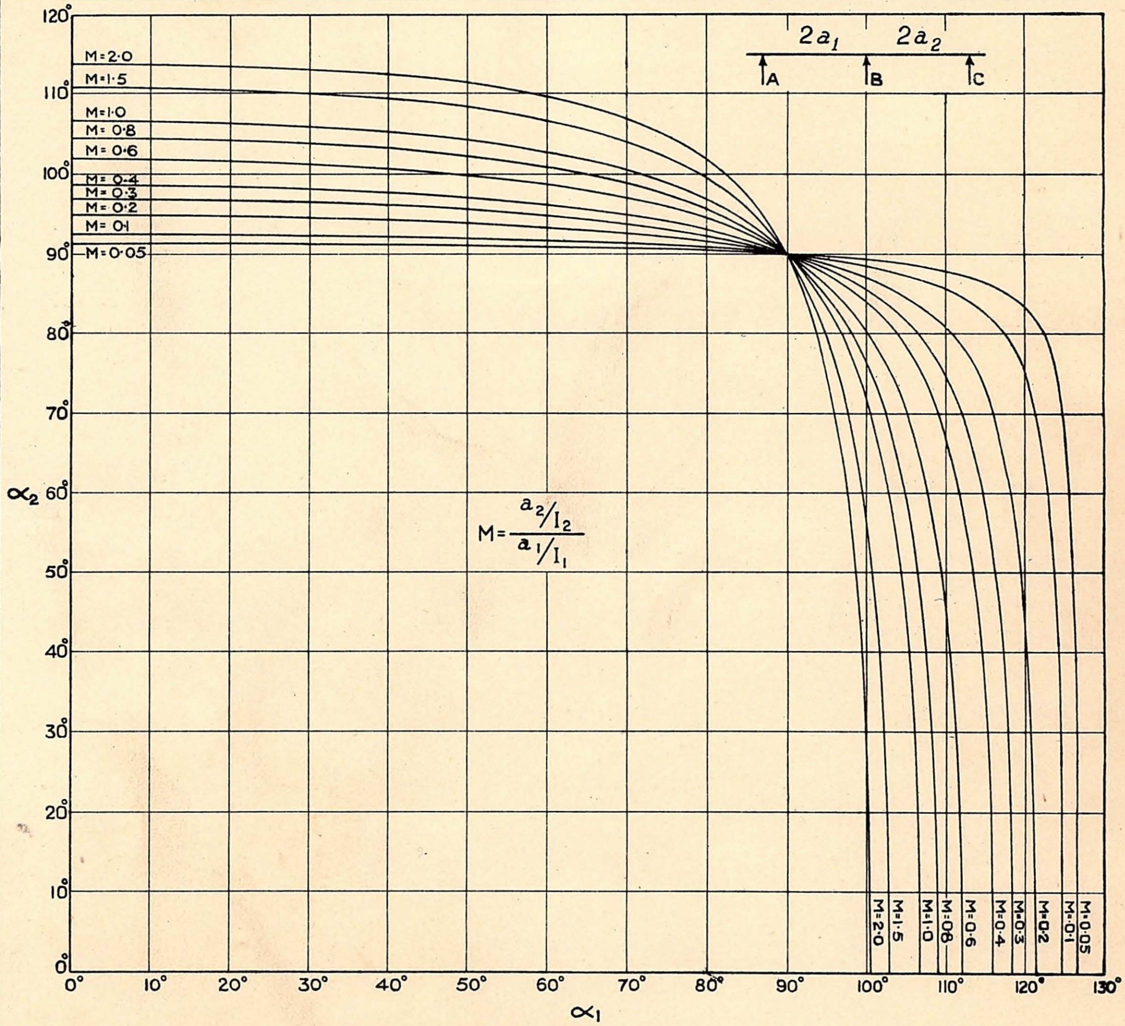

FIG. 9. CHAP. VI.

INSTABILITY CRITERIA FOR SPARS ACCORDING TO THE GENERALISED THEOREM OF THREE MOMENTS.

For any given value of M and $\left\{\frac{\alpha_1}{\alpha_2}\right\}$ the graph shows the value of $\left\{\frac{\alpha_2}{\alpha_1}\right\}$ which corresponds to instability. It is necessary to check that this value has not been exceeded, as it may not be immediately apparent from the three-moment equations that the instability point has been passed.

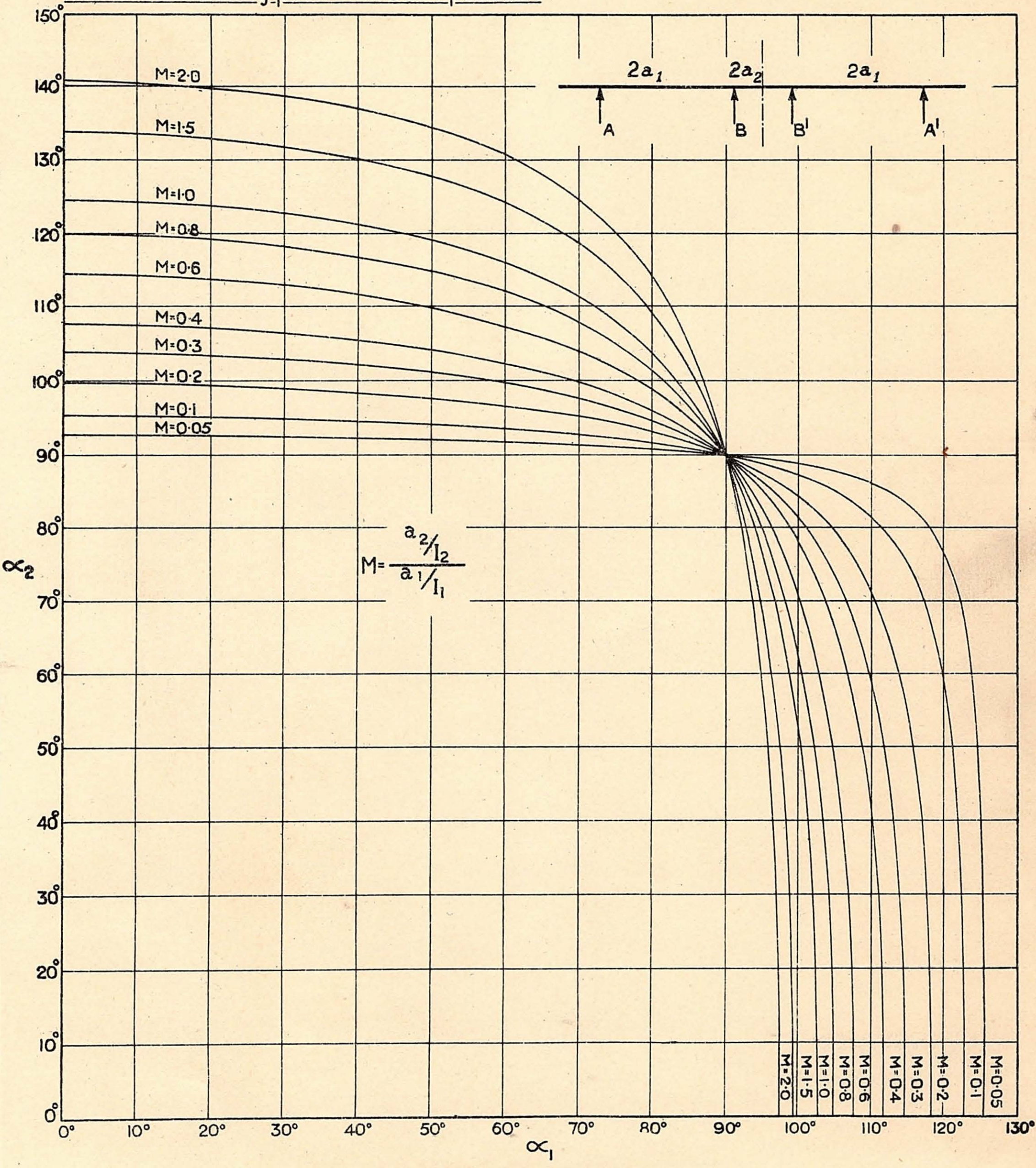

FIG. 10. CHAP. VI.

Referring to fig. 11 let

w_1 and P_1 be uniform up load and end load respectively in bay l_1.

w_2 and P_2 be uniform up load and end load respectively in bay l_2.

Assume reactions positive when acting in the direction shown.

The deflection of D (positive when D is above AB) is the sum of two terms $(\delta + \delta_1)$, where δ_1 is the deflection when the beam is unloaded and δ is the increased deflection under load. δ_1 is known from the geometry of the structure (it will be negative for the configuration of fig. 11), and the five unknowns δ, M_A, M_B, M_C, M_D are given by the following three equations together with two end conditions :—

$$\frac{l_1}{I_1} M_A f(\alpha_1) + \frac{l_2}{I_2} M_B f(\alpha_2) + 2\, M_D \left\{ \frac{l_1}{I_1} \phi(\alpha_1) + \frac{l_2}{I_2} \phi(\alpha_2) \right\} + 6E\, \delta \left(\frac{1}{l_1} + \frac{1}{l_2} \right)$$

$$= \frac{w_1\, l_1^3}{4\, I_1}\, \psi(\alpha_1) + \frac{w_2\, l_2^3}{4\, I_2}\, \psi(\alpha_2) \qquad \ldots \quad (1)$$

$$\frac{l_2}{I_2} M_D f(\alpha_2) + \frac{l_3}{I_3} M_C f(\alpha_3) + 2\, M_B \left\{ \frac{l_2}{I_2} \phi(\alpha_2) + \frac{l_3}{I_3} \phi(\alpha_3) \right\} - \frac{6E\delta}{l_2}$$

$$= \frac{w_2\, l_2^3}{4\, I_2}\, \psi(\alpha_2) + \frac{w_3\, l_3^3}{4\, I_3}\, \psi(\alpha_3) \qquad \ldots \quad (2)$$

$$0 = \frac{w_1\, l_1 + w_2\, l_2}{2} + \frac{M_D - M_A}{l_1} + \frac{M_D - M_B}{l_2} + \left\{ \frac{P_1}{l_1} + \frac{P_2}{l_2} \right\} (\delta + \delta_1) \qquad \ldots \quad (3)$$

This treatment assumes that the end load acts along the straight line joining the two undeflected supports on each side of the deflected supports. This is not strictly accurate when part of the end load is due to the drag bracing, as this will come on to the spar at the drag bay nodes which will usually have deflected above or below the straight line joining the undeflected supports. In most cases, however, the drag bracing contribution at any deflected drag bay node will be a small percentage of the total end load and so the error involved in the above assumption will be small. It will usually be sufficiently accurate to take the mean end load so that instead of dealing with P_1 and P_2, the mean load P could be taken as acting over both bays l_1 and l_2, where

$$P = \tfrac{1}{2}(P_1 + P_2)$$

(iv) *Case* 4. *Eccentrically loaded struts with concentrated lateral loads.—(a) One lateral load only* :—

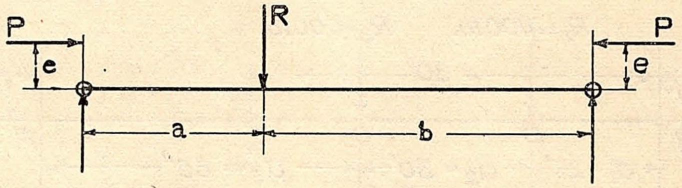

FIG. 12.—CHAP. VI.

Referring to fig. 12.

M_{max} occurs at a point distant x from the outside end of the longer bay (bay b), where x (measured positive towards the concentrated load) is given by :—

$$\tan \mu\, x = \tan \frac{\mu\,(a+b)}{2} + \frac{R \sin \mu\, a}{Pe\, \mu \sin \mu\,(a+b)}$$

and

$$M_{max} = Pe \sec \mu\, x.$$

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If x is greater than $\frac{b}{\text{[illegible]}}$, M_{max} occurs at the concentrated load, and is given by :—

$$M_{max} = \frac{R \sin \mu\, a \sin \mu\, b}{\mu \sin \mu\, (a + b)} + \frac{Pe\, (\sin \mu\, a + \sin \mu\, b)}{\sin \mu\, (a + b)}$$

(*b*) *Several equal lateral loads arranged symmetrically.*

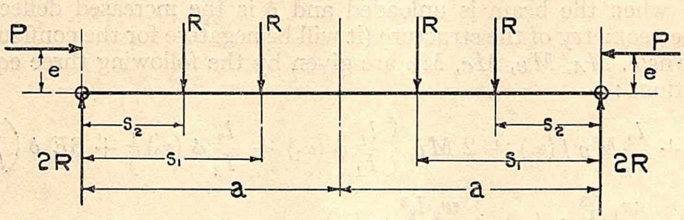


FIG. 13.—CHAP. VI.

Referring to fig. 13.

For 2 pairs of equal loads.—M_{max} occurs at the mid-point of the strut and is given by :—

$$M_{max} = \frac{R}{\mu} \sec \mu\, a\, (\sin \mu\, s_1 + \sin \mu\, s_2) + Pe \sec \mu\, a.$$

For n pairs of equal loads :—

$$M_{max} = \frac{R}{\mu} \sec \mu\, a\, (\sin \mu\, s_1 + \sin \mu\, s_2 + \ldots + \sin \mu\, s_{n-1} + \sin \mu\, s_n) + Pe \sec \mu\, a.$$

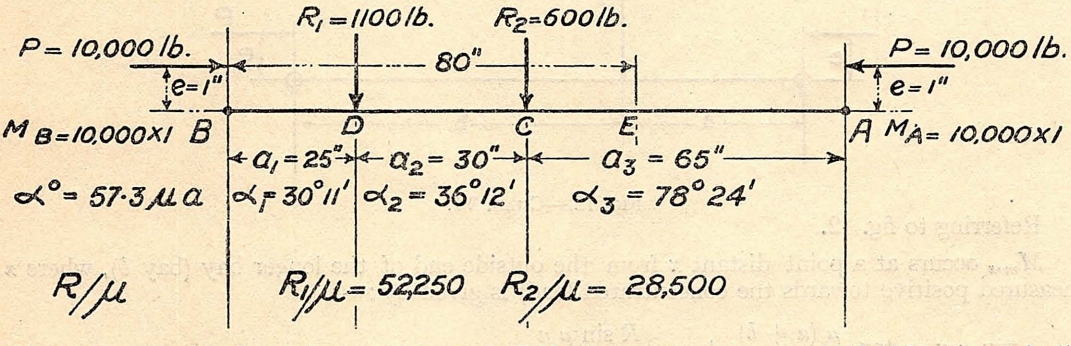
(*c*) *Two unequal loads unsymmetrically placed.*—The easiest way of determining the maximum bending moment and bending moment diagram is by the polar diagram method described in R. & M. 1233. A numerical example is given below.

Example.—Required to draw the bending moment diagram for a strut 120 in. long, end load 10,000 lb., eccentricity of loading 1 in. and concentrated lateral loads of 1,100 and 600 lb. distant 25 in. and 55 in. respectively from one end.

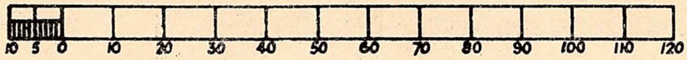
$$E = 1 \cdot 5 \times 10^6 \text{ lb. per sq. in. } I = 15 \text{ ins.}^4.$$

Tabulate the necessary data as follows :—

$$\mu = \sqrt{\frac{P}{EI}} = \cdot 02105 \quad \frac{1}{\mu} = 47 \cdot 5\,''.$$



Construction (*see* fig. 14).—Set out angle $B'OA' = \alpha_1 + \alpha_2 + \alpha_3 = 144° \; 47'$ symmetrically about a vertical. Set off angle $B'OD' = \alpha_1 = 30° \; 11'$ and angle $A'OC' = \alpha_3 = 78° \; 24'$. Mark off OB and $OA = M_B$ and M_B respectively, in this case 10,000 lb. ins.



Scale of Bending Moments in Thousands of lb. ins.

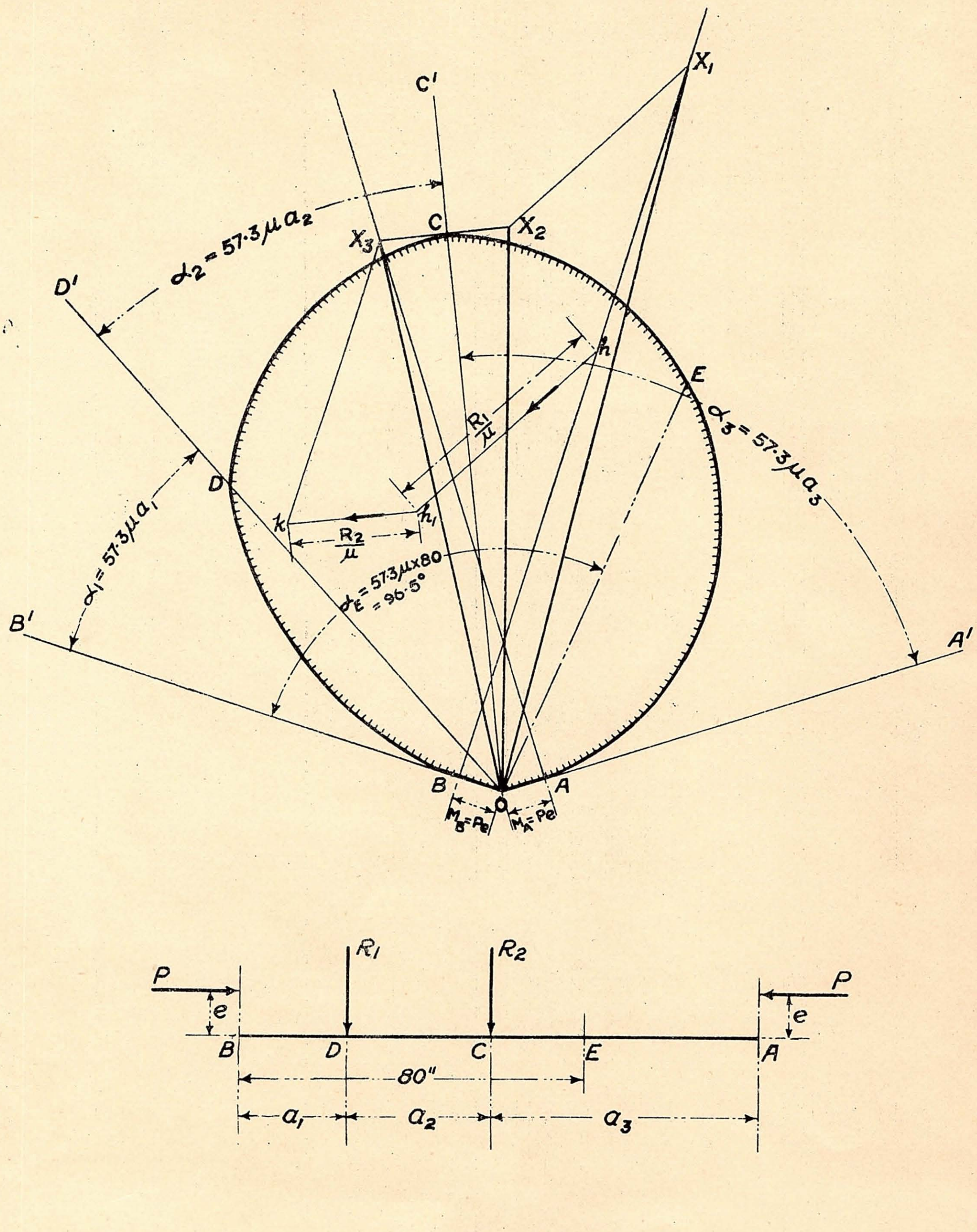


FIG. 14, CHAP. VI.

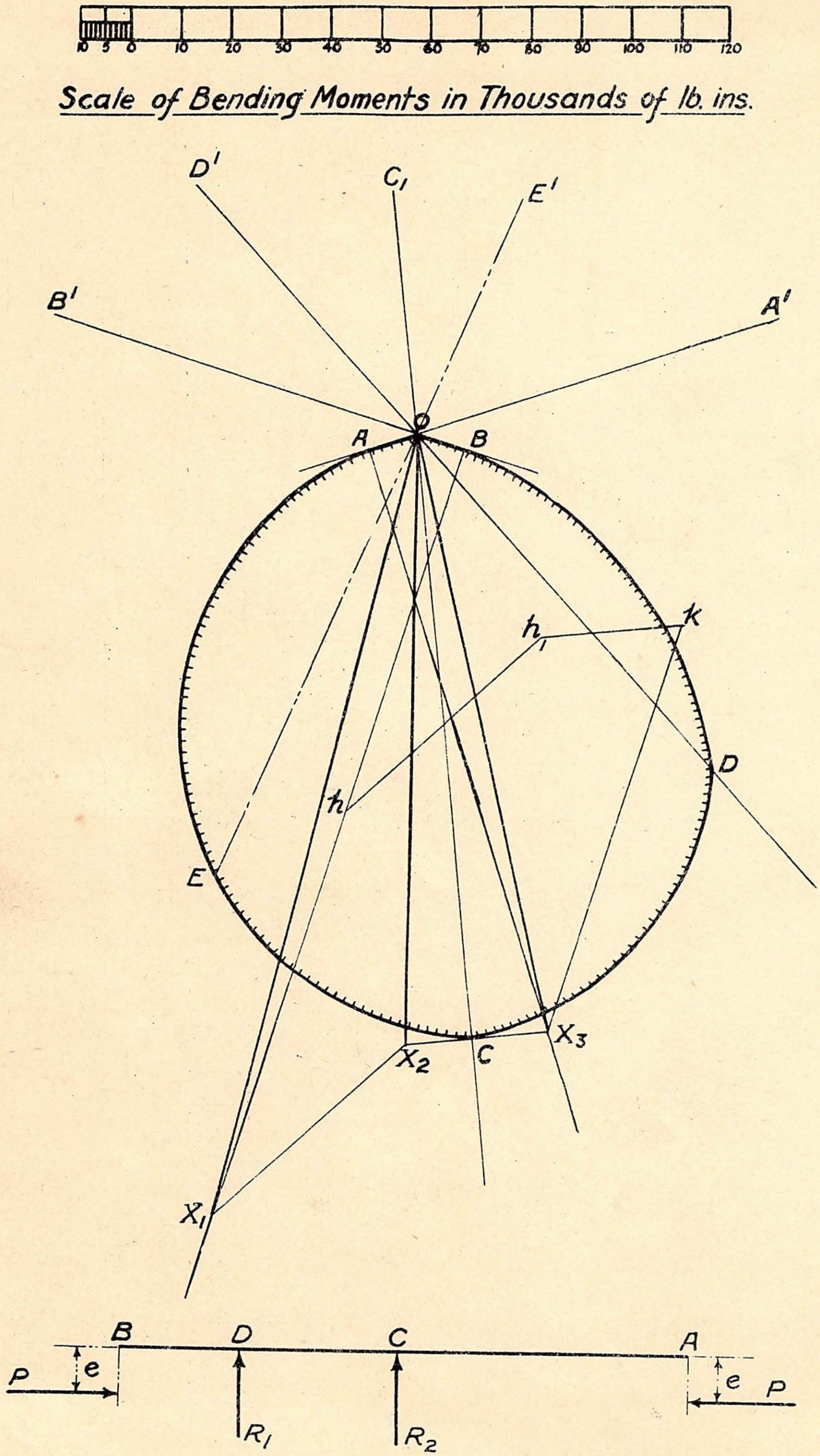
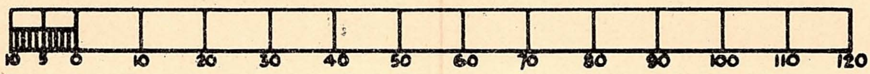
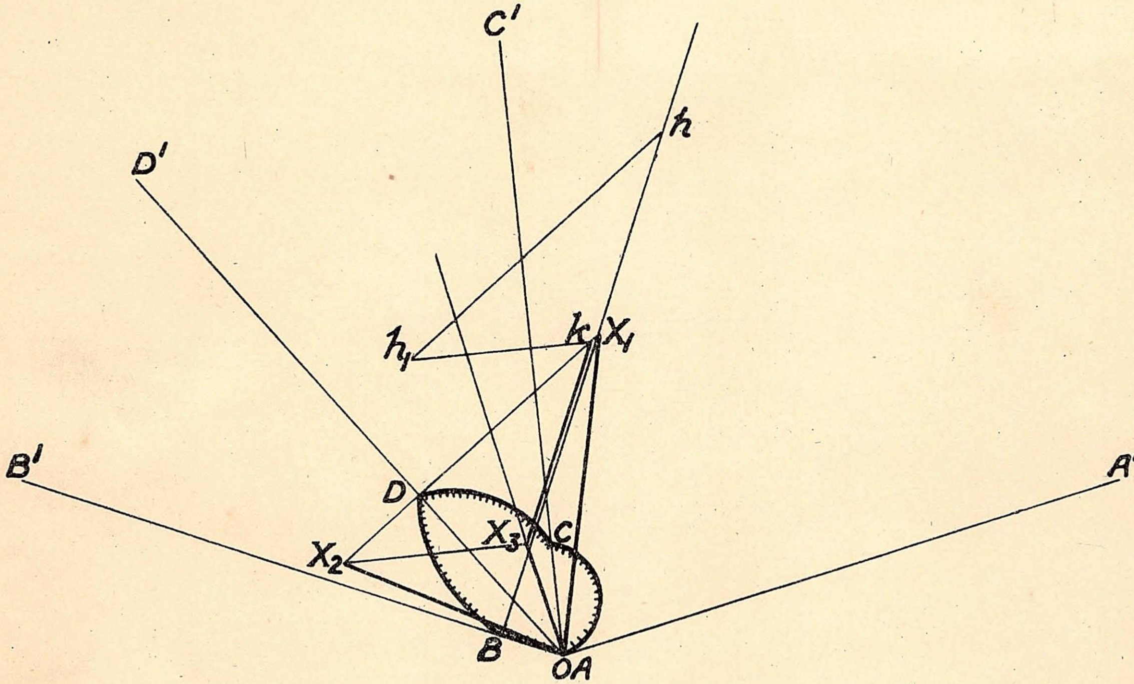


FIG. 15, CHAP. VI.



Scale of Bending Moments in Thousands of lb. ins.



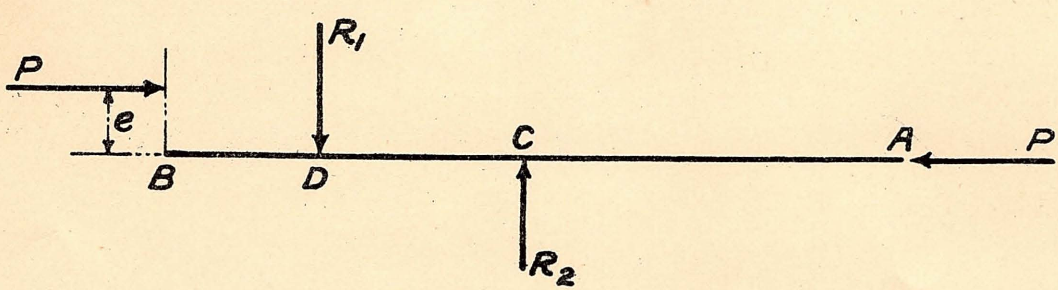


FIG. 16, CHAP. VI.

The following sign conventions should be rigidly adhered to. Bending moments tending to make the beam deflect thus $\smile$ are considered positive and should be marked off along the radius vector *upwards* from 0. Up loads on the beam are considered positive and should be marked off (*see* below) at right angles to the radius vector and *to the right* when looking upwards along the radius vector.

Erect perpendiculars at B and A and through any point h on perpendicular at B draw hh_1 at right angles to OD' and equal to R_1/μ, *i.e.* 52,250. R_1 being a down load, hh_1 is drawn to the left when looking upwards along OD'. Draw h_1k at right angles to OC' and equal to R_2/μ, *i.e.* 28,500. Again h_1k is drawn to the left because R_2 is a down load. Draw kX_3 parallel to Bh, meeting perpendicular at A in X_3. Draw X_3X_2 parallel, equal to, and in the same sense as kh_1 and X_2X_1 parallel, equal to, and in the same sense as h_1h so that X_1 falls on Bh produced. On OX_1 as diameter, describe an arc starting from B, moving in clockwise direction and ending on OD at D (D is on X_1X_2 produced). On OX_2 as diameter, describe an arc from D in clockwise direction and ending on OC' at C (C is on X_2X_3). On OX_3 as diameter, describe an arc from C to A. These three arcs form the bending moment diagram. Thus the bending moment at a point E distant 80 ins. from B is represented in fig. 14 by OE, the angle $B'OE$ being $57{\cdot}3\ \mu \times 80 = 96{\cdot}5°$. The maximum bending moment occurs in this case at C and is equal to OC.

As a further illustration of the method of drawing the diagram, fig. 15 shows the form it takes when R_1 and R_2 act upwards and the eccentricity is in the opposite direction.

The bending moment diagram is of course identical but of different sign.

A third example, fig. 16 shows the shape of the diagram when R_1 and R_2 are in opposite directions and the eccentricity at A is zero.

CHAPTER VII.—THE AERODYNAMIC LOAD DISTRIBUTION ON TAPERED AND TWISTED WINGS

(Monoplanes only. *See* chapter V, section I, para. 1.)

1. General.—(i) This chapter deals with the calculation of the shape of the curve of load distribution along the span of a wing, the chord of which varies from wing root to wing tip. An acceptable alternative method involving the use of " induction factors " is given in R. & M. No. 1643. A method is also given for calculating the pitching moment of such a wing.

(ii) Prior to the development of the Prandtl theory, tapered wings were commonly dealt with on the assumption that each section was independent of those on either side, this method being known as the " Strip Theory." It can be shown that in many cases calculations based on this theory may lead to an over-estimation of the strength of the wing structure, and hence the more accurate method described in this chapter should be used in preference to the " Strip Theory."

(iii) A twisted wing is defined as one in which the no-lift lines of all the sections are not parallel. This may be due to a geometrical twist which tilts the chord lines of the various sections relative to each other, or it may be due to an aerodynamic twist caused by a variation in the no-lift angles of the aerofoils forming the wing at different points along the span. Both these effects are often present simultaneously. For the purpose of the calculations described in this chapter the total twist only is of importance, and it is immaterial how much of this is due to the one cause and how much to the other.

(iv) For an untwisted wing the shape of the load distribution curve is the same for both normal flight cases and for the terminal velocity dive and inverted flight cases. This is not so when the wing is twisted as then the load distribution curve may vary considerably with incidence, particularly at low angles of incidence.

(v) The calculations described below deal with a wing the root of which is on the centre line of the aeroplane. In para. 3 (ix) a correction is given to take account of the loss in lift due to the presence of the fuselage which masks, or replaces, a portion of the wing at the centre. The root chord, c_0, of the wing is found by producing the leading and trailing edges of the actual wing up to the centre line of the aeroplane. The characteristics of the aerofoil section at the wing root (*i.e.*, no-lift angle, slope of lift-incidence curve, etc.) may be found by plotting the known characteristics for the various sections along the actual wing and producing by eye the curves so formed beyond the point at which the wing merges into the fuselage up to the centre line of the aeroplane.

2. Notation.—(i) The notation employed is as shown in fig. 1, together with the following :—

α = angle of incidence of any section of the wing measured (*in radians*) from the no-lift line of that particular section.

a = slope of the curve of lift coefficient against angle of incidence (radians) for the section considered, corrected to infinite aspect ratio (*see* below).

c = chord of section.

$\left.\begin{matrix} a_0 \\ \alpha_0 \\ c_0 \end{matrix}\right\}$ similar quantities at the wing root, *i.e.*, on the centre line of the aeroplane (*see* above).

$$u = \frac{ac}{4s}$$

$$\cos\theta = \frac{x}{s} \text{ (\textit{see} fig. 1).}$$

S = area of complete wing (port and starboard) including that portion of the wing masked or replaced by the fuselage, referred to in the text as the " assumed wing."

λ = tip chord/root chord for straight tapered wing with maximum chord at the wing root.

k_L = mean lift coefficient of the whole wing, assuming the centre (fuselage) portion of the wing to be a lifting surface.

A = aspect ratio of wing = $\dfrac{(2s)^2}{S}$.

s = semi-span.

w = loading in pounds per inch run along span.

(ii) *Determination of " a."*—The slope a, of the lift curve for two-dimensional flow is always very close to π. In the absence of any data the value 3·0 should be used. Generally, tests will be available of finite wings of aspect ratio A the slope of the lift curve being a_A. If the results have not been corrected for wind tunnel interference this must first be done.

(*a*) If a' is the observed slope then a_A the true slope for the aspect ratio A is given by

$$a_A = \frac{a'}{1 + \cdot 274 \dfrac{S_T a'}{C}}$$

where S_T is the area of the model wing and C the cross sectional area of the tunnel. This formula applies to square closed tunnels and to the duplex tunnel when the model is tested horizontally.

(*b*) a_A is now corrected to infinite aspect ratio by the curve given in fig. 2. This shows the ratio $\dfrac{a}{a_A}$ as a function $\dfrac{A}{a_A}$, where A is the aspect ratio of the tested wing.

(*c*) The no-lift angle of the finite wing is the same as that of the infinite wing.

3. Lift coefficient.—(i) *General method.*—The method consists in expressing the load distribution curve as a Fourier's series and determining the coefficients by fitting the Fourier expression to the characteristics of the wing at selected points. To reduce the labour involved in calculating the load distribution curve for any given wing the analysis has been carried as far as possible in general terms, the point at which the general treatment must give place to detailed numerical treatment being dependent upon the type of wing concerned.

(ii) It is convenient to divide tapered wings into the following groups :—

(*a*) Straight tapered untwisted wings. In this type the leading and trailing edges are straight, the chord decreasing uniformly from the root to the tip.

(*b*) Straight tapered twisted wings.

(*c*) Curved tapered untwisted wings.

(*d*) Curved tapered twisted wings.

These four types are dealt with below in this order, the simplest type being considered first.

(iii) *Straight tapered untwisted wings.*—The load distribution curve for a straight tapered wing can be obtained directly from fig. 3. This gives several curves corresponding to a series of values of λ, where

$$\lambda = \frac{\text{chord at wing tip.}}{\text{chord at wing root.}}$$

In practice the wing tip will usually be rounded to some extent, in which case the tip chord is determined by producing the leading and trailing edges to cut the tangent at the wing tip.

The top half of fig. 4 is derived from fig. 3 by cross plotting and is convenient for interpolation. The values of the ordinates in fig. 4 correspond to a root incidence of 1 radian from no-lift, the linear relationship between k_L and α being assumed to hold up to this incidence.

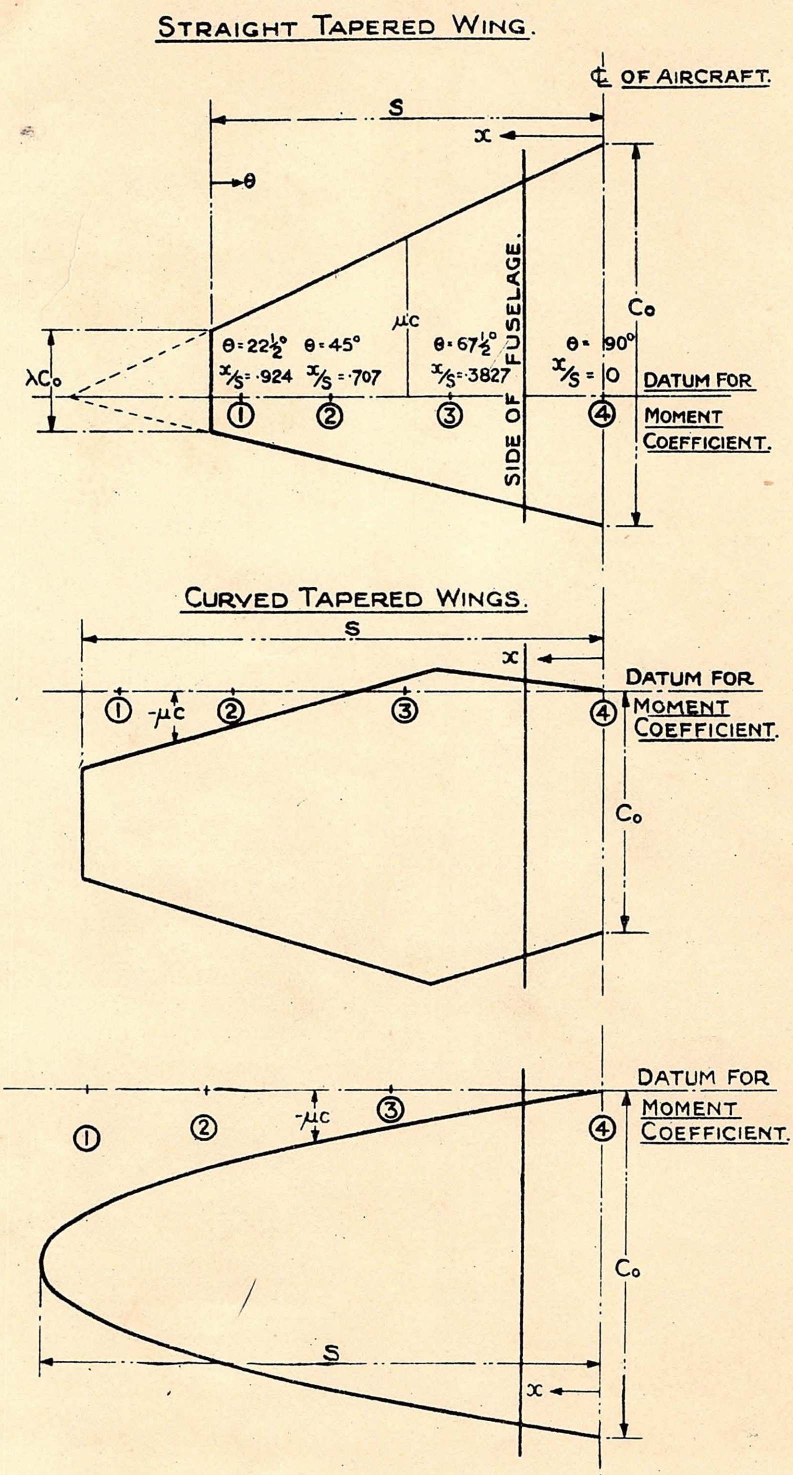


FIG. 1 Chap. VII.

Relation Between Slope of Lift Curve of Infinite and Finite Wing.

A = Aspect Ratio of Tested Wing.

$a_A = \frac{dk_L}{d\alpha}$ = Observed Slope of Lift Curve. (α Being in Radians).

a = Slope of Lift Curve of Same Section in 2 Dimensional Flow.

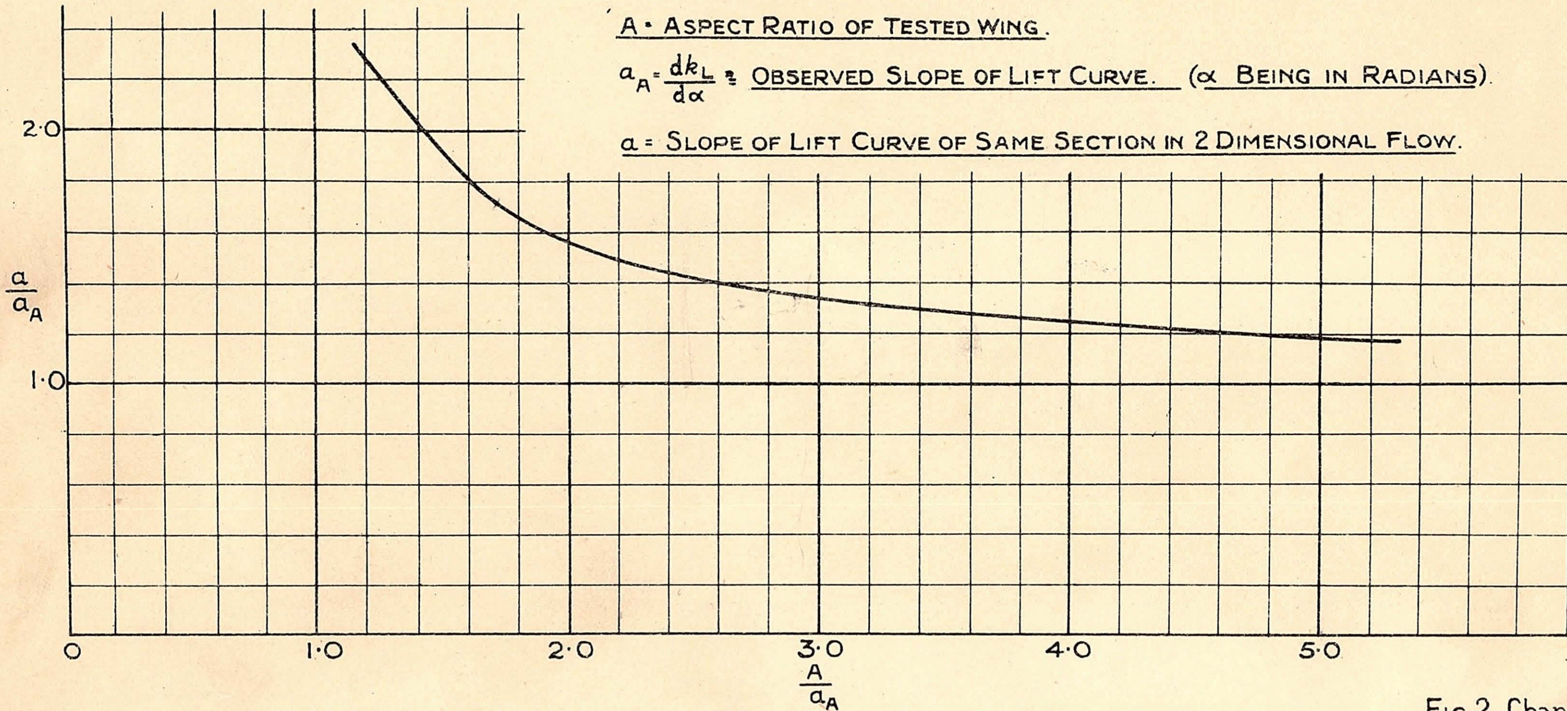
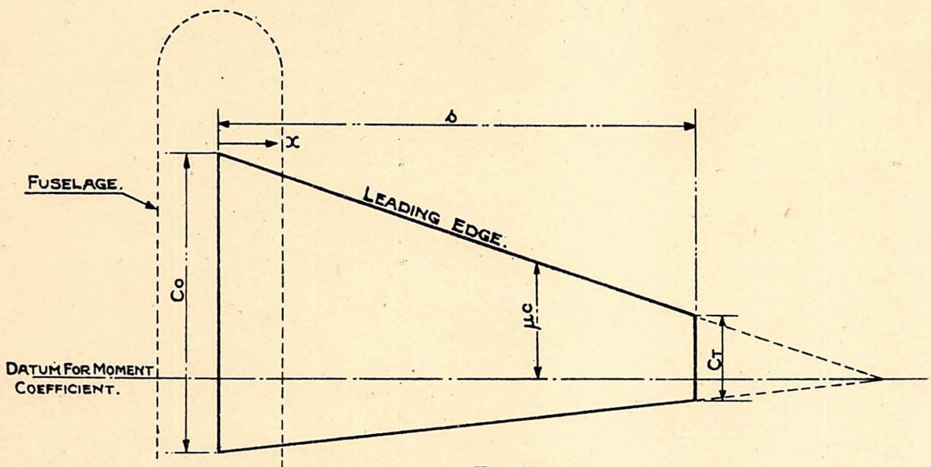


Fig. 2. Chap. VII



$$\lambda = \frac{\text{TIP CHORD}}{\text{ROOT CHORD}} = \frac{C_T}{C_0}$$

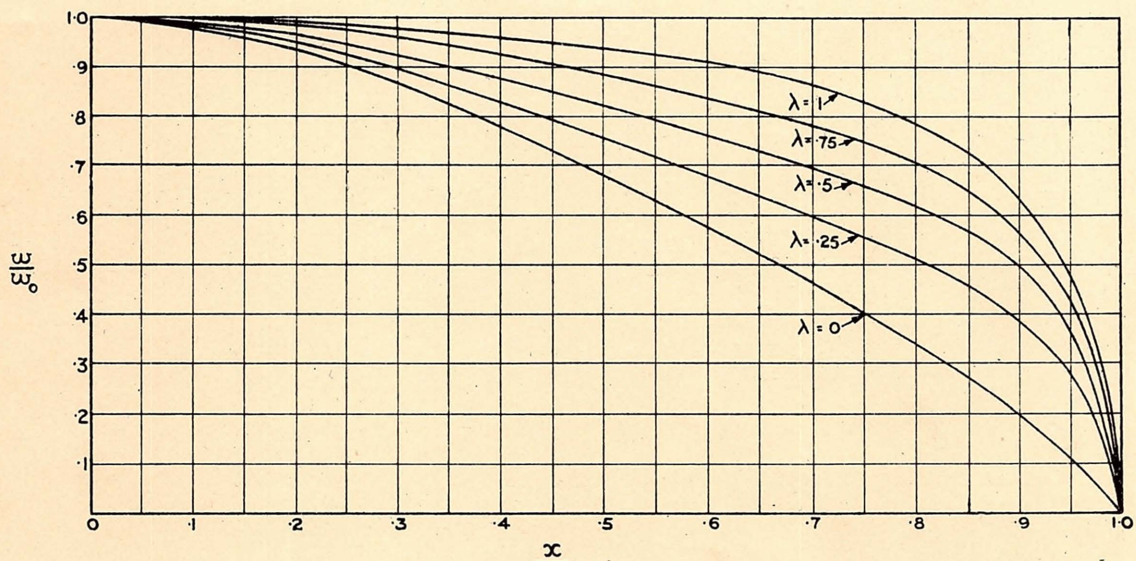


FIG. 3. Chap. VII.

LOAD DISTRIBUTION CURVES FOR CONSTANT x/s - [ASPECT RATIO 6 AND $\frac{dk_L}{d\alpha} = \pi$]

x = DISTANCE FROM WING ROOT

s = SEMI-SPAN.

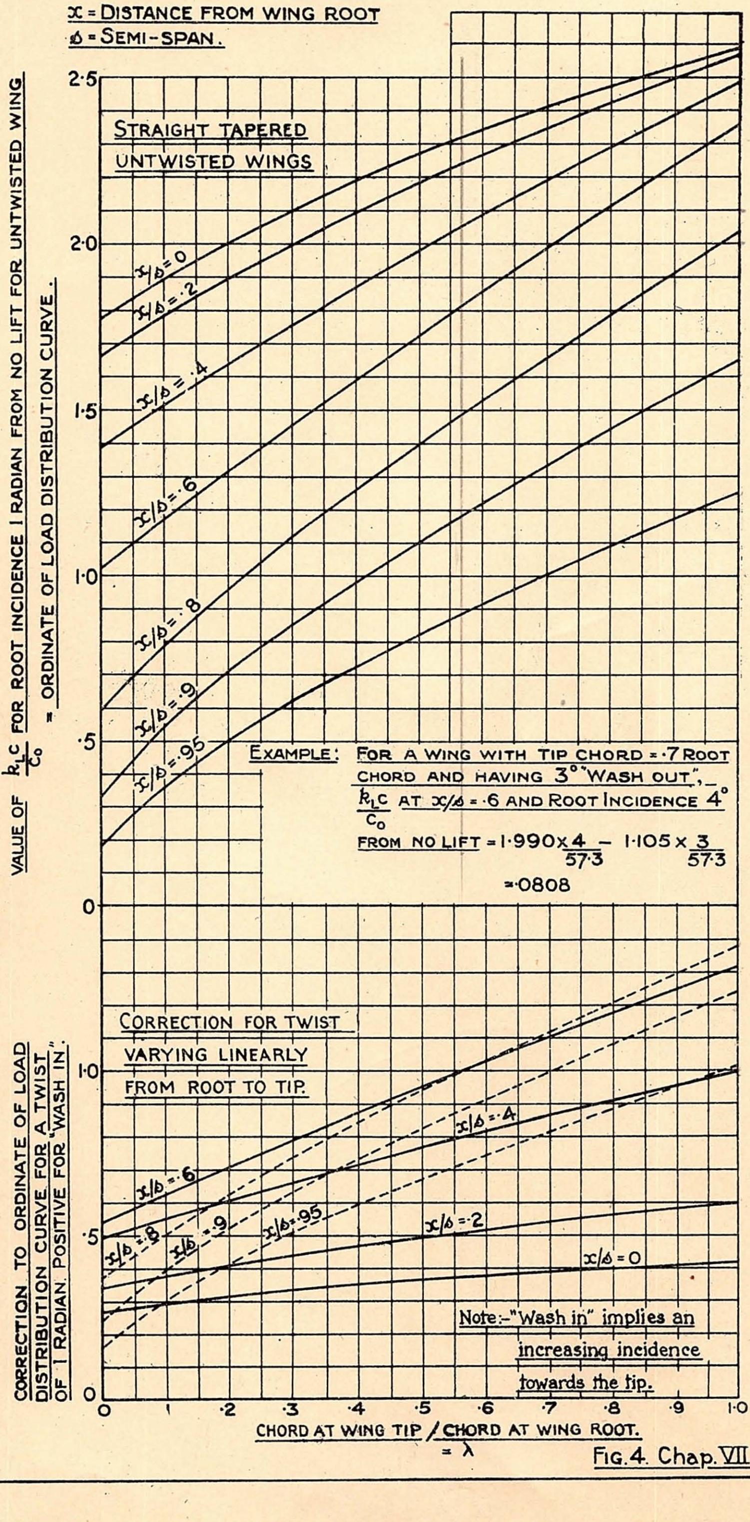
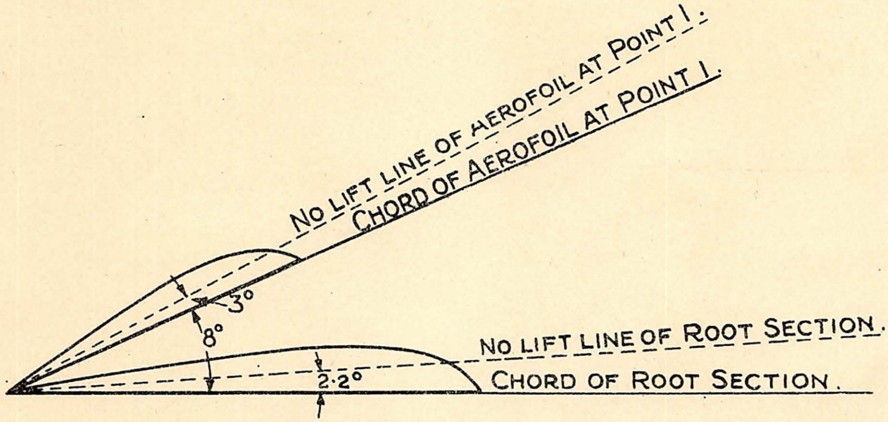


FIG. 4. CHAP. VII.

TWISTED WING.

DIAGRAM TO ILLUSTRATE METHOD OF DETERMINING THE INCIDENCE OF ANY SECTION FROM NO LIFT (α) WHEN THE ROOT SECTION IS AT NO LIFT ($\alpha_0 = 0$)



THE "NO LIFT LINE" GIVES THE DIRECTION OF THE WIND FOR ZERO LIFT.

	SECTION 1	2	3	4 ROOT.
ANGLE BETWEEN CHORD OF SECTION AND CHORD OF CENTRE SECTION. POSITIVE WHEN ROOT CHORD IS AT LESS INCIDENCE THAN THE CHORD OF THE SECTION CONCERNED.	$+8°$			0
NO LIFT ANGLE OF SECTION.	$-3°$			$-2.2°$
INCIDENCE OF SECTION (MEASURED FROM ITS OWN NO LIFT LINE) WHEN THE ROOT SECTION IS AT NO LIFT; I.E. ANGLE BETWEEN THE NO LIFT LINES OF THE SECTION CONCERNED AND THE ROOT SECTION.	$8° + 3° - 2.2° = 8.8°$			0
IN RADIANS.	$\alpha_1 = \cdot 1535$	$\alpha_2 =$	$\alpha_3 =$	$\alpha_4 = \alpha_0 = 0$

FIG. 5. Chap. VII.

Fig. 3 only gives the shapes of the curves. The curves of figs. 3 and 4 are calculated for a wing of aspect ratio of 6 and $\frac{dk_L}{d\alpha} = \pi$. The effect of small changes in these quantities may be ignored.

The load grading curve for a rectangular wing ($\lambda = 1$) should only be used for monoplanes. For a rectangular wing biplane the load distribution is assumed to be uniform from the centre line of the aircraft to a point distant $1 \cdot 2c$ from the wing tip, the loading falling away at the wing tip according to the curve given in chapter V, section I, fig. 1.

(iv) *Straight tapered twisted wings in which the twist varies linearly along the span.*—It may sometimes happen that the twist of a straight tapered wing at any point along the span is a linear function of the distance of that point from the wing root. A tabular method for determining the aerodynamic twist at points along the wing is given in fig. 5, and linear distribution of twist is indicated by α_1, α_2, α_3, and α_4 being proportional to the distances of sections 1, 2, 3 and 4 from the wing root. The load distribution curve for such a wing can be obtained directly from fig. 4, the curves on the lower portion of the diagram giving the correction to be applied to the ordinates of the loading curve of an untwisted wing given in the upper part of the diagram. This correction must be multiplied by the twist appropriate to the wing under consideration, as shown in the example given on the figure. It will be seen at once from this construction that the shape of the curve varies with incidence, since a constant quantity, representing the correction due to twist, is added to a quantity which is a function of the incidence.

(v) *General case of a curved tapered untwisted wing.*—The load distribution curve for this type of wing should be calculated as explained below. The possible variations in plan form are too large for it to be feasible to give generalised curves. The method is as follows :—

(*a*) Choose four points along the span of the wing such that the characteristics of the aerofoils at these points are known. An arithmetical simplification is obtained by taking the points given in table I.

TABLE I

Point.	$\frac{x}{s}$	chord, c.	a (*see* para. 2).	$\frac{ac}{4s} = u$
4	0			$u_4 =$
3	·3827			$u_3 =$
2	·7071			$u_2 =$
1	·9239			$u_1 =$

and it will be assumed to begin with that the wing characteristics at these points are known. Methods are given in sub-para. (vii) for dealing with any four points, and in sub-para. (viii) for taking into account a greater number of points than four.

(*b*) Determine the characteristics of the aerofoils forming the wing at the points given in table I, and fill in the table.

(*c*) Insert the values of u calculated in table I into the following four simultaneous equations, taking α_4 (and hence α_1, α_2 and α_3) equal to 1 :—

$$(0 \cdot 3827 + u_1)\, A_1 + 2 \cdot 414\, (0 \cdot 3827 + 3u_1)\, A_3 + 2 \cdot 414\, (0 \cdot 3827 + 5u_1)\, A_5 + (0 \cdot 3827 + 7u_1)\, A_7 = u_1\, \alpha_1 \quad (1)$$

$$(0 \cdot 7071 + u_2)\, A_1 + (0 \cdot 7071 + 3u_2)\, A_3 - (0 \cdot 7071 + 5u_2)\, A_5 - (0 \cdot 7071 + 7u_2)\, A_7 = u_2\, \alpha_2 \quad (2)$$

$$(0 \cdot 9239 + u_3)\, A_1 - 0 \cdot 4142\, (0 \cdot 9239 + 3u_3)\, A_3 - 0 \cdot 4142\, (0 \cdot 9239 + 5u_3)\, A_5 + (0 \cdot 9239 + 7u_3)\, A_7 = u_3\, \alpha_3 \quad (3)$$

$$(1 + u_4) A_1 - (1 + 3u_4) A_3 + (1 + 5u_4) A_5 - (1 + 7u_4) A_7 = u_4 \alpha_4 \quad \text{(4)}$$

Solve these equations for A_1, A_3, A_5 and A_7. A convenient tabular method of solution is given in table V at the end of this chapter. Columns 6 and 7 in this table refer only to twisted wings. The load distribution curve over the span is given by the equation :—

$$\frac{k_L c}{4s} = A_1 \sin\theta + A_3 \sin 3\theta + A_5 \sin 5\theta + A_7 \sin 7\theta \quad \text{(5)}$$

and can be drawn out by giving a number of values to θ corresponding to various points along the span. Values of $\sin n\theta$ at seven points along the span are given below.

TABLE II

x/s	θ	$\sin\theta$	$\sin 3\theta$	$\sin 5\theta$	$\sin 7\theta$
0	90°	1·0	—1·0	1·0	—1·0
0·2	78° 28′	·9798	— ·8231	·5348	— ·161
0·4	66° 25′	·9164	— ·3307	— ·4624	·9663
0·6	53° 08′	·8001	·3518	— ·9971	·2068
0·8	36° 52′	·60	·9360	— ·0756	— ·9783
0·9	25° 51′	·4361	·9765	·7853	— ·017
0·95	18° 12′	·3123	·8151	·9998	·7945
1·0	0	0	0	0	0

(*d*) The load distribution curve corresponding to equation (5) can be converted into a curve of k_L by multiplying the ordinates by $\frac{4s}{c}$, c having the value appropriate to each individual point considered. The value so obtained will correspond to an angle of incidence of one radian from no lift. k_L is directly proportional to incidence, so that its value for any given angle is easily calculated. The centre of pressure position at each point along the span can then be calculated from the known relation between lift coefficient and centre of pressure as given by standard wind tunnel tests on rectangular aerofoils, and hence the spar loading curves can be obtained.

(*e*) The shape of the wing loading curve is the same for all angles of incidence, the scale alone varying. The mean lift coefficient is dependent upon A_1 only and is given by the expression

$$\bar{k}_L = \frac{2\pi s^2}{S} A_1 \quad \text{(6)}$$

The value of A_1 for a straight tapered untwisted wing can be obtained from fig. 6 by putting $\phi = 0$. This mean lift coefficient applies to the assumed wing which extends to the centre line of the aeroplane. The correction to $\bar{k}_L$ to allow for the centre (fuselage) portion of the wing is given in para. 3 (ix).

(vi) *General case of curved tapered twisted wing.*—The shape of the load distribution curve for a twisted wing varies with incidence and hence it is necessary to calculate values of A_1, A_3, A_5 and A_7 for more than one value of root incidence. The work is conveniently arranged as follows :—

(*a*) Using the same four standard points as for an untwisted wing, insert appropriate values in table III for the wing characteristics at each point and work out values of u and $u\,\alpha$, taking three different values of α_0.

VALUE OF A_1 FOR STRAIGHT TAPERED WING WITH LINEAR DISTRIBUTION OF TWIST.

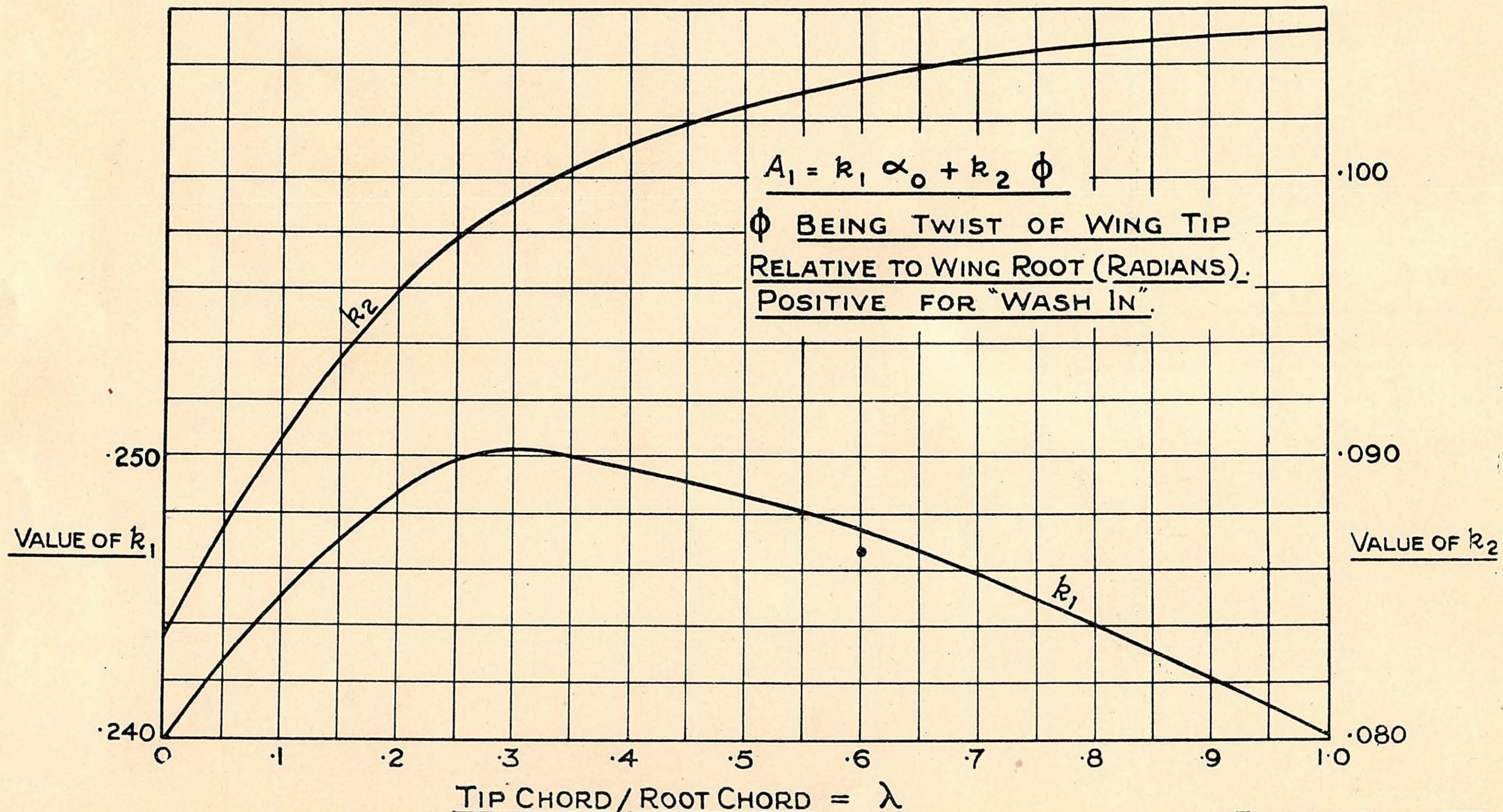


Fig. 6. Chap. VII

TABLE III

Point.	$\frac{x}{s}$	Incidence from no-lift in radians when $\alpha_0 = 0$	Chord c	a (*see* para. 2)	$\frac{ac}{4s} = u$	$u\alpha$ (α in radians) $\alpha_0 = 0$	$\alpha_0 = \cdot 5$	$\alpha_0 = 1 \cdot 0$
4	0	$\alpha_4 = \alpha_0 = 0$			$u_4 =$	$u_4\,\alpha_4 =$		
3	·3827	$\alpha_3 =$ } *See* Fig. 5			$u_3 =$	$u_3\,\alpha_3 =$		
2	·7071	$\alpha_2 =$ } *See* Fig. 5			$u_2 =$	$u_2\,\alpha_2 =$		
1	·9239	$\alpha_1 =$ } *See* Fig. 5			$u_1 =$	$u_1\,\alpha_1 =$		

(*b*) Insert these values of u and $u\alpha$ into equations (1) to (4). This will give three sets of four simultaneous equations, since there are three values of $u\,\alpha$ corresponding to three different root incidences at each point. These three sets of equations can be solved as shown in the numerical example given in para. 6 with little more labour than is involved in solving one set. The values of A_1, A_3, A_5 and A_7 should be linear functions of the root incidence, and hence the three values of α_0 taken above give a partial check upon the work.

(*c*) Express these constants in terms of α_0 :—

$$\left.\begin{aligned} A_1 &= h_1\,\alpha_0 + h_2 \\ A_3 &= h_3\,\alpha_0 + h_4 \\ A_5 &= h_5\,\alpha_0 + h_6 \\ A_7 &= h_7\,\alpha_0 + h_8 \end{aligned}\right\} \quad \ldots \quad (7)$$

α_0 being in radians.

(*d*) To draw the load distribution curves representing stalling, maximum speed, fast glide and terminal velocity dive conditions, calculate the mean lift coefficients (based upon the true wing area, *i.e.*, the $\bar{k}_L$ of para. 3 (ix)) corresponding to these speeds, and hence determine the root incidence for each case from the relation

$$\bar{k}_L = \frac{2\pi\,s^2}{S}\,(h_1\,\alpha_0 + h_2) \quad \ldots \quad (8)$$

The connection between the mean lift coefficient on the true wing area, $\bar{k}'_L$, and $\bar{k}_L$ is given in para. 3 (ix).

(*e*) Knowing α_0, the values of A_1, A_3, A_5, and A_7, are found from equation (7), and inserting these values in equation (5) gives the load distribution curve.

(vii) *Aerofoil characteristics not known at the four standard points.*—In the case of a wing the aerofoil section of which varies along the span, it may not always be convenient to take the four standard points given in tables I and III. Four other points may be taken if desired, at distances x_1, x_2, x_3 and x_4 from the wing root, where

$$\frac{x_1}{s} = \cos\,\theta_1$$

$$\frac{x_2}{s} = \cos\,\theta_2 \qquad \text{etc.}$$

The four equations (1) to (4) must then be replaced by the following :—

$$\sin\theta_1 (\sin\theta_1 + u_1) A_1 + \sin 3\theta_1 (\sin\theta_1 + 3u_1) A_3 + \sin 5\theta_1 (\sin\theta_1 + 5u_1) A_5 + \sin 7\theta_1 (\sin\theta_1 + 7u_1) A_7 = u_1 \alpha_1 \sin\theta_1 \quad \ldots \quad (9)$$

$$\sin\theta_2 (\sin\theta_2 + u_2) A_1 + \sin 3\theta_2 (\sin\theta_2 + 3u_2) A_3 + \sin 5\theta_2 (\sin\theta_2 + 5u_2) A_5 + \sin 7\theta_2 (\sin\theta_2 + 7u_2) A_7 = u_2 \alpha_2 \sin\theta_2 \quad \ldots \quad (10)$$

$$\sin\theta_3 (\sin\theta_3 + u_3) A_1 + \sin 3\theta_3 (\sin\theta_3 + 3u_3) A_3 + \sin 5\theta_3 (\sin\theta_3 + 5u_3) A_5 + \sin 7\theta_3 (\sin\theta_3 + 7u_3) A_7 = u_3 \alpha_3 \sin\theta_3 \quad \ldots \quad (11)$$

$$\sin\theta_4 (\sin\theta_4 + u_4) A_1 + \sin 3\theta_4 (\sin\theta_4 + 3u_4) A_3 + \sin 5\theta_4 (\sin\theta_4 + 5u_4) A_5 + \sin 7\theta_4 (\sin\theta_4 + 7u_4) A_7 = u_4 \alpha_4 \sin\theta_4 \quad \ldots \quad (12)$$

In choosing the positions of the four points it is advisable to arrange for them to be more closely grouped towards the tip of the wing, as the loading curve is varying more rapidly in this region.

(viii) *Aerofoil characteristics known at more than four points along the span.*—A method is described below for taking into account the aerofoil characteristics at more than four points along the span of the wing, without retaining more than four terms of the Fourier's series of equations (9) to (12).

If more terms in the series were to be retained no theoretical difficulty would arise; thus, if five points in the wing were considered, and the first five terms of the series retained, this would lead to five unknowns ($A_1 - A_9$) with five simultaneous equations for their determination. This method, however, soon becomes unmanageable, owing to the difficulty of solving a large number of simultaneous equations of this type with sufficient arithmetical accuracy. The labour involved is, in addition, usually unnecessary, as for most tapered wings the fifth and subsequent terms of the series have a negligible effect upon the shape of the load distribution curve. A type of wing in which these higher terms may be of importance is one in which the maximum chord is not at the wing root. In such a case sufficient terms should be calculated to ensure that subsequent terms may safely be ignored.

Dealing with a normal wing, in which the first four terms only need be considered, a method of taking into account more than four points on the wing is as follows. Suppose that ten points along the wing are taken, leading to ten equations of the type (9) to (12). These equations can be written in tabular form thus :—

A_1	A_3	A_5	A_7	
a_1	b_1	c_1	d_1	$= e_1$
a_2	b_2	c_2	d_2	$= e_2$
—	—	—	—	= —
—	—	—	—	= —
a_{10}	b_{10}	c_{10}	d_{10}	$= e_{10}$

Then by the method of least squares the best values of A_1, A_3, A_5 and A_7 are defined by the four simultaneous equations :—

A_1	A_3	A_5	A_7	
(aa)	(ab)	(ac)	(ad)	$= (ae)$
(ab)	(bb)	(bc)	(bd)	$= (be)$
(ac)	(bc)	(cc)	(cd)	$= (ce)$
(ad)	(bd)	(cd)	(dd)	$= (de)$

where the brackets indicate " the sum of all such terms," *i.e.*, $(ab) = a_1b_1 + a_2b_2 + a_3b_3 + \ldots\ldots + a_{10}b_{10}$. The majority of the summation terms in these equations are repeated once, thus reducing the labour of calculation.

This method can be generalised by replacing sums by integrals. Thus the summations (aa) (ab), etc., can be obtained by plotting the values of aa, ab, etc., along the span, and measuring the area enclosed by these curves by planimeter.

(ix) *Effect of fuselage on lift of wing.*—In the calculations described above the assumption has been made that the wing extends to the centre line of the aeroplane. In practice the centre part of the wing will be replaced by the body of the aeroplane, and hence the lift as calculated above will exceed the true lift by an amount equal to that taken by this non-existent central portion of the wing.

This discrepancy will not affect the shape of the load distribution curve, and hence in the case of untwisted wings no complication arises. That portion of the load distribution curve between the wing tip and the point at which the wing merges into the fuselage is all that is required for strength calculations, the scale being fixed by the consideration that the area under the load distribution curve must represent the known lift of the wing.

In the case of a twisted wing, however, where the load distribution curve varies with incidence, it is necessary to determine what root incidences will correspond to the various conditions of flight for which strength calculations are made. Hence the relation between $\bar{k}_L$ and $\bar{k}_L'$ is required, where $\bar{k}_L$ is as defined in para. 2 and $\bar{k}_L'$ is the mean lift coefficient over the actual wing (*i.e.*, as terminated by the fuselage) based upon the true wing area.

This relation is given approximately by the following expression :—

$$\bar{k}_L' = \bar{k}_L\left\{1 - \frac{S_1}{S - S_1}\left(\frac{4}{\pi} - 1\right)\right\} \quad \ldots \quad (13)$$

where S_1 is the amount by which the area of the assumed wing S exceeds the area of the actual wing. The approximations involved in the above formulæ are :—

(1) That for the assumed wing under consideration the ratio of the k_L on the centre line of the aeroplane to $\bar{k}_L$ is the same as for an elliptically tapered wing, and

(2) That on the assumed wing this centre k_L acts over the centre area S_1.

4. Drag coefficient.—(i) *Distribution of drag along the span.*—The wing drag coefficient at each point along the span is the sum of the induced drag, k_{D_1}, and the profile drag, k_{D_0}. The induced drag at the point $x/s = \cos\theta$ is given by the expression

$$\frac{k_{D_1}c}{4s} = \left\{\frac{k_L c}{4s}\right\}\frac{1}{\sin\theta}\left\{A_1 \sin\theta + 3A_3\sin 3\theta + 5A_5\sin 5\theta + 7A_7\sin 7\theta\right\} \quad \ldots \quad (14)$$

where $\frac{k_L c}{4s}$ has the value already calculated in equation (5).

(ii) Values of A_1, A_3, A_5 and A_7 for a straight tapered untwisted wing, corresponding to an angle of incidence of 1 radian from no-lift, are given below.

λ	A_1	A_3	A_5	A_7
0	·240	— ·051	·002	— ·005
·25	·250	— ·012	·010	— ·002
·50	·249	·007	·010	— ·001
·75	·245	·020	·008	0
1·0	·240	·029	·006	·001

(iii) For strength calculations it will usually be sufficiently accurate to take the profile drag coefficient for most aerofoils in common use as a constant, $k_{D_0} = \cdot 006$. Strictly speaking, k_{D_0} should vary at each section with the aerofoil and the angle of incidence of the section, but for strength calculations this variation may usually be ignored and the above mean value assumed.

(iv) The wing drag at any point along the span is given by $D = (k_{D_0} + k_{D_1})\, c \rho\, V^2$ lb. per unit length along the span, and hence a drag load distribution curve can be drawn out as has been done in the case of the lift load.

5. Moment coefficient.—(i) *General.*—Let

k_m = moment coefficient of any section referred to its leading edge.

$\bar{k}_m$ = mean moment coefficient of wing referred to the datum line (*see* below).

$$= \frac{M}{Sc_0\, \rho\, V^2}$$

μc = distance between the datum and the leading edge at any point, positive when the datum is behind the leading edge.

e and b are constants such that

$$k_m = ek_L + b$$

e and b can be taken from wind tunnel tests on wings of finite aspect ratio.

A datum line is chosen arbitrarily. For the general wing the perpendicular to the line of symmetry through the leading edge of the root section is most convenient. For straight tapered wings it is convenient to choose the perpendicular which passes through the intersection of the leading and trailing edges produced, as for this datum μ is constant.

The mean moment coefficient of a tapered wing is given (neglecting the contribution of k_D) by the definite integral.

$$\bar{k}_m = \frac{2}{Sc_0} \int_o^s (k_m + \mu k_L)\, c^2\, dx \qquad \text{(15)}$$

$$\text{or } \bar{k}_m = \frac{2}{Sc_0} \int_o^s \left\{ (e + \mu)\, k_L + b) \right\} c^2\, dx \qquad \text{(16)}$$

The evaluation of this integral for the various types of tapered wing is dealt with in the subsequent paragraphs.

As in the case of the lift coefficient, the moment coefficient $\bar{k}_m$ will refer to the assumed wing (*see* para. 2). The correct coefficient for the actual wing, $\bar{k}_{m'}$ is related to $\bar{k}_m$ by the formula :

$$\bar{k}'_m = \bar{k}_m \left(1 + \frac{S_1}{S - S_1} \right) - \left\{ b + (e + \mu) \frac{4}{\pi} \bar{k}_L \right\} \frac{S_1}{S - S_1} \qquad \text{(17)}$$

Here e and b have the values appropriate to the root section, *i.e.*, the section on the centre line of the aeroplane. This expression involves the same assumptions as already made in the case of the lift coefficient (*see* para. 3 (ix) (*d*)).

(ii) *Straight tapered untwisted wing.*—For this type of wing a datum is again chosen so that μ is constant (*see* sub-para. (i)). Curves are given in fig. 7 from which the moment coefficient for a straight tapered untwisted wing can be obtained directly. These curves are based upon the following assumptions :—

(1) The moment coefficient of the aerofoil forming the wing is a linear function of the lift coefficient.

(2) The aspect ratio, A, is 6. The aspect ratio appears in the expression for $\bar{k}_m$ given in fig. 7 only in order to define the geometry of the wing (in place of using the semi-span).

MOMENT COEFFICIENT OF UNTWISTED STRAIGHT TAPERED WINGS.

$$\bar{k}_m = [(e+\mu)(1-Af_1(\lambda))]\,\bar{k}_L + bf_2(\lambda)$$

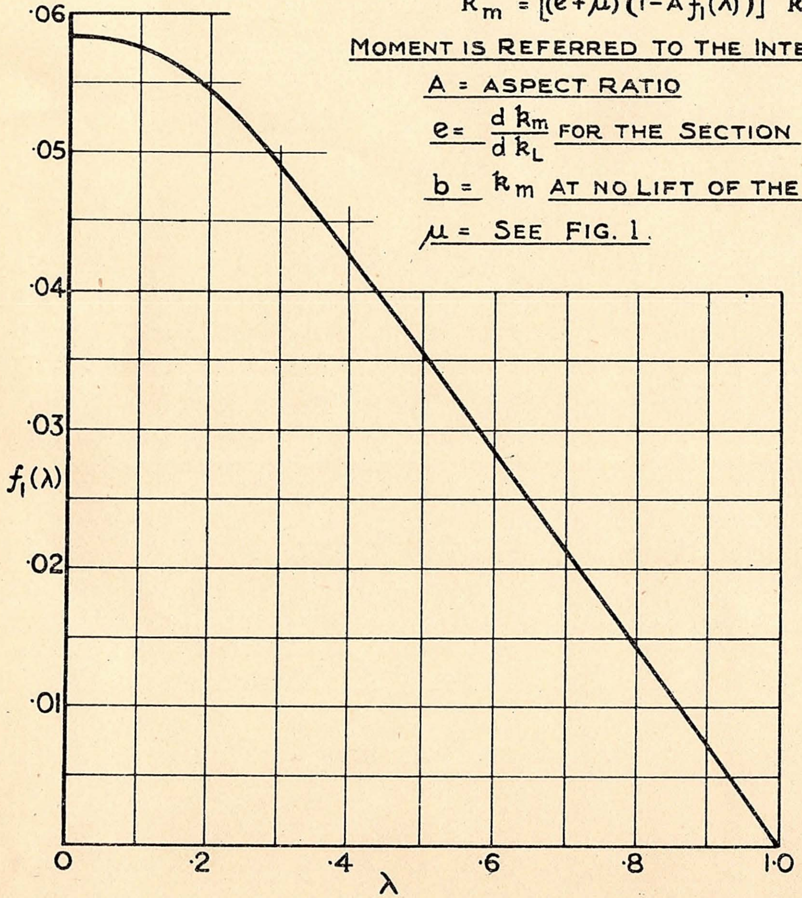
MOMENT IS REFERRED TO THE INTERSECTIONS OF THE LEADING AND TRAILING EDGES. (SEE FIG. 1.)

A = ASPECT RATIO

$e = \dfrac{d\,k_m}{d\,k_L}$ FOR THE SECTION

$b = k_m$ AT NO LIFT OF THE SECTION.

μ = SEE FIG. 1.



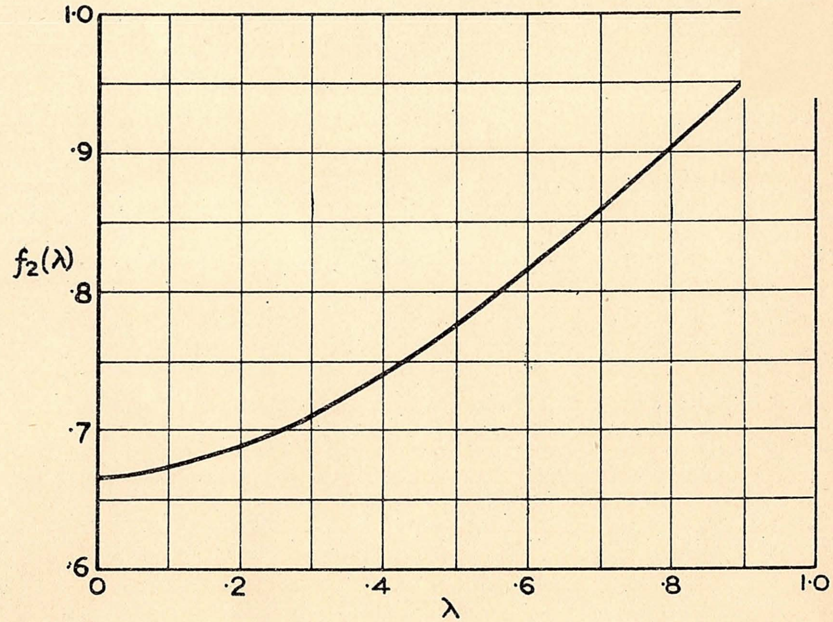


FIG. 7. Chap. VII.

Calculations on aspect ratios 6 and 8 have indicated that the shape of the load distribution curve is not greatly affected by small changes in aspect ratio, and hence the curves given will apply to the majority of wings met with in practice.

(3) The shape of the load distribution curves for different values of λ is as given in fig. 3.

These curves (fig. 7) give $\bar{k}_m$ in terms of $\bar{k}_L$, and hence, by equation (6), in terms of A_1. A_1 is a linear function of α_0, given for a straight tapered wing in fig. 6, and hence $\bar{k}_m$ can be expressed as a linear function of α_0 (α_0 in fig. 6 is in radians). In determining the tail load it must be remembered that $\bar{k}_m$ as calculated above is referred to the datum chosen and not to the leading edge, and that the chord c_0 is as defined in para. 2.

(iii) *Curved tapered untwisted wing.*—The loading curve obtained as described in para. 3 (v) is a curve of $\frac{k_L c}{4s}$ along the span corresponding to an incidence of one radian from no-lift. From this curve another curve representing $(e + \mu)\, k_L c^2$ can be constructed, and integrating this curve graphically will give the first term of equation (16). This value will correspond to $\alpha_0 = 57 \cdot 3^\circ$, and since this term is directly proportional to the incidence it can be expressed in terms of α_0 (in degrees) by dividing by $57 \cdot 3$.

The second term, $\int_0^s bc^2 dx$, can be integrated either graphically or analytically according to the shape of the wing, and hence the moment coefficient can be expressed in the form

$$\bar{k}_m = K_1\, \alpha_0 + K_2 \quad \dots \quad (18)$$

(iv) *Straight and curved tapered twisted wings.*—Inserting the values of A_1, A_3, A_5, A_7, given by equation (7) in equation (5), at the four standard points, will lead to the following equations :—

Point No. 1

$$\frac{k_L c}{4s} = \cdot 3827\, (h_1 + h_7)\, \alpha_0 + \cdot 9239\, (h_3 + h_5)\, \alpha_0 + \cdot 3827 (h_2 + h_8) + \cdot 9239\, (h_4 + h_6)$$

Point No. 2

$$\frac{k_L c}{4s} = \cdot 7071\, (h_1 + h_3 - h_5 - h_7)\, \alpha_0 + \cdot 7071\, (h_2 + h_4 - h_6 - h_8)$$

Point No. 3

$$\frac{k_L c}{4s} = \cdot 9239\, (h_1 + h_7)\, \alpha_0 - \cdot 3827\, (h_3 + h_5)\, \alpha_0 + \cdot 9239\, (h_2 + h_8) - \cdot 3827\, (h_4 + h_6)$$

Point No. 4

$$\frac{k_L c}{4s} = (h_1 - h_3 + h_5 - h_7)\, \alpha_0 + (h_2 - h_4 + h_6 - h_8) \quad \dots \quad (19)$$

$k_L c$ is now represented at four points along the span as the sum of two curves, one in terms of α_0 and one independent of α. Hence $(e + \mu)\, k_L c^2$ will be represented by two similar curves, the areas of which will give $\int_0^s (e + \mu)\, k_L c^2 dx$ in the form of equation (18). To this integral must be added the term $\int_0^s bc^2 dx$ which is independent of α_0 and contributes to the constant term K_2. The final expression for $\bar{k}_m$ will be in the form of equation (18).

6. Numerical example.—(i) A typical numerical example is given below to illustrate a convenient tabular method of solution and to indicate the magnitude of the constants involved. In the case of a wing with maximum chord not at the root the terms A_3, A_5 and A_7, will probably be larger in comparison with A_1 than in the case given. The example has not been carried further than the determination of A_1, A_3, A_5, and A_7, the work from this point onwards being quite straightforward. The particulars of the twisted wing dealt with are given in table IV (in the form of table III, para. 3 (vi)).

TABLE IV

Point.	$\frac{x}{s}$	Incidence from no-lift in radians when $\alpha_0 = 0$	Chord c	a (*see* para 2).	$\frac{ac}{4s} = u$	$u\alpha$ (α in radians) $\alpha_0 = 0$	$\alpha_0 = \cdot 5$	$\alpha_0 = 1\cdot 0$
4	0	$\alpha_4 = \alpha_0 = 0$	10 ft.	3·142	·2990	$u_4\,\alpha_4 =$ 0	·1495	·2990
3	·3827	$\alpha_3 = -\cdot 0334$	9·043	3·142	·2702	$u_3\,\alpha_3 = -\cdot 009025$	·12607	·26117
2	·7071	$\alpha_2 = -\cdot 0617$	8·233	3·142	·2460	$u_2\,\alpha_2 = -\cdot 01518$	·10782	·23082
1	·9239	$\alpha_1 = -\cdot 0806$	7·690	3·142	·2300	$u_1\,\alpha_1 = -\cdot 01854$	·09646	·21146

(ii) Putting the above values of u and $u\alpha$ into equations (1) to (4)gives the following three sets of four simultaneous equations :—

$$\cdot 6127\,A_1 + 2\cdot 5900\,A_3 + 3\cdot 7010\,A_5 + 1\cdot 9927\,A_7 = -\cdot 01854 \quad \text{I.}$$
$$\cdot 9531\,A_1 + 1\cdot 4451\,A_3 - 1\cdot 9371\,A_5 - 2\cdot 4291\,A_7 = -\cdot 01518 \quad \text{II.}$$
$$1\cdot 1941\,A_1 - \cdot 7185\,A_3 - \cdot 9425\,A_5 + 2\cdot 8153\,A_7 = -\cdot 009025 \quad \text{III.}$$
$$1\cdot 2990\,A_1 - 1\cdot 8970\,A_3 + 2\cdot 4950\,A_5 - 3\cdot 0930\,A_7 = 0 \quad \text{IV.}$$

$$\cdot 6127\,A_1 + 2\cdot 5900\,A_3 + 3\cdot 7010\,A_5 + 1\cdot 9927\,A_7 = \cdot 09646 \quad \text{V.}$$
$$\cdot 9531\,A_1 + 1\cdot 4451\,A_3 - 1\cdot 9371\,A_5 - 2\cdot 4291\,A_7 = \cdot 10782 \quad \text{VI.}$$
$$1\cdot 1941\,A_1 - \cdot 7185\,A_3 - \cdot 9425\,A_5 + 2\cdot 8153\,A_7 = \cdot 12607 \quad \text{VII.}$$
$$1\cdot 2990\,A_1 - 1\cdot 8970\,A_3 + 2\cdot 4950\,A_5 - 3\cdot 0930\,A_7 = \cdot 1495 \quad \text{VIII.}$$

$$\cdot 6127\,A_1 + 2\cdot 5900\,A_3 + 3\cdot 7010\,A_5 + 1\cdot 9927\,A_7 = \cdot 21146 \quad \text{IX.}$$
$$\cdot 9531\,A_1 + 1\cdot 4451\,A_3 - 1\cdot 9371\,A_5 - 2\cdot 4291\,A_7 = \cdot 23082 \quad \text{X.}$$
$$1\cdot 1941\,A_1 - \cdot 7185\,A_3 - \cdot 9425\,A_5 + 2\cdot 8153\,A_7 = \cdot 26117 \quad \text{XI.}$$
$$1\cdot 2990\,A_1 - 1\cdot 8970\,A_3 + 2\cdot 4950\,A_5 - 3\cdot 0930\,A_7 = \cdot 2990 \quad \text{XII.}$$

(iii) It will be seen that these three sets only differ from each other in the terms on the right-hand side. A convenient way of solving these three sets together is shown in table V.

To make the explanation of the table clear, neglect to begin with the figures printed in italics. The remaining figures in columns 2, 3, 4 and 5 are then the coefficients of A_1, A_3, A_5 and A_7 respectively, as given at the head of each column. Column 6 contains the terms on the right-hand side of equations I to IV ; column 7 the terms on the right-hand side of equations V to VIII ; and column 8 the terms on the right-hand side of equations IX to XII.

Equation I is written down at the head of the table (line 1) and the appropriate values from equations V and IX are written in columns 7 and 8. Equations II, III, and IV are written down in a similar way, several lines being left between each equation (equation II on line 23, III on line 42, and IV on line 55). Each equation is then divided through by the coefficient of A_1, giving equations IA, IIA, IIIA and IVA (lines 2, 24, 43 and 56). These are subtracted from one another as shown in column 1. In choosing which two equations to use together it is advisable to avoid equations which have the two coefficients of the same unknown nearly equal, as otherwise the solution may become arithmetically indeterminate.

The three equations given as a result of the subtractions are shown on lines 7, 29 and 48, and bringing the coefficients of A_3 to unity gives the three " B " equations, lines 8, 30 and 49.

This process is repeated, giving two " C " equations (lines 14 and 36) and one " D " equation (line 20), this latter equation giving the numerical value of A_7 when $\alpha_0 = 0$, $\cdot 5$ and $1\cdot 0$. It is advisable to check at this point that A_7 is a linear function of α_0.

The figures in italics refer to the calculation of A_5, A_3 and A_1.

Since A_7 is known, A_5 can be determined from equation Ic (line 14). Under the coefficient of A_7 in equation Ic ($-\ 1\cdot 170$) write its values when A_7 has the values corresponding to $\alpha_0 = 0$, $\cdot 5$ and $1\cdot 0$. A_5 is then determined by subtracting in turn the numerical values of A_7 from the terms in columns 6, 7 and 8. As a check upon the work determine A_5 in a similar way from equation IIc (line 36) and take the mean of the two values thus found for each of the three values of α_0.

Similarly three determinations of A_3 can be made from equations IB, IIB and IIIB (lines 8, 30 and 49) and four determinations of A_1 from equation IA, IIA, IIIA and IVA (lines 2, 24, 43, 56). It is advisable to calculate these in every case and to take the mean, as it gives a valuable check on the arithmetic and tends to reduce discrepancies due to small sliderule errors.

The mean of the values calculated in table V are as follows :—

	A_1	A_3	A_5	A_7
$\alpha_0 = 0$	$-\cdot 00914$	$-\cdot 00498$	$\cdot 000277$	$-\cdot 000536$
$\alpha_0 = \cdot 5$	$\cdot 1130$	$\cdot 00488$	$\cdot 000421$	$-\cdot 000487$
$\alpha_0 = 1\cdot 0$	$\cdot 2351$	$\cdot 01475$	$\cdot 00815$	$-\cdot 000442$

and hence equations (7) para. 3 (vi) (*c*) are as follows :—

$$A_1 = \cdot 2442\alpha_0 - \cdot 00914$$
$$A_3 = \cdot 01973\alpha_0 - \cdot 00498$$
$$A_5 = \cdot 007873\alpha_0 + \cdot 000277$$
$$A_7 = \cdot 000094\alpha_0 - \cdot 000536$$

(iv) In the above calculations A_7 has been evaluated first. It will usually be advisable to do this, as the probable error in A_1 is generally of the same order of magnitude as A_7, so that if the latter is made to depend upon the accuracy of the calculation of A_1 erratic results will probably be obtained.

TABLE V

	1	2	3	4	5	6	7	8
						Right Hand Side of Equation when $\alpha_0 =$		
		A_1	A_3	A_5	A_7	0	·5	1·0 radian.
1	I	·6127	2·590	3·701	1·9927	— ·01854	·09646	·21146
2	IA	1	4·2272	6·0405	3·2523	— ·03026	·15743	·34513
3	$\alpha_0 = 0$		— ·02092	·00167	— ·00174	$A_1 = -$ ·00926		
4	$\alpha_0 = \cdot 5$		·02061	·02545	— ·00158	$A_1 =$ ·11296		
5	$\alpha_0 = 1\cdot 0$		·06234	·04920	— ·00143	$A_1 =$ ·23502		
6	IVA	1	—1·4603	1·9207	—2·3816	0	·11509	·23018
7	IA—IVA		5·6875	4·1198	5·6339	— ·03026	·04234	·11495
8	IB		1	·7244	·9906	— ·00532	·00744	·02021
9	$\alpha_0 = 0$			·00016	— ·00053	$A_3 = -$ ·00495		
10	$\alpha_0 = \cdot 5$			·00305	— ·00048	$A_3 =$ ·00488		
11	$\alpha_0 = 1\cdot 0$			·00590	— ·00044	$A_3 =$ ·01475		
12	IIIB		1	1·4127	·1852	— ·00470	·01073	·02617
13	IIIB—IB			·6883	— ·8054	·00062	·00329	·00596
14	IC			1	—1·170	·000901	·00478	·00866
15	$\alpha_0 = 0$				·00062	$A_5 =$ ·000276		
16	$\alpha_0 = \cdot 5$				·00057	$A_5 =$ ·004213		
17	$\alpha_0 = 1\cdot 0$				·00051	$A_5 =$ ·008146		
18	IIC			1	1·2459	— ·000391	·003604	·00760
19	IIC—IC				2·4159	— ·001292	— ·001176	— ·00106
20	ID	Final Equation for A_7			1	— ·000536	— ·000487	— ·000442
21								
22								
23	II	·9531	1·4451	—1·9371	—2·4291	— ·01518	·10782	·23082
24	IIA	1	1·5162	—2·0324	—2·5486	— ·01589	·11314	·24218
25	$\alpha_0 = 0$		— ·00756	— ·00056	·00136	$A_1 = -$ ·00912		
26	$\alpha_0 = \cdot 5$		·00740	— ·00856	·00124	$A_1 =$ ·11304		
27	$\alpha_0 = 1\cdot 0$		·02235	— ·01656	·00113	$A_1 =$ ·23525		
28	IA	1	4·2272	6·0405	3·2523	— ·03026	·15743	·34513
29	IA—IIA		2·7110	8·0729	5·8009	— ·01437	·04429	·10295
30	IIB		1	2·9778	2·1398	— ·00531	·01638	·03806
31	$\alpha_0 = 0$			·00082	— ·00115	$A_3 = -$ ·00499		
32	$\alpha_0 = \cdot 5$			·01254	— ·00104	$A_3 =$ ·00488		
33	$\alpha_0 = 1\cdot 0$			·02426	— ·00094	$A_3 =$ ·01474		
34	IIIB		1	1·4127	·1852	— ·00470	·01074	·02617
35	IIB—IIIB			1·5651	1·9546	— ·00061	·00564	·01189
36	IIC			1	1·2459	— ·000391	·003604	·00760
37	$\alpha_0 = 0$				— ·00067	$A_5 =$ ·000277		
38	$\alpha_0 = \cdot 5$				— ·00061	$A_5 =$ ·004212		
39	$\alpha_0 = 1\cdot 0$				— ·00055	$A_5 =$ ·00815		
40								
41								
42	III	1·1941	— ·7185	— ·9425	2·8153	— ·009025	·12607	·26117
43	IIIA	1	— ·6017	— ·7893	2·3576	— ·00756	·10558	·21872
44	$\alpha_0 = 0$		·00300	— ·00022	— ·00126	$A_1 = -$ ·00908		
45	$\alpha_0 = \cdot 5$		— ·00293	— ·00332	— ·00115	$A_1 =$ ·11299		
46	$\alpha_0 = 1\cdot 0$		— ·00887	— ·00643	— ·00104	$A_1 =$ ·23507		
47	IA	1	4·2272	6·0405	3·2523	— ·03026	·15743	·34513
48	IA—IIIA		4·8289	6·8298	·8947	— ·02270	·05185	·12641
49	IIIB		1	1·4127	·1852	— ·00470	·01073	·02617
50	$\alpha_0 = 0$			·00039	— ·00010	$A_3 = -$ ·00499		
51	$\alpha_0 = \cdot 5$			·00595	— ·00009	$A_3 =$ ·00487		
52	$\alpha_0 = 1\cdot 0$			·01151	— ·00008	$A_3 =$ ·01475		
53								
54								
55	IV	1·2990	—1·8970	2·4950	—3·093	0	·1495	·2990
56	IVA	1	—1·4603	1·9207	—2·3816	0	·11509	·23018
57	$\alpha_0 = 0$		·00729	·00053	·00128	$A_1 = -$ ·00910		
58	$\alpha_0 = \cdot 5$		— ·00712	·00809	·00116	$A_1 =$ ·11295		
59	$\alpha_0 = 1\cdot 0$		— ·02153	·01565	·00105	$A_1 =$ ·23501		

CHAPTER VIII.—ESTIMATION OF STRENGTH OF INDIVIDUAL MEMBERS*

This chapter contains approved methods of estimating the strength of individual members, together with schedules of strengths and stresses. It is not possible to give a general ruling governing the circumstances in which appeal to an *ad hoc* strength test is (i) essential or (ii) permissible. The strength of some types of built up members cannot be calculated with confidence from the scheduled material properties. In such cases the strength should be determined from an *ad hoc* test made under the conditions described in chapter I. The strength of standardised members such as solid drawn tubular struts should be obtained from the formulæ and tables given in this chapter and not from *ad hoc* tests. The generalised information given in this chapter will often prove inadequate for calculating the strength of fittings, particularly thin plate fittings incorporating bolts, pins or rivets loaded in shear and bearing. Generalised test information from other sources is therefore admissible, provided it is supported by adequate and satisfactory tests and is not inconsistent with the data given in this chapter. Instances of serious inconsistency should be brought to the attention of the Airworthiness Department. Alternatively the strength of a fitting may be determined from an *ad hoc* destruction test on a specimen identical in all essential respects with the fitting concerned, the test being carried out and interpreted in accordance with chapter I.

Section I.—Tubular struts

1. Strength formulæ.—It can generally be assumed that compliance with the ultimate factor requirement will automatically ensure compliance with the proof factor requirement in the case of tubular struts.

The strength of axially loaded hard drawn tubular struts with diameter/thickness less than 80 is to be obtained from the following formula. The strength of struts with diameter/thickness greater than 80 should in general be determined from an *ad hoc* test.

$$p_2 = \frac{P}{A} + \frac{P\,e\,h \sec\alpha}{A\,k^2}$$

where p_2 = 0·2 per cent. proof stress of the material.

P = maximum load which the strut will withstand (*i.e.*, the crippling load).

e = equivalent eccentricity of end-load (*i.e.*, eccentricity due to manufacture).

$$\alpha = \frac{l}{2}\sqrt{\frac{P}{EI}}$$

A, I and k are respectively the area, the moment of inertia and minimum radius of gyration of the cross section.

h is the distance from the normal position of the neutral axis to the most highly stressed fibre.

l is the length of the strut.

This formula is applicable to any long strut of uniform cross-section of any elastic material. The formula gives reasonable values for short struts of uniform section and its use is therefore, general for all struts irrespective of length.

* For spars, *see* chapter VI.

2. Eccentricity.—The equivalent eccentricity, e, of end load given in table I below, is the sum of two terms :—

(*a*) The crookedness of the tube, *i.e.* the maximum deflection of any point on the centre line of the tube from the straight line joining the centres of areas of the end sections. The maximum allowable crookedness is usually given in the specification as $l/600$ or $l/300$, l being the length of the tube.

(*b*) A quantity representing the eccentricity of bore or neutral axis of the tube relative to the centre of the external diameter, expressed as a fraction of the nominal internal diameter.

If the end load is intentionally offset, the amount of this intentional offset should be added to e found as above.

In the case of unwelded seamless and cold-drawn circular steel tubes loaded in pure compression, two-thirds only of the equivalent eccentricity need be taken. This ruling does not apply to welded tubes.

In calculating the eccentricity of bore from the tolerances given in the specification, the following assumptions have been made :—

(*a*) The external and internal perimeters are truly circular.

(*b*) The mean thickness is the minimum permitted by the specification, *i.e.* the nominal thickness.

(*c*) The linear eccentricity (*i.e.* the distance between the centres of the external and internal perimeters) is such that at one point the tube has the minimum thickness permitted by the specification.

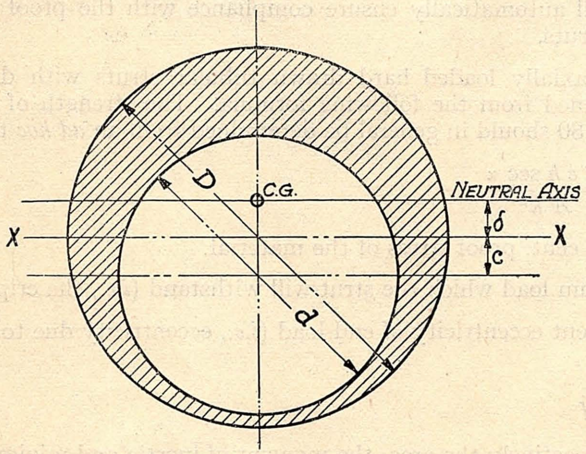


FIG. 1.—CHAP. VIII., Section I.

Referring to fig. 1

D and d are the external and internal diameters respectively.

c is the linear eccentricity.

δ is the eccentricity of bore, *i.e.* the eccentricity of the neutral axis relative to XX its nominal position.

Let t_0 be the minimum mean thickness permitted by the specification then $D - d = 2t_0$

Taking moments about XX, difference of areas of external and internal circles $\times\ \delta$

$$= \text{area of internal circle} \times c$$

A.L.2.

so that
$$\delta = \frac{\frac{\pi}{4} d^2 \times c}{\frac{\pi}{4}(D^2 - d^2)}$$
or
$$\delta = \frac{cd^2}{D^2 - d^2}$$
Now
$$D = d + 2t_0$$
hence
$$\delta = \frac{cd}{4t_0} \text{ approx.}$$

In the majority of relevant specifications the minimum thickness permitted is $\cdot 9t_0$. Thus
$$c = \cdot 1t_0.$$
With this value of c
$$\delta = \frac{d}{40}$$

d = internal diameter having regard to the tolerance allowed, but for all practical purposes eccentricity of bore = (nominal internal diameter)/40.

The equivalent eccentricity is $\frac{d}{40} + \frac{l}{600}$ for the following specifications :—

B.S.I.	D.T.D.	
2 T.1.	41	102
2 T.2.	~~89A~~	105
T.5.	91A	113
T.45	97	167

A.L.3

For B.S. Specification 3 T.4 for tubes of $\frac{3}{4}$ in. outside diameter or more, the eccentricity is as above. For tubes less than $\frac{3}{4}$ in. outside diameter, $e = \frac{d}{40} + \frac{l}{300}$.

3. Solution of strength formulæ.—The formula given in para. 1 may be written
$$p_2 = p\left\{1 + \lambda \sec \tfrac{1}{2}\frac{l}{k}\sqrt{\frac{p}{E}}\right\}$$
where
$$p = \frac{P}{A}$$
and
$$\lambda = \frac{eh}{k^2}$$
If
$$e = \frac{d}{40} + \frac{l}{600}$$
then it may be shown that
$$\lambda = \frac{d_m{}^2 - t^2}{10\,(d_m{}^2 + t^2)} + \frac{\sqrt{2}}{600}\,\frac{d_m + t}{\sqrt{(d_m{}^2 + t^2)}}\,\frac{l}{k}$$
where d_m = mean diameter
and t = thickness

Having regard to the dimensions of the tubes in practical use as struts, it can be demonstrated that with the above value of e the following faired value of λ
$$\lambda = \frac{1}{10} + \frac{1}{400}\,\frac{l}{k}$$
will be applicable to tubes of all diameters and gauges in practical use with a maximum error of about 2 per cent.

For
$$e = \frac{d}{40} + \frac{l}{300}$$
the appropriate faired value of λ will be
$$= \frac{1}{10} + \frac{1}{200}\,\frac{l}{k}$$
For an eccentricity $\frac{2}{3}e$ the corresponding values of λ will be $\frac{2}{3}\lambda$. With such values of λ, p becomes a function of $\frac{l}{k}$ only and hence strength curves may be compiled on a limiting stress-$\frac{l}{k}$ basis which are applicable to all tubes in practical use with a sufficient degree of accuracy. Such curves for various B.S.I. and D.T.D. tubes may be obtained on application to the Air Ministry.† In the absence of such curves the analytical solution of the formula can be avoided by the use of the diagram given in fig. 2.

Procedure in using the diagram (fig. 2).—(*a*) Calculate, from the known conditions of the strut, the following values :—

(1) q = Euler failing stress

$= \frac{\pi^2 E k^2}{l^2}$ where k = minimum radius of gyration of cross section of strut.

l = length of strut between pin centres.

E = Young's Modulus, taken from section VI of this chapter.

(2) $\beta = p_2/q$ where p_2 = 0·2 per cent. proof stress of the material, taken from section VI of this chapter.

(3) $\lambda = \frac{eh}{k^2}$ where h = distance from neutral axis of cross-section to the most highly stressed fibre.

e = equivalent eccentricity of end load taken from table below for tubes to B.S. and D.T.D. specifications. In the case of unwelded seamless and cold-drawn circular steel tubes loaded in pure compression, two-thirds only of the equivalent eccentricity need be taken. This ruling does not apply to welded tubes.

(*b*) Deduce the value β/λ and $\frac{\beta - 1}{\lambda}$.

(*c*) Draw a straight line across the diagram by joining appropriate points on the scales β, β/λ and $\frac{\beta - 1}{\lambda}$. Three points are known; the most convenient two may be used. Where the straight line cuts the curve read off the value of x.

(*d*) Then the crippling load P for the strut is given by Axq where A = area of cross-section of strut. To bring the diagram within a reasonable compass, two curves are given. Curve Y can be used for small values of β/λ and curve Z, which has a vertical scale $\frac{1}{10}$th that of Y, for larger values of β/λ.

A.L.3

Example.

Required to find the crippling load of a pin-ended unwelded strut $1\frac{1}{4}$ in. outside diameter, 20 gauge thick, 30 in. pin-centres to Specification ~~D.T.D. 80A~~. T.45

For this strut $\begin{cases} A = \cdot 1373 \text{ sq. ins.*} \\ k = \cdot 4295 \text{ ins.*} \\ l = 30 \text{ ins.} \end{cases}$ $l/k = 69 \cdot 8$, $d = 1 \cdot 178$ ins.

From section VI of this chapter

$$0 \cdot 2 \text{ per cent. proof stress} = 40 \text{ tons per sq. in.}$$

---

* These are nominal values which differ from the minimum values permitted by the specification by inappreciable amounts; this applies only to steel tubes.

† 737155/38

Hence $p_2 = 40$ tons per sq. in.

$E = 30 \times 10^6$ lb. per sq. in.

$$\therefore q = \frac{\pi^2 Ek^2}{l^2} = \frac{\pi^2 \times 30 \times 10^6}{(69 \cdot 8)^2} \text{lb. per sq. in.} = 60{,}800 \text{ lb. per sq. in.}$$

$$\text{and } \beta = p_2/q = \frac{40 \times 2{,}240}{60{,}800} = 1 \cdot 474$$

$$e = d/40 + l/600.$$

Only two-thirds of this eccentricity need be assumed as this is an unwelded seamless and cold-drawn circular steel tube loaded in pure compression.

$$\therefore e = \frac{2}{3}\left(\frac{1 \cdot 178}{40} + \frac{30}{600}\right) \text{ins.} = \cdot 053 \text{ ins.}$$

$$\therefore \lambda = \frac{eh}{k^2} = \frac{\cdot 053 \times \cdot 625}{(\cdot 4295)^2} = \cdot 179$$

$$\therefore \beta/\lambda = \frac{1 \cdot 474}{\cdot 179} = 8 \cdot 23 \text{ and } \frac{\beta - 1}{\lambda} = 2 \cdot 65$$

Hence, by drawing a straight line across the diagram joining the above points on the β/λ and $\frac{\beta - 1}{\lambda}$ scales, the value of $x = \cdot 768$.

$$\begin{aligned} \therefore \text{Crippling load} &= A\,x\,q \\ &= \cdot 1373 \times \cdot 768 \times 60{,}800 \text{ lb.} \\ &= 6{,}410 \text{ lb.} \end{aligned}$$

4. Note on thickness of 22 gauge tubes.—Only tubes of nominal thickness ·028 in. are to be referred to on drawings and in schedules as " 22 gauge ". Tubes of ·025 in. nominal thickness are to be labelled " ·025 in. thick " and are not to be called 22 gauge.

5. Note on use of non-corrosive steel tubes.—Tests carried out at the Royal Aircraft Establishment on a number of austenitic non-corrosive steel tubes show that this type of tube is likely to give low values of maximum compression stress. For those tubes which are " soft ", the ratio maximum compression strength to maximum tensile strength is of the order of ·5. For those tubes which are " hard " a higher ratio of the order of ·75–·85 is obtained. Further details relating to the tests are given in Royal Aircraft Establishment Report No. 06/9956.

Section II.—Tubes in bending, torsion and bearing

1. Bending.—It will usually be acceptable to calculate the ultimate strength in pure bending of members of circular section on a stress equal to the 0·1 per cent. proof stress increased by half the difference between the 0·1 per cent. proof stress and the ultimate stress. Compliance with the proof factor condition will then follow automatically.

Alternatively the strength of such members may be based upon an *ad hoc* test carried to the proof factor load only. If undamaged at this load it will usually be safe to assume, without continuing the test, that the ultimate factor condition will also be complied with.

The above is not applicable to tubes of diameter/thickness greater than 80 (eighty). For such tubes an *ad hoc* test will usually be required, carried to complete failure.

2. Torsion.—The available information on the strength of tubes in torsion is summarised in fig. 1. The strength needs to be calculated under both the proof factor condition and the ultimate factor condition, the allowable stresses for each of these conditions being shown in fig. 1.

When calculating the stress corresponding to the proof factor loads the following formula is to be used

$$\text{Shear stress} = \frac{16\,D}{\pi\,(D^4 - d^4)} \times \text{torque}$$

where D and d are the external and internal diameters. This assumes linear variation of stress with distance from neutral axis.

When calculating the stress corresponding to the ultimate factor loads the following formula is to be used

$$\text{Shear stress} = \frac{12}{\pi\,(D^3 - d^3)} \times \text{torque}$$

which assumes uniform stress throughout the thickness of the tube wall.

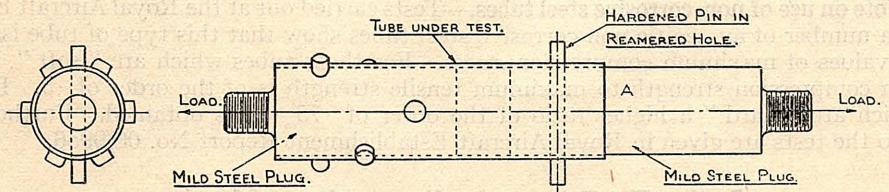
These two assumptions give the most consistent interpretation of test results, and have been adopted in preparing fig. 1.

The proof stress given in fig. 1 corresponds to a permanent set of ·001 radians on a length equal to the outside radius.

The dotted lines on fig. 1 indicate that the curves in these regions are tentative only, pending further tests.

3. Bearing.—The formulæ given below are derived from tests on the type of specimen shown in fig. 2. The plugs were made a sliding fit and the pins a push fit in the tubes. Steel and duralumin tubes 1 in. × 17 gauge and $1\frac{1}{2}$ in. × 22 gauge to Specification T.1, T.5, T.4 and D.T.D. Specification 89 were used in conjunction with high tensile steel pins $\frac{1}{4}$ in. and ·185 in. dia. (2.B.A.), the pins being in effect part of the test apparatus. This investigation and the formulæ given below deal only with the bearing strength of the tube wall. In any actual joint the strength of the pin or bolt in shear will also need consideration. (*See* section IV of this chapter.)

BEARING OF PINS IN TUBES.



TYPE OF TEST PIECE.

FIG. 2.—CHAP. VIII., Section II.

The above sizes of tubes and pins give a range of ratios thickness of tube/pin diameter from 0·1 to 0·3.

The allowable bearing pressure based on the above tests is as follows (for notation *see* below) :—

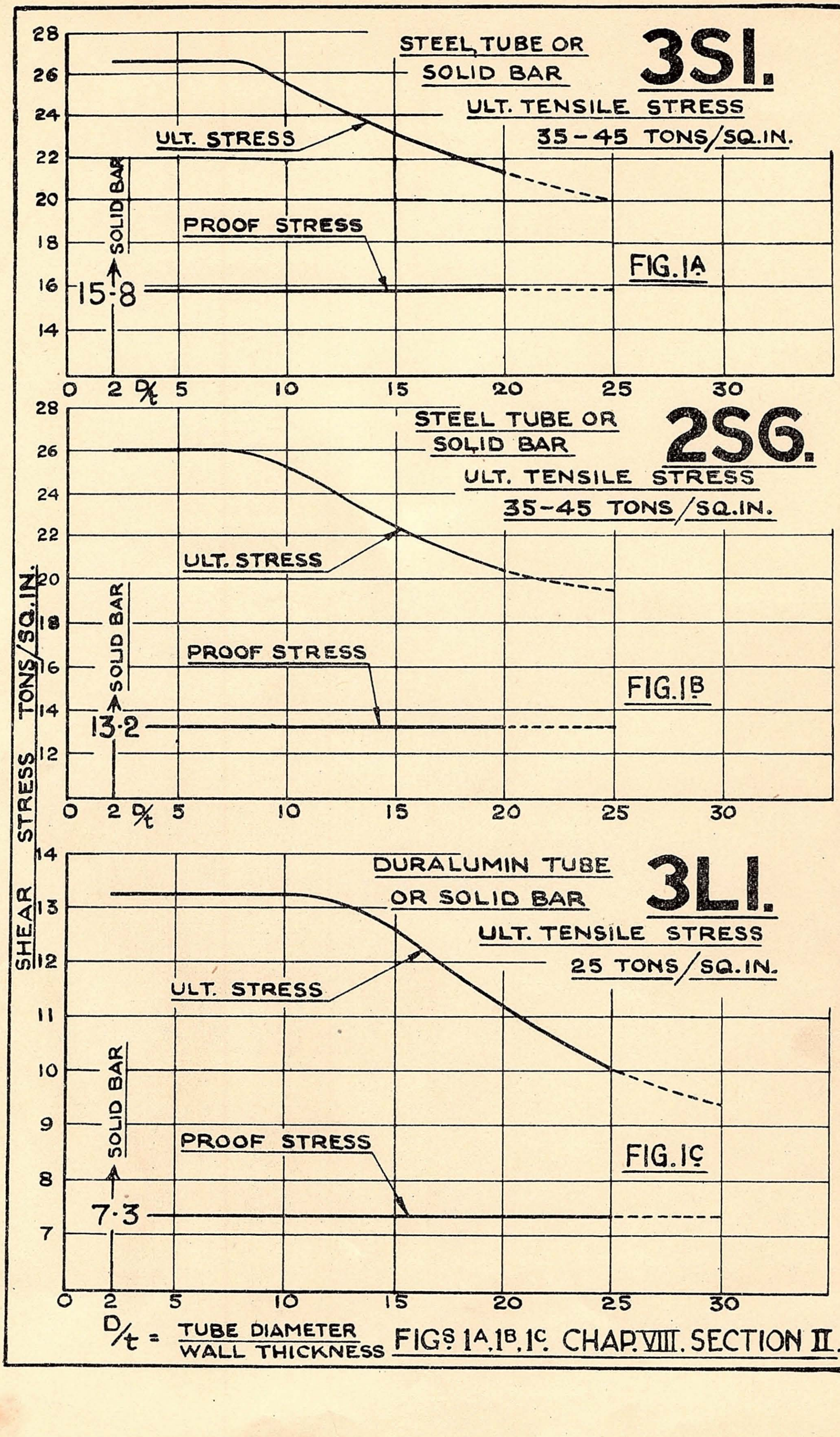
(*a*) For joints, not subject to serious vibration, where the design load is always applied in the same direction

$$f_b\ (\text{ult.}) = f_t\left(4{\cdot}45\frac{t}{d} + 1{\cdot}22\right) \text{ for steels.}$$

$$f_b\ (\text{proof}) = 1{\cdot}4\,f_t \text{ for steels.}$$

$$f_b\ (\text{ult.}) = f_t\left(1{\cdot}9\frac{t}{d} + 1{\cdot}63\right) \text{ for duralumin.}$$

$$f_b\ (\text{proof}) = 1{\cdot}2\,f_t \text{ for duralumin.}$$



FIGS. 1A, 1B, 1C. CHAP. VIII. SECTION II.

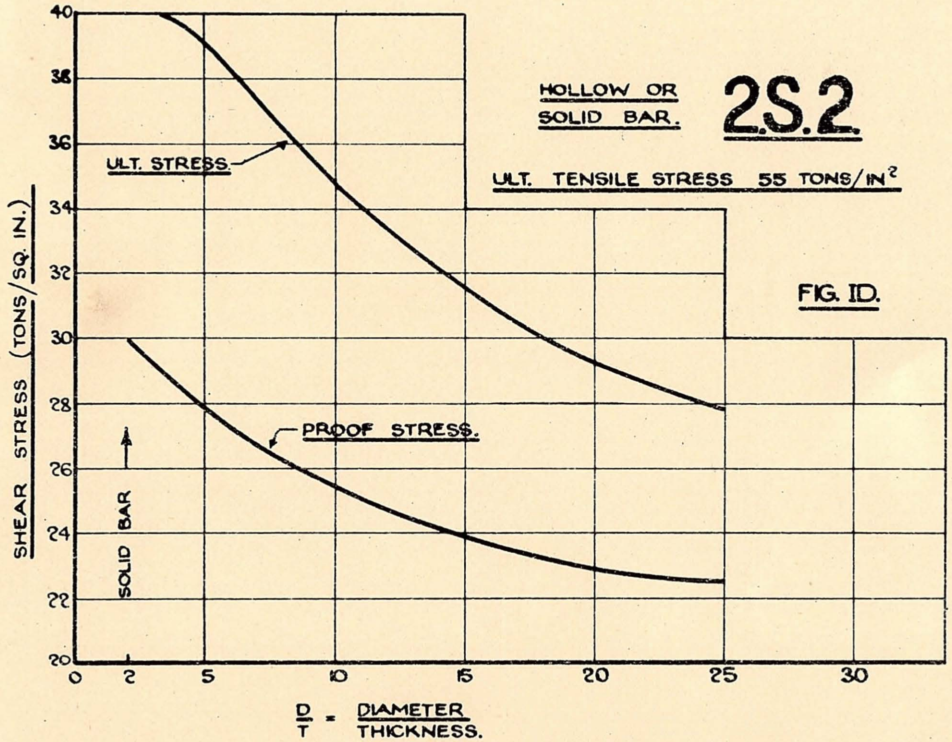
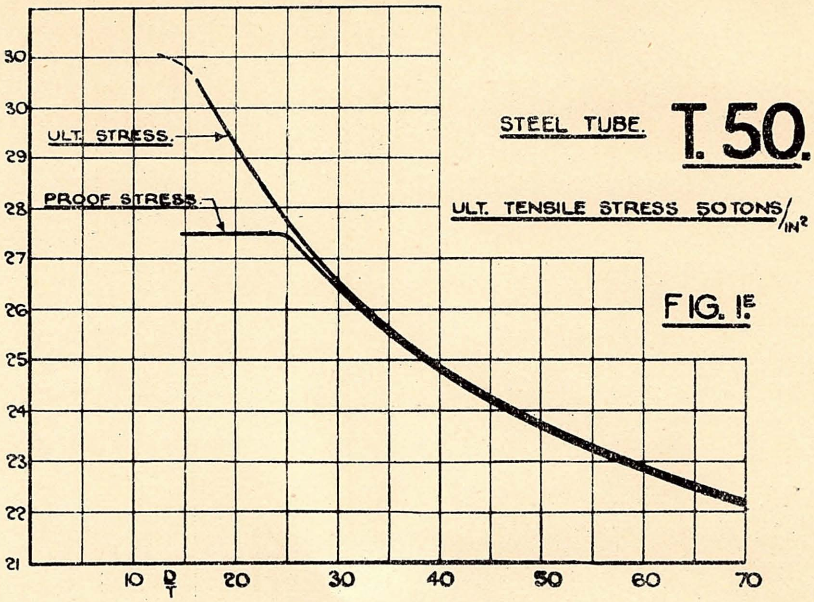
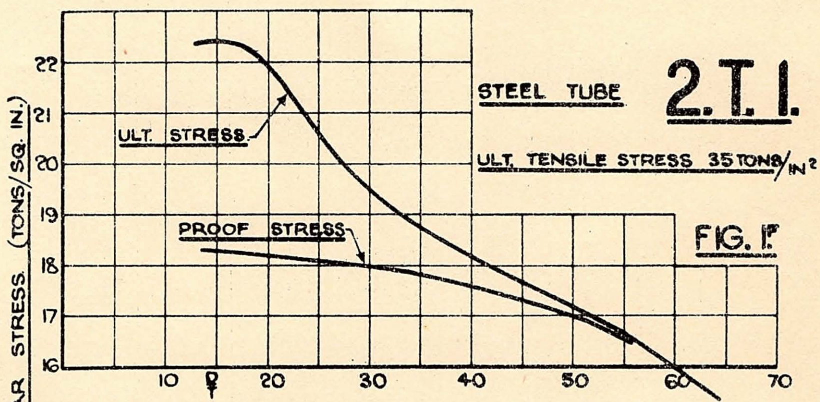
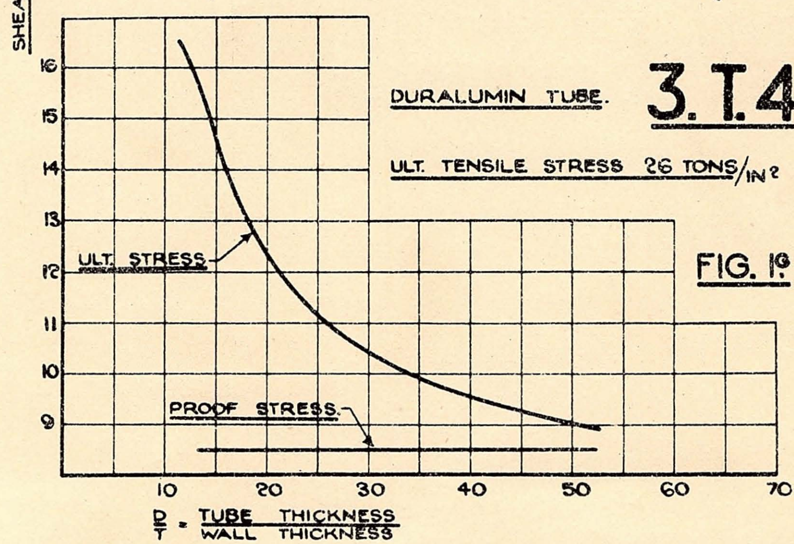


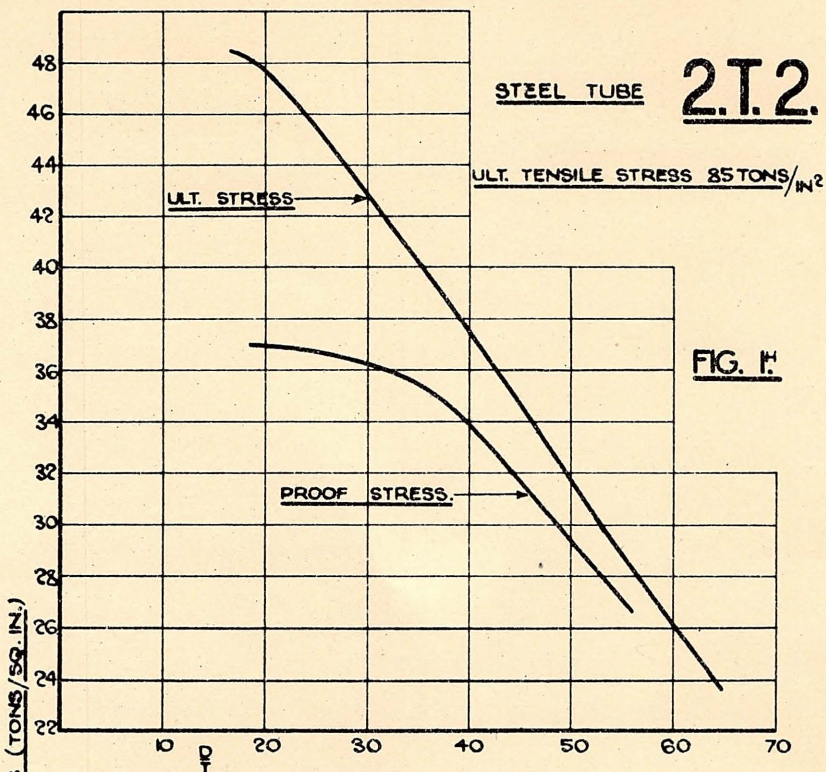
FIG. I^D CHAP. VIII SECTION II.

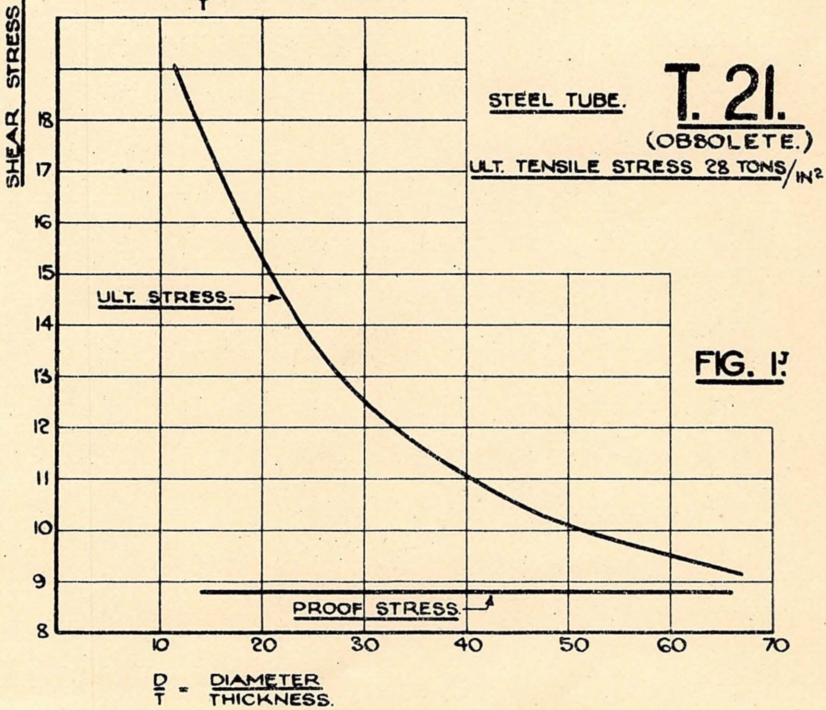






FIGS. 1E 1F & 1G. CHAP. VIII SECTION II.

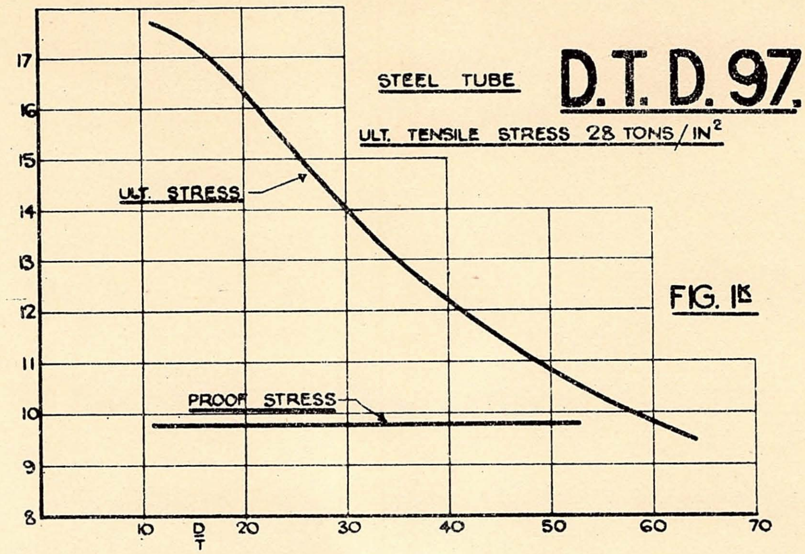


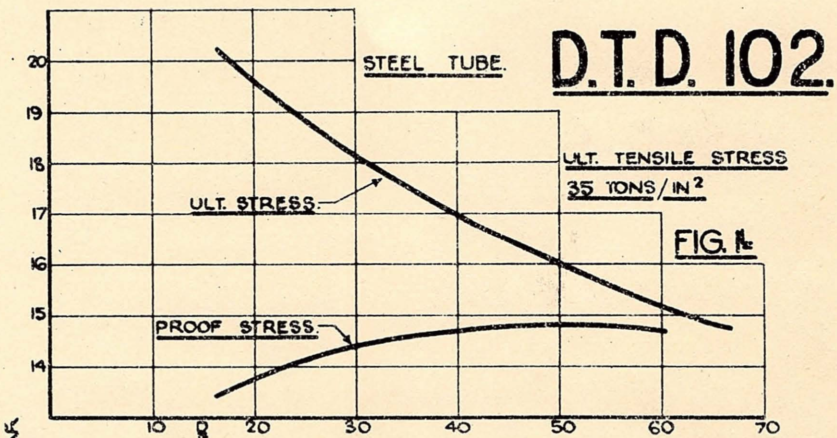


$\frac{D}{T} = \frac{\text{DIAMETER}}{\text{THICKNESS.}}$

FIGS. 1H & 1J CHAP. VIII. SECTION II.

Rpt (20T2)





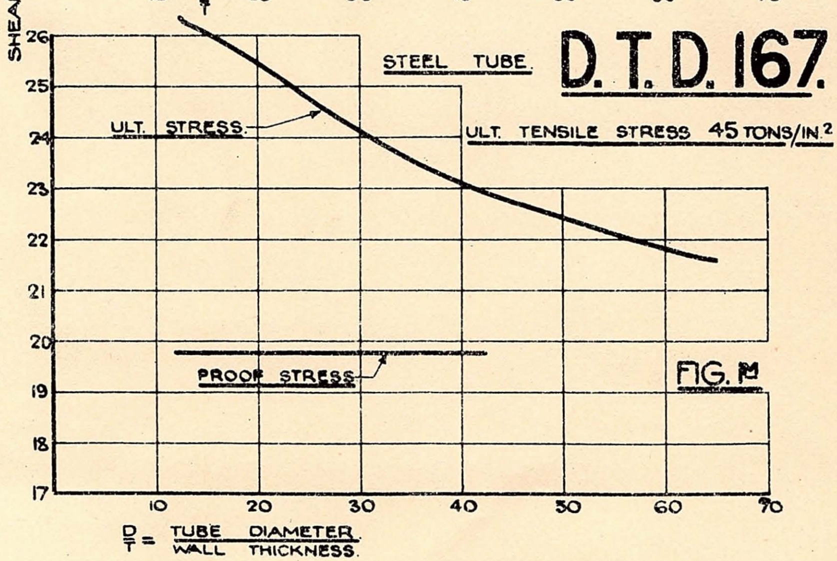
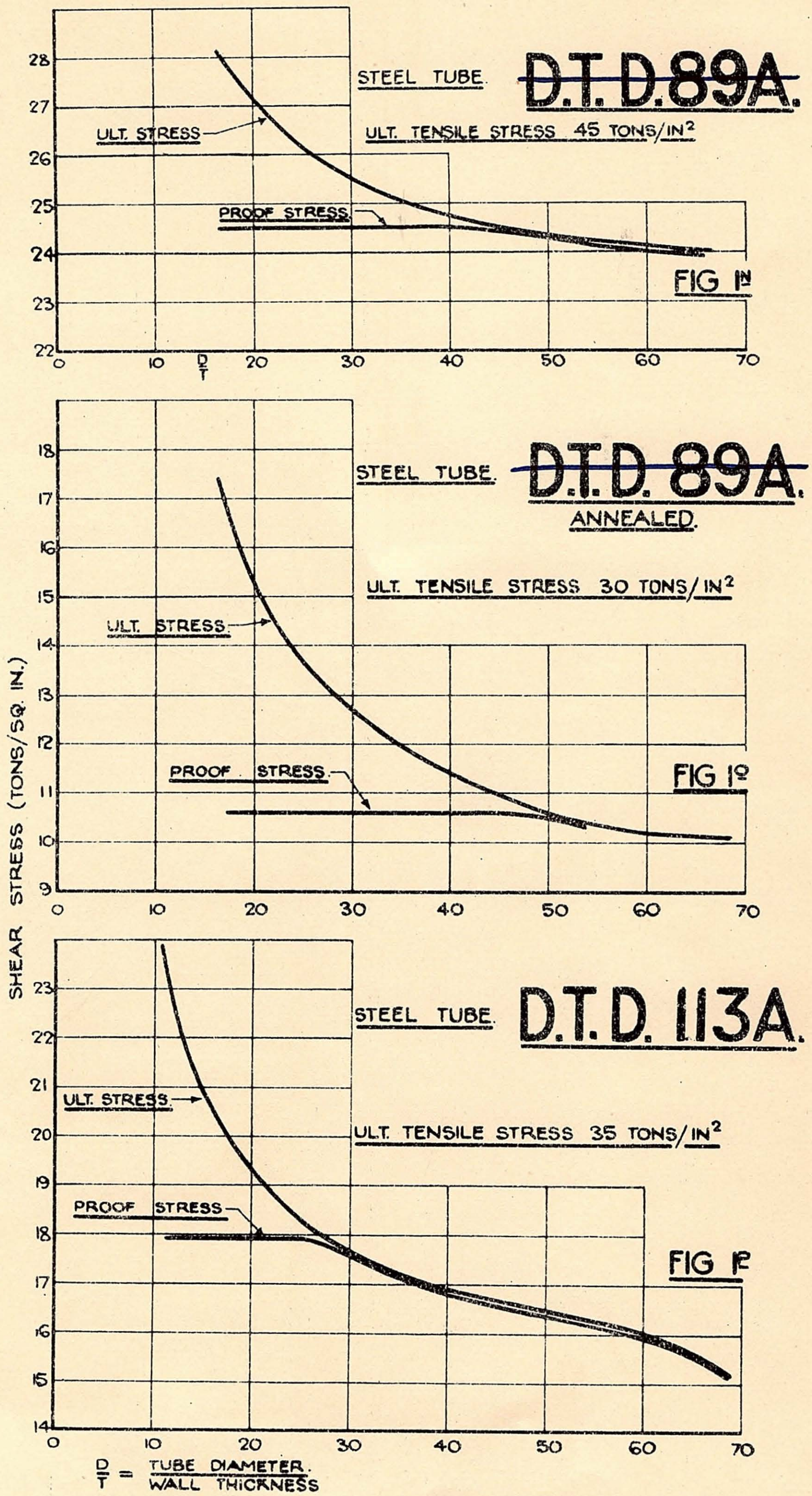


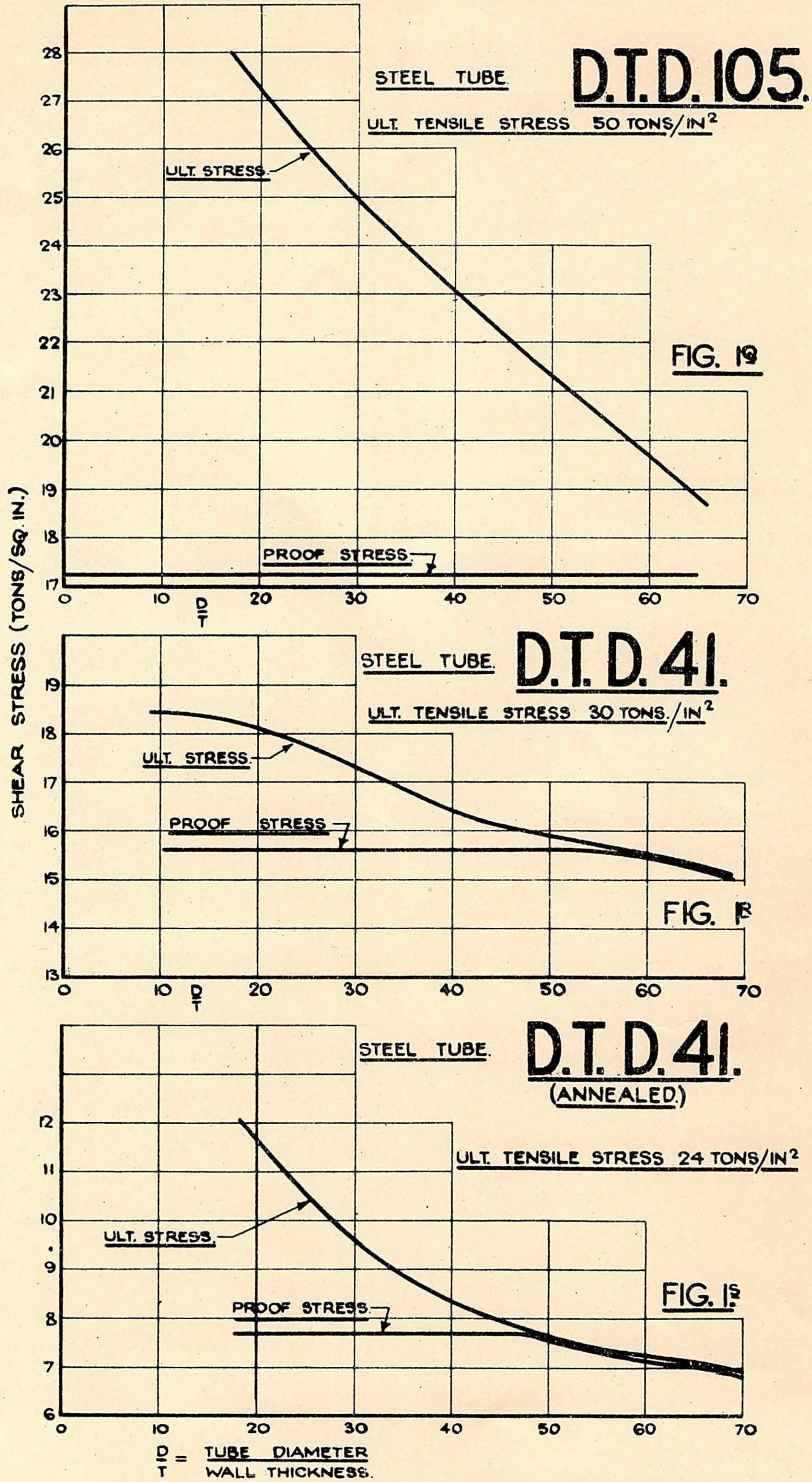
FIG. 1K 1L & 1M CHAP. VIII. SECTION II.

T.45 A.L.3

T.45 A.L.3



FIGS. 1n, 1o & 1p CHAP VIII SECTION II



FIGS. 1a, 1b & 1c CHAP. IX. SECTION II.

(*b*) For joints subject to serious vibration, and joints where the design load may be applied in either direction :—

f_b (ult.) as in (*a*) above for steels.

f_b (proof) $= f_t$ for steels.

f_b (ult.) as in (*a*) above for duralumin.

f_b (proof) $= \cdot 8\ f_t$ for duralumin.

In the above expressions f_b and f_t refer respectively to the allowable bearing pressure and the minimum specified ultimate tensile stress, t is the thickness of the tube walls and d the diameter of the pin or bolt.

In any joint of the type considered the bearing pressure corresponding to the ultimate factor load transmitted by the joint should not exceed f_b (ult.) and the bearing pressure corresponding to the proof factor load should not exceed f_b (proof). For most values of t/d the proof factor formula will be critical since the strength given by the ultimate factor formula usually exceeds $1\frac{1}{3}$ times that given by the proof factor formula.

The above data are directly applicable only to tube joints similar to that shown in fig. 2 with proportions of pin diameter to tube thickness within the specified range. It will, however, give some guidance in other cases until more precise information is available but must be used with increasing caution as conditions depart more and more from those of these tests.

The above figures refer to joints transmitting the load via a single pin or bolt. The information at present available indicates that when two or more bolts are used it is impossible to ensure that the load is uniformly distributed among them. If, therefore, it is necessary to use the preceding formulæ for calculating the strength of joints with two or more bolts it is necessary to make some allowance for this unequal distribution of load between them. The test evidence at present available indicates that the strength of a two-bolt joint under proof factor conditions should not be assumed to be more than 80 per cent. of the sum of the strength contributed by each bolt acting alone calculated from the preceding formulæ.

Test data on multi-bolt joints from other sources, as mentioned at the beginning of this chapter, may be used for calculating the strength of multi-bolt joints instead of calculating the strength from the preceding formulæ with the above 80 per cent. assumed distribution of load between bolts.

Under ultimate factor conditions the load can usually be assumed to be more uniformly distributed between the bolts. No general ruling, however, can yet be given.

Section III.—Torsional stresses—General formulæ

1. General.—The formulæ and methods of calculation described below give a rough guide to the maximum shear stress in any member corresponding to a given applied torque. Except in the case of metal members of circular section (*see* section II of this chapter) the maximum shear stress which the member can develop before failure will usually have to be obtained from an *ad hoc* test. For such members, therefore, the formulæ are of little practical value as the *ad hoc* test will give a direct measure of torsional strength and the corresponding stress need not be calculated. The formulæ may, however, be of interest as a rough preliminary guide. *See* also R. & M. 1393.

2. Pure torsion.

T = torque in lb. in.

q = shear stress in lb. per sq. in.

Type of Section.	Formulæ for Stress.	Position of Shear Stress.
Solid circle Diameter D	$q = \dfrac{16T}{\pi D^3}$	At boundary.
Hollow circle Outside diameter D Inside diameter d	$q = \dfrac{16TD}{\pi (D^4 - d^4)}$	At boundary.
If thickness t is small compared with the outside diameter D.	$q = \dfrac{2T}{\pi t D^2}$	At boundary.
Solid square Side S	$q = 4 \cdot 8 \dfrac{T}{S^3}$	At middle of sides.
Solid ellipse Major axis $2a$ Minor axis $2b$	$q = \dfrac{2T}{\pi ab^2}$	At end of minor axis.
	$q = \dfrac{2T}{\pi a^2 b}$	At end of major axis.
Hollow ellipse Outer major axis $2a$ Outer minor axis $2b$ Inner major axis $2a'$ Inner minor axis $2b'$ such that $a^2 - a'^2 = b^2 - b'^2$	$q = \dfrac{2T \dfrac{a^2 b}{a^2 + b^2}}{\pi \left\{ \dfrac{a^3 b^3}{a^2 + b^2} - \dfrac{a'^3 b'^3}{a'^2 + b'^2} \right\}}$	At end of minor axis.
	$q = \dfrac{2T \dfrac{ab^2}{a^2 + b^2}}{\pi \left\{ \dfrac{a^3 b^3}{a^2 + b^2} - \dfrac{a'^3 b'^3}{a'^2 + b'^2} \right\}}$	At end of major axis.
Solid rectangle Long side a Short side b	Approximately $q = \dfrac{T}{ab^2}\left(3 + 1 \cdot 8 \dfrac{b}{a}\right)$	At middle of long side.
Any hollow section Thickness t small compared with smallest outside dimensions. A is area bounded by mean perimeter.	Approximately $q = \dfrac{T}{2tA}$	Any point on boundary where thickness $= t$.

Any irregular solid section (*see* R. & M. 334, A.R.C. Report 1917/18, Vol. 3).—(i) Draw the largest possible inscribed circle and measure its radius $= a$ (fig. 1).

(ii) If there be a sharp projecting corner A, round it off by an arc of radius r, where r is found from the curve of r/a in R. & M. 334 for values of $\frac{\theta}{\pi}$.

Note.—r/a is given sufficiently well for values of $\frac{\theta}{\pi}$ from 0 to 0·8 by the formula :—

$$r/a = 1 - 0{\cdot}96\,\frac{\theta}{\pi}.$$

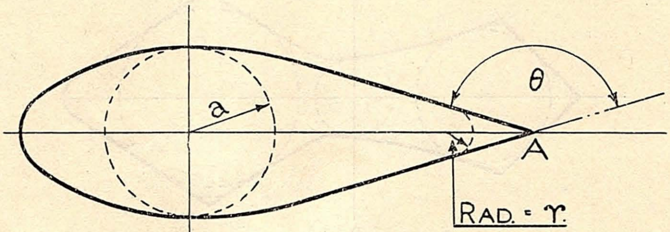


FIG. 1.—CHAP. VIII., Section III.

(iii) Let the area of the original section be A and its perimeter P : let the area and perimeter of the section be modified according to (ii) be A_1 and P_1. Let $h = \frac{2A}{P}$. Find K from the curve in the above R. & M. for values of a/h.

Note.—K is given sufficiently well for values of a/h from 0·5 to 1·0 by the formula :—

$$K = 3{\cdot}09\frac{a}{h} - 1{\cdot}614\left(\frac{a}{h}\right)^2 - 0{\cdot}476$$

Then find the quantity C given by :—

$$C = \frac{1}{2}\,KA\left(\frac{2A_1}{P_1}\right)^2$$

This formula gives C correct within 1 per cent. for sections such as triangles, ovals, etc., where only one value of a is possible. Other sections must be divided into component parts in the manner to be described below.

(iv) Having found C the shear stress can be estimated. The following remarks apply either to the whole of a simple section or to each component of a compound section, such as the spar section below. The maximum stress occurs at or near one of the points of contact of the largest inscribed circle which can be drawn in the section. The stresses at the three or more points of contact of the inscribed circle of maximum radius are given by :—

$$\frac{2aT}{(1 + m^2)\,C}\left\{1 + 0{\cdot}15\left(m^2 - \frac{a}{\rho}\right)\right\}$$

when $m = \frac{\pi a^2}{A}$ and ρ is the radius of curvature of the boundary at the point in question. (If the boundary is straight $\frac{a}{\rho} = 0$; if the boundary is concave $\frac{a}{\rho}$ is negative). The mean stress round the boundary of any component is given accurately by :—

$$\frac{2TA}{CP}$$

By means of these two formulæ the stresses all round the boundary can be estimated fairly accurately. The first of the two does not, strictly, apply to points where the boundary is concave, but it is found to agree with soap-film experiments for many sections, such as H, L and T sections, where the re-entrant angle is approximately a right angle, and is not very small.

In the division of compound solid sections there are really two classes to consider. In sections where there are two or more positions of the inscribed circle which give three or more points of contact, division is made by a straight line through the points of contact of the minimum inscribed circle which lies between two maximum inscribed circles.

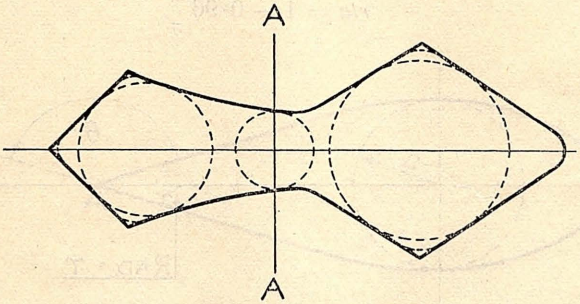


FIG. 2.—CHAP. VIII., Section III.

In fig. 2 the division line is AA. For sections such as H, T or L beams, the figure must be divided as follows :—In each of the sections, the points where the straight part of a boundary meets the curved parts are marked A. (*See* fig. 3.) The lines of division B are to be drawn at a distance from the commencement of the straight part, equal to half the thickness of the straight part. Each component can be treated separately (the spar and L section have three, the other has five components) ; each must have its corners rounded off as described above. The inscribed circles are arcs for rounding off the corners are shown dotted. The value of C can then be found for each component and the total C found by addition, but for each component, P must stand for only that part of the perimeter which forms part of the boundary of the whole section.

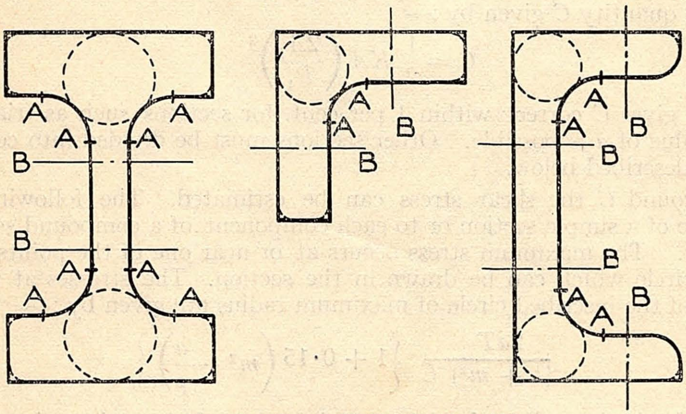


FIG. 3.—CHAP. VIII., Section III.

3. Torsion combined with other types of loading.—*Torsion combined with lateral load.*—In aeroplane structures torsion is usually accompanied by lateral load. The shear stress due to the lateral load occurs in a plane normal to the axis and in horizontal planes through the section (assuming the load vertical). Torsional shear stresses occur in a plane normal to the axis and in all planes through the axis of the section. Where a horizontal plane through the axis cuts the boundaries of the section the shear stresses due to twisting and bending will be algebraically additive. On one side they will add, on the other subtract. At the top and bottom of the section the shear stresses will be those due to torsion alone. Both cases may need consideration. In wooden beams the points on the side will probably be of the greater importance.

Combination of shear and direct stress.—On an element of material subject to a direct stress p and shear stress q, the following two formulæ giving the principal direct stresses and the maximum shear stresses are well known.

(*a*) Principal direct stress $= \frac{1}{2}p \pm \sqrt{(\frac{1}{4}p^2 + q^2)}$

(*b*) Maximum shear stress $= \sqrt{(\frac{1}{4}p^2 + q^2)}$

The shear stress may be due to torsion or lateral load or both, and the direct stress may be due to bending or axial load, or both. In beams subjected to combinations of these loads the section should be explored by means of the above formulæ.

N.B.—Owing to the fact that timber is neither homogeneous nor isotropic the above does not apply to timber members.

Section IV.—Shear and bearing of bolts, pins and rivets

1. Shear.—The strength in shear of bolts, pins and rivets depends on the associated bearing pressure, and decreases with increasing bearing pressure.

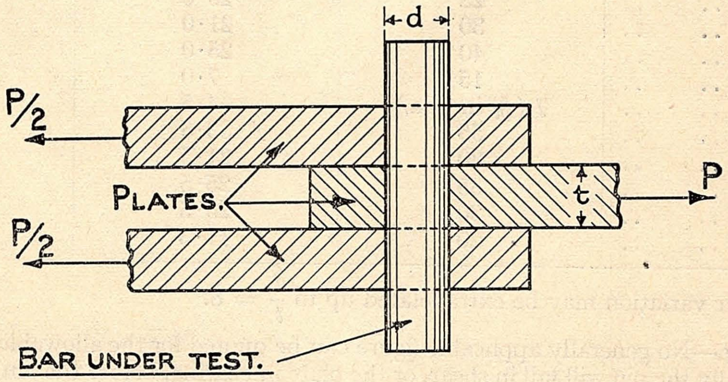
Until the bearing stress exceeds the shear stress this effect is very slight, but with high bearing stresses failure by shearing occurs at much lower shear stresses. The information available concerning the relationship between shear failure, shear stress, and bearing stress for pins in double shear can most conveniently be conveyed by recording allowable shear stresses against the geometrical proportions of the fitting under consideration.

If d = diameter of pin

and t = thickness of the centre plate of the three by which load is applied (*see* fig. 1)

then low d/t ratios will correspond to low bearing pressures and high ratios to high pressures.

Investigations have only been carried up to $\dfrac{d}{t} = 4$ at present.



SHEAR STRESS = P/2A WHERE A = CROSS-SECTIONAL AREA OF BAR.

NOTE:- PLATES TO BE HARD TO ENSURE THAT BAR FAILS FIRST.

FIG. 1.—CHAP. VIII, Section IV.

The allowable shear stress, s, is given in the following table. This shearing stress is to be associated with the specified ultimate factor. Compliance with the proof factor requirement may then be assumed.

Allowable Shear Stress for Pins in Double Shear

(To be associated with the specified ultimate factor)

Material.	Minimum specification ultimate tensile stress (tons/sq. in.).	Shear stress (tons/sq. in.). (Linear variation from $\frac{d}{t} = 1\cdot75$ to $4\cdot0$.) Up to $\frac{d}{t} = 1\cdot75$	At $\frac{d}{t} = 4\cdot0$
B.2	16	15·0	12·5*
B.13	25	12·8	9·8*
3L.1	25	12·5	10·0
L.32	10	6·5	5·2*
L.35	14	10·0	8·0*
S.1	35	23·0	17·0
S.2	55	35·5	26·5
S.6	35	23·5	18·0
S.11	55	35·0	26·0
S.21	25	17·5	13·0
S.61	35	24·0	17·5
S.62	46	30·0	22·0
S.71	25	17·1	12·8
S.76	40	27·0	19·5
S.77	30	21·0	14·0
S.80	55	34·0	25·0
D.T.D.24A (non-magnetic)	30	22·0	18·5
D.T.D.24A (magnetic) ..	30	21·0	15·0
D.T.D.53A	28	20·0	15·0
D.T.D.78A	30	21·0	15·5
D.T.D.126	40	25·0	18·5
D.T.D.142	15	7·0	5·2
D.T.D.148	7 ($\frac{5}{16}$ in. dia.)	4·5	2·7
D.T.D.153	55	33·5	24·0
D.T.D.161	30	21·5	16·0
D.T.D.176	35	25·5	21·0
D.T.D.189	30	23·5	18·5
D.T.D.198	20	11·3	8·5

* The linear variation may be extrapolated up to $\frac{d}{t} = 8$.

2. Bearing.—No generally applicable figure can be quoted for the allowable bearing pressure on a pin. Usually the pin will fail in shear, or the plate in bearing, before the pin itself is damaged in bearing. This does not hold, however, if a bolt is used and is loaded in bearing on the threaded portion. A very low bearing pressure will then damage the bolt, and hence bolts must be so fitted that their threaded portions are not loaded in bearing.

Section V.—Strength schedules of wires and tie rods and their end fastenings

1. General.—The strengths given in this section are for use in conjunction with the ultimate factor load, and compliance with the proof factor requirement may be assumed without further investigation.

2. Streamline wires to Specification 5.W.3 and swaged tie rods to Specification 5.W.8.

Table I

Size.	Strength (lb.).
4.B.A.	1,050
2.B.A.	1,900
$\frac{7}{32}$ in.	2,600
$\frac{1}{4}$ in.	3,450
$\frac{9}{32}$ in.	4,650
$\frac{5}{16}$ in.	5,700
$\frac{11}{32}$ in.	7,150
$\frac{3}{8}$ in.	8,500
$\frac{13}{32}$ in.	10,250
$\frac{7}{16}$ in.	11,800
$\frac{15}{32}$ in.	13,800
$\frac{1}{2}$ in.	15,500
$\frac{9}{16}$ in.	19,300
$\frac{5}{8}$ in.	23,630
$\frac{11}{16}$ in.	29,610
$\frac{3}{4}$ in.	34,520
$\frac{7}{8}$ in.	48,190
1 in.	63,850
$1\frac{1}{8}$ in.	81,270
$1\frac{1}{4}$ in.	103,500

Note.—The strength of both streamline wires and swaged tie rods is the same for corresponding sizes.

3. High tensile steel wires to Specification 3.W.1.

TABLE II

Gauge.	Strength (lb.).
8	3,603
9	2,919
10	2,450
11	2,131
12	1,713
13	1,339
14	1,070
15	866
16	721
17	552
18	405
19	282
20	229
21	179
22	137
23	102
24	84

Note.—The strengths tabulated above are for the wire only. The maximum load will, however, usually be limited by the type of end fastening. *See* para. 4.

4. Strength of terminal connections of solid wires.—The end connections of solid wires have usually less strength than the wires themselves. The weakest point may be in—

(*a*) the fastening of the loop,

(*b*) the loop of the wire,

(*c*) the lug or eye through which the wire is threaded.

(*a*) *The fastening of the loop.*—The standard method of securing a loop of wire is by a ferrule or close coil of about six turns of wire, as shown in fig. 1.

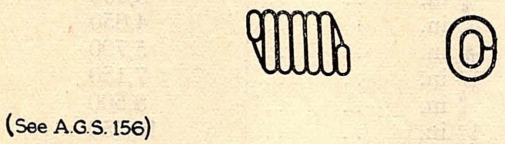


FIG. 1.—CHAP. VIII., Section V.

The tail of the loop is normally bent back over the ferrule, *see* fig. 2. It has been found, especially with the thinner wires, that unless special precautions are taken the bent portion opens out under load and the wire pulls through the ferrule. Because of this only 45 per cent. of the nominal strength of the wire can be relied upon for end fastenings such as fig. 2. If, however, the

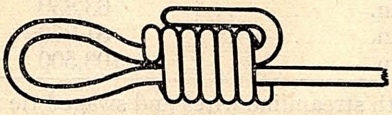


FIG. 2.—CHAP. VIII., Section V.

turned back end is bound to the ferrule with—for example—a soft iron wire, pulling through is prevented and some other type of failure supervenes. Sometimes sweating as well as binding is employed, but it is doubtful if this further precaution adds to the effective strength. If the turned back end is bound to the ferrule 60 per cent. of the nominal wire strength can be relied upon, failure then probably being in the loop.

(*b*) *Failure of loop.*—Tests show a considerable variation of strength of loops even between examples of the same design. Failure usually occurs in the neck at the entrance of the ferrule. Because of the variability of results it is unsafe to allow for the loop more than 60 per cent. of the nominal strength of the wire.

The form and diameter of the pin or roller round which the wire passes has little effect on the loop strength. Rollers used with turnbuckles (fig. 3) give practically the same failing load in the loop as when plain round pins are used instead of rollers.

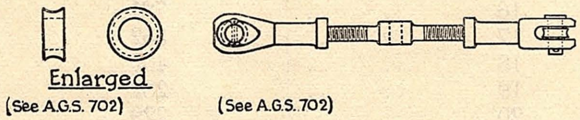


FIG. 3.—CHAP. VIII., Section V.

(*c*) *Lug or eye.*—No general rule can be made for strength of lugs loaded by a wire, but warning is given that the intense bearing stress under the wire may cause the wire to cut through the lug at loads much less than the nominal failing load of the eye. Special attention is drawn to the reduction of strength of standard barrel type turnbuckles (fig. 4) when used with solid high tensile steel wires.

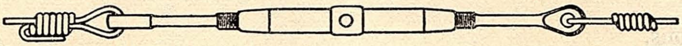


FIG. 4.—CHAP. VIII., Section V.

The following table gives allowable strength of the turnbuckles and wires in combination. The wires are assumed to be fastened with standard ferrules and the ends effectively turned back and tied. Strength is determined by failing load of loop or turnbuckle eye, whichever is the lower.

TABLE III

Turnbuckle.		Wire.		Strength of Combination.
A.G.S. No.	Nominal strength lb.	Gauge.	Nominal strength lb.	
490	560	18	405	243
		16	721	432
		14	1,070	530
491	1,120	16	721	432
		14	1,070	642
		12	1,713	1,028
		10	2,450	1,240
492	2,240	12	1,713	1,028
		10	2,450	1,470

5. Loop splices in straining cord and steel wire rope.—The efficiency of a loop splice in flexible steel wire rope of the standard length of $4\frac{1}{2}$ tucks is to be assumed not greater than 80 per cent.

For standard straining cord, where the loop is made by means of four bindings of copper wire sweated in position, the efficiency can be taken as 100 per cent.

Section VI.—Schedule of strength of materials.

1. General.—It is emphasised that the standard material properties quoted below should be used with considerable caution. If failure due to instability, or change in distribution of stress due to deformation under load, is anticipated the scheduled material properties may give no reliable indication of strength and an appeal to test will be necessary.

When compliance with requirements is based upon strength calculations the following rulings will generally apply.

(a) *Members in tension.*—The ultimate factor is to be based upon the ultimate stress, the proof factor on the 0·1 per cent. proof stress.

(*b*) *Members in compression.*—The ultimate factor is to be estimated from strut formulæ based upon the 0·2 per cent. proof stress (*see* section I, para. 1 of this chapter). It can generally be assumed that compliance with the ultimate factor requirement will automatically ensure compliance with the proof factor requirement.

2. Abbreviations.

p_1 0·1 per cent. proof stress, tons/sq. in., tensile unless otherwise stated.
p_2 0·2 per cent. proof stress, tons/sq. in., tensile unless otherwise stated.
p_5 0·5 per cent. proof stress, tons/sq. in., tensile unless otherwise stated.
Ult. Ultimate tensile stress, tons/sq. in.
y Yield stress, tons/sq. in.
E Young's Modulus/10^6 lb./sq. in.

Figures in heavy type are extracted direct from the material specifications.

A "?" preceding a figure indicates that it is a tentative value only and requires confirmation.

When upper and lower limits are quoted in a material specification the lower limit should always be used in strength calculations.

3. Strength of B.S.I. Materials.

Note.—Specifications marked * admit material of less than the nominal size. The most adverse tolerances must be taken into account in strength calculations. *See* Chapter I, para. 3.

*4.A.1. Bolts and Nuts (Low Tensile). *See* S.1 or S.61 stainless.

2.B.2. Bronze (Gun Metal) Castings. Ult. **16,** $p_1$7, $p_2$7·5, E 12·5.

*3.B.13. Brass Bars. Ult. **25,** $p_1$11, $p_2$11·5, E 14.

†4.L.1. Light Aluminium Alloy Bar (Dural.) up to 3 in. diameter. Ult. **25,** p_1**15,** $p_2$15·5, E 10·5.

*3.L.3. Light Aluminium Alloy Sheets (Dural.). Ult. **25,** p_1 **15,** E 10·5. *See* chapter IV, paras. 27 and 34.

2.L.32. Aluminium Bars. Ult. **10,** $p_1$3, $p_2$3·3, E 9. Cancelled.

L.35 Y Aluminium Alloy Castings (Heat-Treated). Ult. **14,** $p_1$12, $p_2$13, E 9·5.

*3.S.1. Bright Steel Bars. Ult. **35–45,** $p_1$27, $p_2$28·5, E 28·5.

2.S.2. 55-Ton Alloy Steel Bars. Ult. **55–65,** $p_1$45·5, $p_2$46, E 28·5.

2.S.3. Hot Rolled Mild Steel Sheets (for welding). Ult. **28,** $p_1$16·5, E 29·5.

2.S.4. 5 per cent. Nickel Steel Sheets (not for welding). Ult. **48,** $p_1$40, E 29.

2.S.6. "40" Carbon Steel Bars, etc. Ult. **35–45,** $p_1$20, $p_2$20·5, E 28·5.

3.S.11. 55-Ton Nickel-Chrome Steel Bars, etc. Ult. **55–65,** $p_1$43, $p_2$45, E ? 28.

† *Note.*—This material should preferably be used with the direction of the principal tensile or shear stress in the direction of the "grain" of the material, i.e., parallel to the axis of the bar. If the stress is transverse to the grain, the above allowable values must be reduced by 25 per cent. on proof stresses and 33 per cent. on ultimate stress.

Strength of B.S.I. Materials—*contd.*

S.61. High Chromium Steel Bars, etc. (Non-corrosive). Ult. **35–45,** $p_1$20, $p_2$21, E ? 29. *See* A.D.M. 206.

S.62. High Chromium Steel Bars, etc. (Non-corrosive). Ult. **46–52,** $p_1$30, $p_2$32, E ? 29. *See* A.D.M. 206.

S.77. " 30 " Carbon Steel, Hardened and Tempered. Ult. **30–40,** $p_1$19, $p_2$19, E 28·5.

S.80. 55-Ton Nickel-Chrome Steel Bars, etc. Ult. **55,** $p_1$45, $p_2$47, E 28.

*2.T.1. 35-Ton Steel Tubes. Ult. **35,** $p_1$29, $p_2$30, y **30,** E 28.

*2.T.2. Nickel-Chrome Steel Axle Tubes. Ult. **85–110,** $p_1$68, p_2**78,** E 29. *See* chapter IV, para. 21. 20.

*3.T.4. Duralumin Tubes, as supplied by manufacturers. Ult. **26,** p_1**18,** $p_2$19, E 10·5. *See* Chapter IV, para. 34.

*3.T.4. Duralumin Tubes, subsequently heat-treated. Ult. **25,** p_1**15,** $p_2$17, E 10·5. *See* Chapter IV, para. 34.

*T.5. Carbon Steel Tubes. Ult. **45,** $p_1$40·0, p_2**40·0,** E 29. Cancelled, *see* T.50.

*2.T.18. Hard Drawn Brass Tubes. Ult. **25–35,** $p_1$17·6, E ? 15.

T.21. Annealed Carbon Steel Tubes (obsolete). Ult. **28,** $p_1$18, $p_2$18, y **18,** E 28·5.

*T.35. 35-Ton Steel Tubes, suitable for welding.
(i) Before welding, Ult. **35,** $p_1$30, p_2**30,** E 27·4.
(ii) After welding, Ult. **30,** p_2**25,** E 27·4.

*T.45. 45-Ton Steel Tubes, suitable for welding.
(i) Before welding, Ult. **45,** $p_1$40, p_2**40,** E 28·8.
(ii) After welding, Ult. **30,** p_2**25,** E 28·8.

*T.50. 50-Ton Steel Tubes. Ult. **50,** $p_1$44·5, $p_2$45, E 28·5.

*4.V.3. All-birch plywood (3-ply)—
Tension : With outer grains at 0° to load 10,000 lb./sq. in.
,, ,, ,, ,, 45° ,, ,, 4,000 lb./sq. in.
,, ,, ,, ,, 90° ,, ,, 6,500 lb./sq. in.

Shear : With outer grains parallel to longitudinal axis of member : 1,800 lb./sq. in.

With grains at 45° to longitudinal axis of member : 2,100 lb./sq. in.

3.V.4. Ash E **1·5** ; modulus of rupture **10,500** lb./sq. in. ; end grain tension 12,700 lb./sq. in. ; end grain compression **5,800** lb./sq. in.

3.V.5. Walnut E **1·5** ; modulus of rupture **11,500** lb./sq. in. end grain compression **7,000** lb./sq. in.

4.V.7. Mahogany E **1·5** ; modulus of rupture **10,000** lb./sq. in. ; end grain compression **6,250** lb./sq. in.

Strength of D.T.D. Materials.

24A. Non-Corrosive Steel Rivets and Split Pin Wire. Ult. **30,** $p_1$15, E ? 30. Cancelled, *see* D.T.D. 161, 185, 189.

36A. Silver spruce or approved substitutes. E **1·5** ; modulus of rupture 8,000 lb./sq. in. ; end grain compression **5,000** lb./sq. in. ; end grain tension 10,000 lb./sq. in. ; combined bending and compression 5,500 lb./sq. in. ; shear 900 ~~1,600~~ lb./sq. in. ; crushing on side grain 600 lb./sq. in.

A.L.3.

*41. Weldable Mild Steel Tube (before welding). Ult. **30,** $p_1$26·5, $p_2$27, y **28,** E 28.

*41. Weldable Mild Steel Tube (after welding). Ult. **24,** $p_1$14, $p_2$14, y **17,** E 27.

42. Chromium Nickel Non-Corrosive Steel Sheets (Hard). Ult. **55,** $p_1$31, y **35,** E 27. Cancelled, *see* D.T.D. 144.

42. Chromium Nickel Non-Corrosive Steel Sheets (Soft). Ult. **40,** $p_1$13, y **15,** E 27. Cancelled, *see* D.T.D. 144.

43. Non-Corrosive Steel Bars (Hard) up to 2 in. diameter. Ult. **50,** y **22,** E ? 28·5. Cancelled, *see* D.T.D. 156.

43. Non-Corrosive Steel Bars (Hard) above 2 in. diameter. Ult. **45,** y 22, E ? 28·5. Cancelled, *see* D.T.D. 156.

43. Non-Corrosive Steel Bars (Soft). Ult. **40,** y **15,** E ? 30. Cancelled, *see* D.T.D. 156.

*46A. Non-Corrosive Steel Strip, Hardened and Tempered. Ult. 86·7, p_1**65,** E 29.

*53. Non-Corrosive Low Tensile Steel Bar. Ult. **28–35,** $p_1$13, $p_2$14, E 26·5.

*54A. High Tensile Nickel Chromium Steel Strip (Hard). Ult. 86·7, p_1**65,** E 29.

*57B. Chromium Nickel Non-Corrosive Steel Sheet and Strip. Ult. **54,** $p_1$37, p_5**50–60,** E ? 26·5. Cancelled, *see* D.T.D. 166.

60A. High Chromium Non-Corrosive Steel Sheet and Strip. Ult. **55,** $p_1$40, $p_2$43, p_5**45,** E 29·0.

78A. Hard Drawn Phosphor Bronze Bars, up to 2 in. dia. Ult. **30,** p_1**15,** $p_2$16, E 17·0.

78A. Hard Drawn Phosphor Bronze Bars, above 2 in. dia. Ult. **28,** p_1**15.**

*89A. Weldable Steel Tubes (before welding). Ult. **45,** $p_1$40, $p_2$40, y **40,** E 29. Cancelled, *see* B.S. T.45.

*89A. Weldable Steel Tubes (after welding). Ult. **30,** $p_2$18·5, y **25,** E 29. Cancelled, *see* B.S. T.45.

*91A. 50-Ton Steel Tubes (Hard Drawn and Blued). Ult. **50,** y **45,** E ? 30. Cancelled, *see* B.S. T.50.

*97. Low Tensile Non-Corrosive Tubes. Ult. **28,** $p_1$15·5, $p_2$16, y **18,** E 28·5.

Strength of D.T.D. Materials—*contd.*

*99. Nickel-Chrome Steel Strip (Hardened and Tempered). p_1**55,** E ? 27.

*102. 35-Ton Non-Corrosive Tubes. Ult. **35,** $p_1$28, $p_2$29, y **30,** E 29.

*105. 50-Ton Non-Corrosive Tubes. Ult. **50,** $p_1$32, $p_2$36, y **40,** E 29·5.

*111. " Alclad " Aluminium Coated Light Alloy Sheet. Ult. **24,** $p_1$13·5, E ? 9·7. *See* chapter IV, para. 27.

*113. Steel Tubes suitable for welding (before welding). Ult. **35,** $p_1$30, $p_2$30, y **30,** E 28·5. Cancelled, *see* B.S. T.35.

*113. Steel Tubes suitable for welding (after welding). Ult. **30,** $p_2$19, y **25,** E 29. Cancelled, *see* B.S.T.35.

*118. Magnesium Alloy Sheets, Ult. **11.**

*124. 40-Ton Proof Carbon Steel Strip (before welding). Ult. 42, p_1**40,** E ? 28.

*124. 40-Ton Proof Carbon Strip (after welding). Ult. **30,** E ? 28.

126. Carbon Steel Bars, suitable for welding. Ult. **40**, $p_1$31, $p_2$34, E 28·0.

129. Magnesium Alloy Bars. Up to 2 in. dia. Ult. **20**, p_1**12**, $p_2$14, E 6·4.

*141. Cold Rolled, Close Annealed, Mild Steel Sheets, Ult. **20–28.**

142. Magnesium Alloy Bars. Ult. **15,** p_1**8,** $p_2$10, E 6·0.

*146A. High Chromium, Non-Corrodible Steel Sheet and Strip, Ult. **40,** p_1**30.**

153. Bright Steel Bars (for Pins and H.T. Bolts). Ult. **55,** $p_1$44, $p_2$49, E 27.

161. Non-Corrodible Steel Rods, Ult. **30,** $p_1$14, $p_2$14·5, E 26.

*166A. Chromium Nickel Non-Corrosive Steel Sheet. Ult. **52–70,** p_1**40–50,** E ? 25·4.

*167. 45-Ton Steel Tubes. Ult. **45,** $p_1$35·5, $p_2$39, y **40,** E 29·5.

*168. High Chromium Non-Corrodible Steel Sheet and Strip.
(i) Soft, Ult. **50.**
(ii) Hardened and Tempered, Ult. **60.**

*171A. Chromium Nickel Non-Corrodible Steel Sheet and Strip, Ult. **35,** p_1**15.**

176A. Chromium Nickel Non-Corrodible Steel, Ult. **35,** p_1**15,** $p_2$16·0, E 28·0.

185A. High Chromium Non-Corrodible Steel Rod, Ult. **30–50,** $p_1$17·5, $p_2$19, E 24·0.

Strength of D.T.D. Materials—*contd.*

*186A. 7 per cent. Magnesium-Aluminium Alloy Tubes (hard).
(i) 12 s.w.g. and under, Ult. **26,** p_1**18.**
(ii) Over 12 s.w.g., Ult. **25,** p_1**17.**

189. Chromium Nickel Non-Corrodible Steel Rod, Ult. **30,** $p_1$10·5, $p_2$11·0, E 28·0.

*190. 7 per cent. Magnesium-Aluminium Alloy Tubes (annealed), Ult. **20–23,** p_1**10.**

198. 7 per cent. Magnesium Aluminium Alloy Rivets. Ult. **20,** $p_1$13·5, $p_2$14·0, E 10·0.

*199. 50-Ton High Chromium Non-Corrodible Steel Tubes, Ult. **50,** $p_1$40, p_2**45,** E 30·5.

*203. 50-Ton Non-Corrodible Steel Tubes, Ult. **50,** p_2**45.**

*207. 35-Ton Chromium Nickel Non-Corrodible Steel Tubes, Ult. **35,** $p_1$14, $p_2$16, E 27.

*211. 50-Ton Chromium Nickel Non-Corrodible Steel Tubes, Ult. **50,** $p_1$37, p_2**45,** E 25·5.

*220. Wrought Light Aluminium Alloy Tubes, Ult. **27,** p_1**22.**

*225. High Chromium, Non-Corrodible Steel Sheets and Strips, Ult. **35–40,** p_1**20.**

Section VII—Direction of grain in fittings

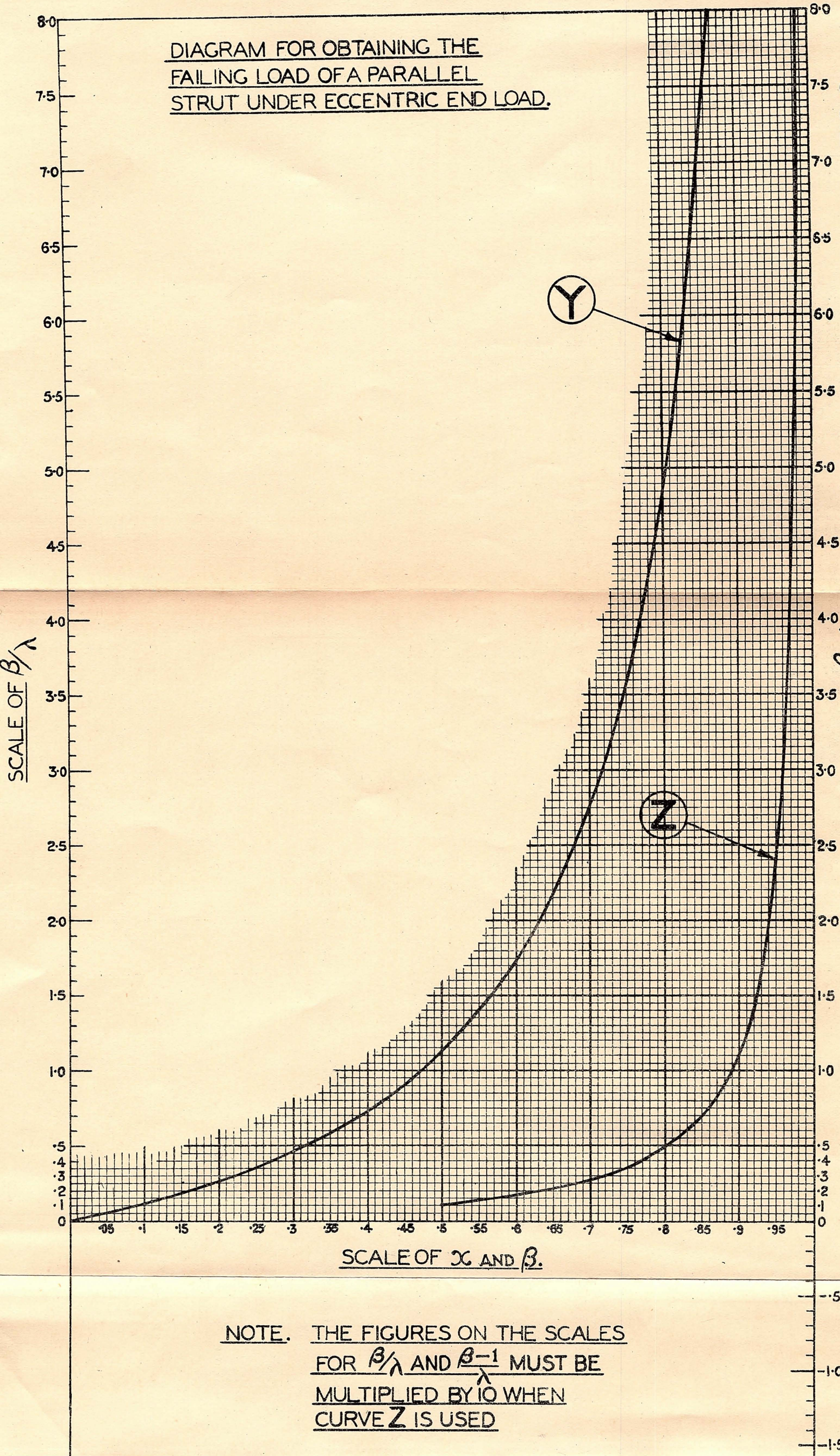
The importance of grain direction as affecting the ability of parts to withstand shock loading is clearly indicated by the Izod test, the Izod figure being considerably reduced when the direction of the blow is parallel to the grain.

This is of special importance in the case of machined parts, and generally the best disposition of grain can be obtained by suitable forging or stamping. It may be that considerations of cost will sometimes preclude this course, and in such cases the design of the part should be carried out * in such a way as best to dispose the grain of the metal to resist shock. This especially applies to B.S. Specification S.80. This material must not be used for machined fittings unless the grain is suitably disposed and the drawings of the parts are to show the grain direction.

In the case of bars of $2\frac{1}{2}$ in. diameter and over, attention is drawn to the note with regard to the use of heat-treated bars of large diameter now issued with B.S. Specifications for aircraft steels. This note affords an added reason for adopting the course outlined in the preceding paragraph.

FIG. 2.

DIAGRAM FOR OBTAINING THE FAILING LOAD OF A PARALLEL STRUT UNDER ECCENTRIC END LOAD.



NOTE. THE FIGURES ON THE SCALES FOR β/λ AND $\frac{\beta-1}{\lambda}$ MUST BE MULTIPLIED BY 10 WHEN CURVE Z IS USED

CHAPTER IX.—AIRSCREWS

Section I.—Calculation of performance and stresses

1. Estimation of aerodynamic performance

The aerodynamic performance of an airscrew can be calculated by the " Vortex Theory " of R. & M. 786 and 869 or by the later theory of R. & M. 1521, and the torque and thrust coefficients and the efficiency over the normal working range of advance per revolution obtained. Such analysis is, however, somewhat protracted and it is usually not necessary to calculate the torque and thrust coefficients at more than two or three values of advance per revolution in order to determine whether the strength requirements are satisfied. For this reason, it is usual to employ a method of calculating, by successive approximation, the aerodynamic performance at certain predetermined values of advance per revolution.

By this method, the inflow factor a, for the particular value of advance per revolution under consideration, is first estimated and the subsequent analysis follows the well-known method of Lanchester and Drzewiecki. In determining the inflow factor, it is assumed that $a = \frac{b}{2}$, which is in accordance with the " Vortex Theory," where

$V(1 + a)$ = velocity of the inflowing air at the disc, and

$V(1 + b)$ = maximum velocity of the outflowing air behind the disc, both relative to the screw ;

and V = velocity of advance of the screw with reference to still air.

The inflow factor is assumed to be constant over the annulus bounded by circles of radii $\frac{D}{4}$ and $\frac{D}{2}$ and is assumed to fall linearly to zero from radius $\frac{D}{4}$ to radius $\frac{D}{6}$.

In the analysis, due allowance must be made for the effect of the actual density of the air in which the screw is working, both on the airscrew characteristics and on the B.H.P. developed by the engine.

For air at standard sea level temperature and pressure, the density ρ is equal to $0 \cdot 002378$ slugs per cu. ft. and the density at any other altitude is given by fig. 1 where the density ratio σ is plotted against altitude.

In the case of normally aspirated engines, if.—

H_h = B.H.P. developed by the engine at the altitude under consideration and at the crankshaft speed permitted by the airscrew,

and H_o = B.H.P. developed by the engine at the same crankshaft speed at ground level,

then $H_h = f\, H_o$

where f is the power factor and may be taken as equal to the pressure ratio (also given in fig. 1).

In the case of ground boosted or supercharged engines, the B.H.P. developed below the rated altitude will be obtained directly from power curves. Above the rated altitude the power may be taken as proportional to the ratio of the atmospheric pressure to that at the rated altitude.

2. Estimation of radial fibre stresses

(i) *General.*—The selection of the particular conditions of flight in which the maximum steady radial fibre stresses are likely to occur is a matter requiring careful consideration in each case. Generally, with wooden airscrews, the stresses are greatest in the static or " take-off " condition when the engine is at full throttle or maximum permissible boost. In addition, the conditions must be considered of full throttle climb at the rated height, full throttle level flight at the maximum power altitude and high speed dives engine on or off.

With metal airscrews, each of the above-mentioned conditions must also be considered, but, in general, the maximum steady radial stresses will occur in the static or " take-off " condition or in high speed dives.

A fairly rigid determination of the radial stresses in airscrew blades can be made by use of either of the two methods of R. & M. 420. The estimation of the stresses by either of these methods is, however, somewhat involved, and experience has shown that the following approximate method is sufficiently accurate for the investigation of the strength of blades of conventional design under normal conditions of use.

The important steady radial stresses occurring at any blade section are as follow.—

(*a*) Direct tension due to centrifugal force,
(*b*) Fibre stress due to bending resulting from the air reactions of torque and thrust,
(*c*) Fibre stress due to bending resulting from centrifugal force,

each of which is calculated separately and summed algebraically to give the total stress in the tabular form shown in fig. 2.

At the blade roots and at the hub sockets or boss, these stresses will be accompanied by important longitudinal shear stresses which will usually be greatest for the high speed dive condition of flight and which must be determined and considered in the design.

(ii) *Calculation of tension due to centrifugal force.*—Consider a strip of the blade of radial length dr at a mean radius r and let the area of cross-section of the blade at this point be A. Then the centrifugal force of the element is

$$dF = A\frac{\lambda}{g}\,\Omega^2\,rdr$$

where λ is the density of the material of which the blade is constructed, and Ω is the angular velocity in radians per second. Hence the total centrifugal force acting at a section of the blade distant r_1 from the centre is

$$F_{r1} = \frac{\lambda}{g}\,\Omega^2 \int_{r_1}^{D/2} A r dr$$

The centrifugal tensile stress at any radius r_1 is then given by

$$f_{r1} = \frac{F_{r1}}{A_{r1}}$$

where A_{r1} is the area of the cross-section. The areas of cross-section can be obtained by graphical means or by use of the formulæ given in fig. 3, which will be found sufficiently accurate in most cases.

For wooden airscrews with blades to which metal sheaths are fitted, the additional tensile stress due to the sheath is calculated on the assumption that any section of the blade has to withstand the centrifugal force of the portion of the sheath between that section and the tip of the blade.

(iii) *Calculation of fibre stresses due to bending resulting from the air reactions of torque and thrust.*—Consider an element of the blade of radial length dr at a mean radius r, where the chord width is c and the air pressure p. Then the load on the element is $pcdr$ and the bending moment at radius r_1, caused thereby, is

$$dM = pc\,(r - r_1)\,dr$$

Assuming that the resultant forces on the blade act normal to the chord and neglecting the variation of angle along the length of the blade, the total aerodynamic bending moment on any section is the sum of all such elemental moments over that portion of the blade between the section under consideration and the tip, i.e.,

$$Mr_1 = \int_{r_1}^{D/2} pc\,(r - r_1)\,dr.$$

Hence the aerodynamic bending moments at all radii can be evaluated by integrating curves of $pc\ (r - r_1)$ plotted against r (fig. 4).

The air pressure p is obtained directly from the formula

$$p = \frac{\rho\, k_L\, W^2}{\cos \gamma}$$

where W is the velocity of the air relative to the element, and

$$\gamma = \cot^{-1} \frac{k_L}{k_D}$$

As γ is usually small, $\cos \gamma$ in general approximates to unity. The values of lift and drag coefficients used must be appropriate to the aerofoil sections of the element considered and must have been corrected to infinite aspect ratio (*see* Appendix A).

If, however, the " Vortex Theory " is used, then according to R. & M. 869

$$W = r\Omega\ (1 - a_2) \sec \phi,$$

where a_2 is the rotational interference factor, so that p can be obtained from the equation

$$p = \frac{\rho k_L r^2 \Omega^2\ (1 - a_2)^2 \sec^2 \phi}{\cos \gamma}$$

where ρ, r, Ω, a_2 and γ are as previously defined and ϕ is the angle which W makes with the plane of rotation.

Employing the engineer's theory of bending, and assuming the major principal axis of the cross-section of the blade to be parallel to the chord, then the fibre stresses due to the bending moments calculated as described, follow immediately when the moduli of resistance of the sections Z_c and Z_r, for compression and tension respectively, are known. These moduli are determined about an axis through the centroid of the section and parallel to the chord. Their exact values can be determined by graphical means, but for most purposes the approximate values given in fig. 3 will be found sufficiently accurate. In any case of doubt or where the sections are of appreciably different form from those illustrated in fig. 3, the true moduli should be obtained graphically.

(iv) *Calculation of bending stresses due to centrifugal force.*—Unless the locus of the centroids of the cross-sections of the blade is a radial line contained in a plane parallel to that of rotation, bending stresses arise from centrifugal force. As in para. 2 (ii) the centrifugal force of each element of the blade is

$$dF = \frac{\lambda}{g}\, \Omega^2\, Ar\, dr$$

which force lies in a plane parallel to that of rotation and acts radially through the centroid of the element under consideration. If the line of action of each such elemental centrifugal force be projected towards the centre of the screw and z be the departure of the point of intersection of each such line with respect to the centroid of the section about which the bending moment is to be found, then

$$dM = \frac{\lambda}{g}\, \Omega^2\, Arz\, dr \quad \text{.. .. .. .. (fig. 5).}$$

is the bending moment at that section due to the centrifugal force of the outer element considered. This elemental bending moment can be resolved into two components

$$dM_1 = \frac{\lambda}{g}\, \Omega^2\, Arz \sin \beta\, dr$$

and

$$dM_2 = \frac{\lambda}{g}\, \Omega^2\, Arz \cos \beta\, dr$$

where β is the angle between z and the principal axis of the section considered.

The former moment, which produces bending about the assumed major principal axis, thereby inducing radial fibre stresses on the pressure and suction faces of the blade, need alone be considered. The latter moment can be neglected as the fibre stresses occurring near the leading and trailing edges which result therefrom are small.

The bending moment at any section at radius r_1 about the assumed major principal axis, due to the centrifugal forces of all outer elements, is

$$M_{r1} = \frac{\lambda}{g}\,\Omega^2\int_{r_1}^{D/2} Arz \sin\,\beta\,dr.$$

The component departures $z \sin\,\beta$ are best found graphically on a true-to-scale diagram of the blade, whence curves of $Arz \sin\,\beta$ are plotted against r (fig. 6), and the areas enclosed thereby determined. For wooden airscrews with blades to which metal sheaths are fitted, similar measurements are made of the component departures of the centrifugal forces of elements of the metal sheathing. The bending moments at any section at radius r_1 about the assumed major principal axis, due to the centrifugal forces of all outer elements of the sheathing, is then obtained in a similar manner and added algebraically to the bending moments due to the centrifugal force of the outer part of the blade.

(v) *Effect of blade deflection.*—The stresses in the blades will be affected by their deflection under load. For wood, however, the deflection cannot be calculated with accuracy and, as the effect of deflection will usually be to decrease the calculated stresses, the neglect of this factor errs on the safe side.

For solid metal blades it is possible to estimate the deflection under load, and where necessary such deformation, which will reduce radial stresses, can be taken into account in strength determination. The deflection at any section of the blade at radius r_1 is given by

$$\frac{1}{E}\int\int_{r_1}^{D/2}\frac{M}{I}\,dr\,dr$$

where M is the nett bending moment, I the minor moment of inertia of the section and E is Young's Modulus. As I varies along the length of the blade in a manner which cannot be simply expressed as a function of r, and since the nett bending moment varies with the deflection, the integration cannot be performed analytically, and graphical methods of successive approximation have to be adopted.

For this purpose, the blade is regarded as flat at an angle corresponding to that obtaining at 0·7 of the tip radius and a probable deflection curve assumed, the deflection being taken normal to the assumed flat blade. The centrifugal bending moments corresponding to this deflected state are determined in the manner previously described, and added algebraically to the aerodynamic bending moments, thus giving the resultant bending moment at each section. The deflection at any section of the blade at radius r_1 is then found from the formula given above.

The method is continued as a series of successive approximations until the assumed and calculated deflection curves are in reasonable agreement. Unless the actual stresses are required it is not necessary to carry the process further than the stage at which it can be seen that the stresses will not exceed the maximum permitted. Deflection need only be taken into account in cases where the nett bending moments calculated for the undeflected blade lead to stresses in excess of the maximum permissible.

3. Blade stiffness of wooden airscrews

It is not yet possible to issue precise rules for the prevention of airscrew blade flutter, but the following information supplements that already published in R. and M. Nos. 1258 and 1518.

R. and M. 1258 gives the variation of thickness-chord ratio with radius and the minimum values of this ratio recommended for mahogany blades of normal chord-diameter ratio. Experience has shown, further, that a heavy blade tip may induce a tendency to flutter unless the stiffness of the other parts of the blade is above normal.

The values of maximum thickness and chord, therefore, when plotted against radius, should give faired curves and these values should in general decrease with increase of radius ove outer part of the blade. This is consistent with the principle, which is important for airs of all types, that change in section and in stiffness throughout the length of the blades shou be gradual.

Section II.—Design requirements

1. General requirements

The following requirements apply to airscrews irrespective of the material of construction.

(i) The design of fixed-pitch airscrews and also of adjustable-pitch airscrews, with the blades set at the appropriate pitch, must be such that.—

(*a*) in full throttle level flight at the maximum power altitude of the engine, the maximum emergency crankshaft speed is not exceeded ;

(*b*) in a full throttle climb at the best climbing speed at the rated height of the engine, the maximum crankshaft speed established for the type engine in climbing flight is not exceeded ;

(*c*) at take-off, the crankshaft speed is not less than the minimum nor more than the maximum take-off r.p.m. established for the type engine at the take-off power or boost.

The design of variable-pitch airscrews must be such that the crankshaft speeds can be regulated as above by means of the pitch or governor speed control.

(ii) The maximum total steady radial stress, including any stress due to metal sheathing and protective covering, must not exceed, in all conditions of use, the allowable value for the material of construction (*see* paras. 2 (ii) and 3 (i)).

(iii) For blades of a material or of a form of construction with which a high margin of torsional strength is not obtainable, the plan form must be geometrically symmetrical, or approximately so, about a radial line extending from the centre of rotation to the tip. Moreover, the axis of symmetry of each blade, when viewed in elevation, must be straight or approximately so, and must not be tilted forward of a line normal to the axis of rotation by an amount appreciably in excess of that required to avoid the stresses exceeding the allowable value for the material.

(iv) The blades must be sufficiently rigid to resist flutter in all conditions of use. The first or a similar airscrew of any conventional design may be required satisfactorily to complete test (*a*) and, where applicable, test (*b*) of para. 1 (viii) if doubt exists whether the blades are sufficiently stiff to resist flutter.

(v) The design must be such that static, dynamic and aerodynamic balance, within permissible limits, can be attained during manufacture of the airscrew and, in the case of detachable blades, the design must be suitable for the production of blades which are interchangeable to a reasonable extent with a standard blade of that design.

(vi) The diameter and (in the case of fixed-pitch airscrews) the pitch are to be quoted in the drawings in feet and in decimals of a foot, the pitch being calculated by the formula.

$$P = \text{pitch in feet} = 2{\cdot}2D \tan \theta,$$

where D is the diameter of the airscrew in feet and θ is the angle in degrees with the plane of rotation of a section of the blade distant $0{\cdot}7$ of the tip radius from the axis of rotation. The angle θ is that included between the plane of rotation and the chord of the section, when the pressure face is curved, or between the plane of rotation and the pressure face of the section when this face is flat over the greater part of the width.

(vii) The blades and hubs of adjustable-pitch airscrews must be provided with suitable marks so that the blades can be set at the same pitch within the range of pitches required in flight.

(viii) The first or a similar airscrew which incorporates untried features, whether of design or construction, or which incorporates detachable blades of new design may be required satisfactorily to complete the test (*a*) detailed below. Where the pitch is adjustable or variable, the airscrew must satisfactorily complete tests (*b*) and (*c*) following the satisfactory completion of test (*a*), and the design is to be such that the blades can be set and locked at the pitches required to enable tests (*a*) and (*b*) to be made. Where the pitch is fixed, the airscrew may be required satisfactorily to complete endurance bench tests on an engine of the type for which it is designed, details of the tests being arranged in consultation with the designer. Tests (*a*), (*b*) and (*c*) will normally be made under static conditions of running. When the airscrew is provided with a detachable spinner and/or nose cap these parts are to be fitted to the airscrew for tests (*a*) and (*c*).

(*a*) Spinning, by means of electric motors at a rotational speed 5 per cent. in excess of that at which is absorbed the established minimum take-off power of the engine for which the airscrew is designed. Where the pitch is adjustable or variable this test will be made with the blades set to absorb the established take-off power when the airscrew is running at a speed corresponding to the minimum take-off r.p.m. This test will normally be of 30 min. duration.

(*b*) Spinning, by means of electric motors at a rotational speed 5 per cent. in excess of that corresponding to the established maximum diving speed of the engine for which the airscrew is designed, except for variable-pitch airscrews with automatic governor control, where the speed of the test will be 5 per cent. in excess of the highest speed in flight permitted by the automatic control. The pitch of the blades for this test may be determined by the designer and will normally be such that, during the test, not more than the established maximum climbing power of the engine is absorbed. The duration of this test will normally be 1 hour.

(*c*) Endurance bench test of 50 hr. duration on an engine of the type for which the airscrew is designed (*see* para. 4).

(ix) Adjustable or variable-pitch airscrews which have previously been approved for use on an engine of specified B.H.P. and r.p.m. may, after investigation, be approved for flight tests, without further spinning or bench tests, for use on another type of engine of similar power rating.

2. Particular requirements for fixed-pitch wooden airscrews

(i) The designs are to be suitable for construction in mahogany, B.S. Specification V.7 and must satisfy strength requirements in this material.

(ii) The allowable stresses for the design of mahogany airscrews are.—

Compression	2,500 lb./sq. in.
Tension	4,000 lb./sq. in.
Longitudinal shear	500 lb./sq. in.

(iii) Blades must be designed to be capable of carrying, under all conditions of use, a metal sheath of approved design suitable for the type of aeroplane and the size of the airscrew.

(iv) Airscrew drawings must comply with the requirements of S.I.S. No. 145 and must specify the type of protective finish to be used.

(v) All laminae are to be continuous at the boss except that, when essential to the design, discontinuous outer laminae may be permitted up to a maximum total thickness of one-hundredth of the diameter of the airscrew on each side.

(vi) Except where specially conceded, four-blade airscrews must be of two-part construction.

(vii) Where metal spinners are provided the drawings are to call for discs of emery, glued back to back, or of other approved friction material to be interposed between the adjacent metal surfaces of spinner plates and hub flanges.

3. Particular requirements for metal airscrews

(i) The allowable tensile or compressive stresses for the design of metal airscrews, subject to the provisions of sub-paras. (ii) and (iii) are.—

(*a*) Duralumin to Specification No. D.T.D.147 or D.T.D.150	6·75 tons/sq. in.
(*b*) Mild Steel sheet to B.S. Specification S.3	10·0 tons/sq. in.

(ii) In blades of hollow construction, the maximum steady compressive stress, in any condition of use, must not exceed one-half of the steady compressive stress at which instability of the section occurs.

(iii) In order to reduce the risk of fatigue failure, a margin of safety, additional to that indicated by the allowable stresses given in sub-para. (i), must be allowed at the roots and over the inner parts of the blades, particularly where abrupt changes of section are necessary. The allowable values for the stresses in these parts of the blade and, in the case of detachable blades, the corresponding values for shear and bearing stresses, will be determined in consultation with the designer.

(iv) The drawings must be fully dimensioned and must give particulars of the material and finish. Furthermore they must either call for manufacture to be in accordance with Specification G.E.139 and give particulars of the manufacturing tolerance and processes, or refer to an approved specification which contains full instructions for manufacture.

(v) The surface of metal airscrews, unless manufactured from non-corrodible material, must be protected in an approved manner against corrosion, both internal and external surfaces being treated when hollow construction is employed. For duralumin, mild steel and magnesium alloy airscrews, the approved protectives are anodic treatment, stove enamel and chromate treatment, respectively.

4. Engine bench tests of adjustable or variable pitch airscrews

(i) For test (*c*) of para. 1 (viii), the airscrew is to be fitted to the engine and the undermentioned schedule of tests is to be followed, the pitch or pitch controls and the engine throttle or boost pressure being adjusted to permit the engine to run under the specified conditions. If the airscrew has already been submitted to engine tests during the type trials of the engine, the tests so made need not be repeated for approval of the airscrew, provided the same airscrew is used for any remaining bench tests.

(ii) A proof of two hours' duration is to be made with the engine running at the maximum r.p.m. and power established for continuous cruising flight at sea level. The airscrew and pitch controlling mechanism, if any, are then to be removed from the engine and after balance, blade angles and track dimensions of the airscrew have been checked, are to be completely dismantled for examination and the recording of the condition and the clearances between any parts in relative motion one with another.

(iii) On re-assembly and re-fitting of the airscrew, the engine is to be run, at the r.p.m. and power specified in sub-para. (ii), for 20 hours in two 10-hour non-stop runs. During the last few minutes of every two hours of this test the airscrew pitch controls, if any, are to be operated through the normal range without change of engine throttle opening provided the r.p.m. are not reduced below the established minimum take-off r.p.m. or raised above the maximum emergency r.p.m. At the end of each 10 hours the airscrew hub and blades are to be examined in position for excessive wear or other defects.

(iv) The engine is then to be run, at the maximum r.p.m. and power established for climbing flight at sea level, for 20 hours in convenient periods, the airscrew pitch controls, if any, being operated during the last few minutes of each two hours as specified in sub-para. (iii). At the end of each 10 hours and also at other convenient periods, the hub and blades are to be examined in position for excessive wear or other defects.

(v) The following non-stop runs are then to be made.—

(*a*) Two hours at the established minimum take-off conditions.

(*b*) Two hours at the established maximum take-off conditions.

(*c*) Two hours at r.p.m. intermediate between the minimum and maximum take-off r.p.m. and at the established take-off boost or power.

(*d*) Four quarter-hour runs to cover the normal cruising range of r.p.m. and power, at 80, 85, 90 and 95 per cent. of the maximum r.p.m. established for continuous cruising flight at approximately 50, 60, 70 and 85 per cent., respectively, of the power established for continuous cruising at sea level.

(*e*) One hour at 5 per cent. in excess of the maximum emergency r.p.m. at approximately 30 per cent. of the rated power or, if necessary, at the higher power associated with the minimum pitch obtainable. For variable-pitch airscrews with automatic governor speed control, this test will be made at the highest speed permitted by the governor control in flight and at the minimum power required to run the engine and airscrew at this speed.

(vi) With the engine running under the conditions specified in sub-para. (ii) for the proof test, the throttle setting is to be reduced until the engine runs at 25 per cent. of the maximum r.p.m. established for continuous cruising flight. At the end of five minutes the engine is to be accelerated rapidly to full throttle or maximum boost for level flight at least five times.

(vii) Finally, the airscrew and pitch controlling mechanism, if any, are to be removed from the engine and, after balance, blade angles and track dimensions of the airscrew have been checked, are to be completely dismantled for examination and the recording of the condition and the clearances between any parts in relative motion one with another.

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CURVES OF MEAN DENSITY AND PRESSURE
RELATIVE TO STANDARD ATMOSPHERE.

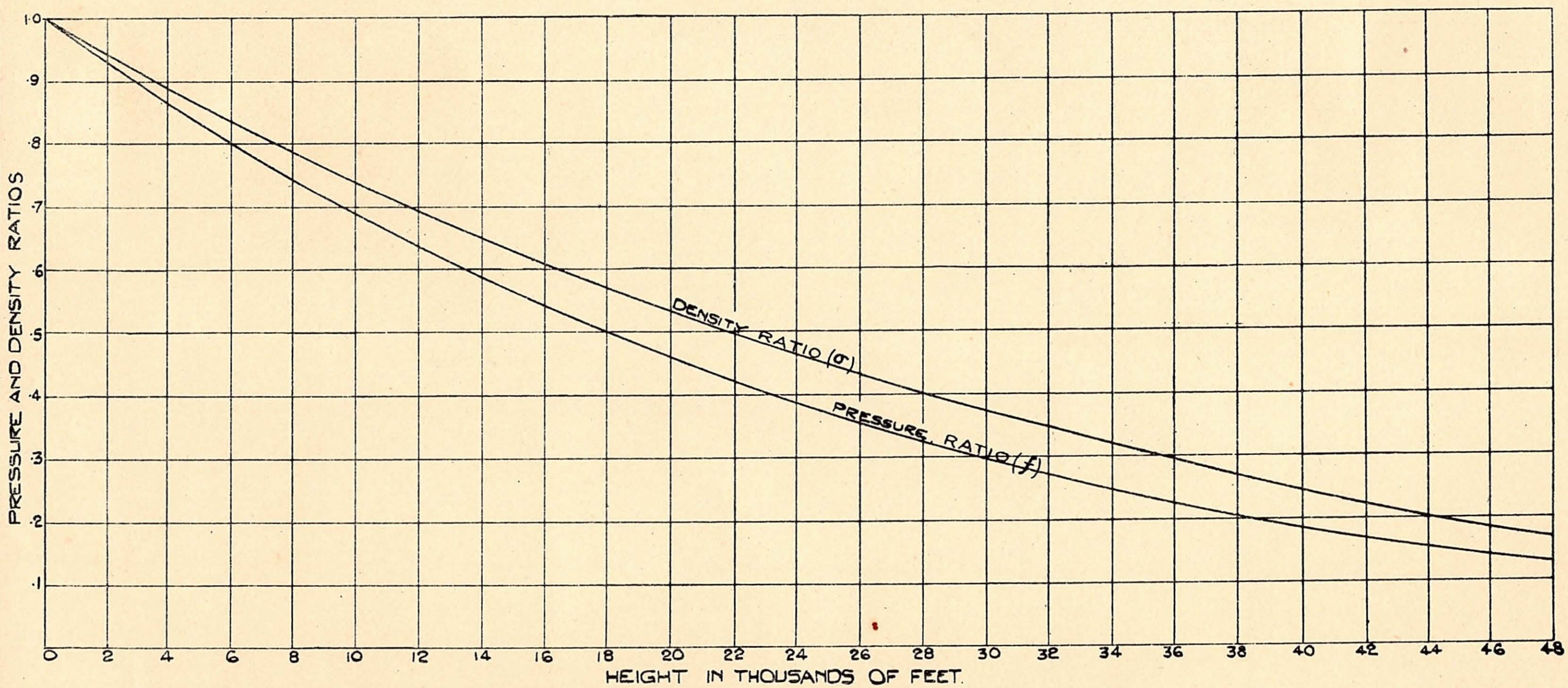

FIG. 1. CHAP. IX.

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SUMMARY OF AIRSCREW CALCULATIONS.

AIRCRAFT:-

ENGINE:- GEAR RATIO:-

DRG Nº:- DIA. FT:- PITCH. FT:- Nº OF BLADES:-

ALTITUDE FT:- M.P.H:- R.P.M:- =R.P.S:-

B.H.P.:- η :-

AEROFOIL DATA USED:-

MATERIAL:-

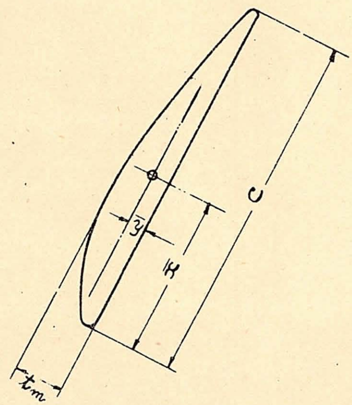
RADIUS r FEET								
CHORD C INS.								
THICKNESS t_m INS.								
* DEFLECTION INS.								
Z INS.^{3} COMPRESSION								
Z INS.^{3} TENSION								
AERO. B.M. LB. INS.								
CENT. B.M. LB. INS.								
NETT. B.M. LB. INS.								
NETT B.M. STRESS LB./INS.^{2} COMPN								
NETT B.M. STRESS LB./INS.^{2} TENSION								
CENTRIFUGAL FORCE								
AREA OF SECTION INS.^{2}								
CENTRIFUGAL TENSILE STRESS LB./INS.^{2}								
TOTAL STRESS LB./INS.^{2} COMPN								
TOTAL STRESS LB./INS.^{2} TENSION								

* SOLID METAL AIRSCREWS ONLY.

FIG. 2. CHAP. IX.

APPROX: FORMULAE FOR POSITIONS OF CENTROIDS, AREAS AND MODULI OF RESISTANCE OF TYPICAL AIRSCREW SECTIONS.

THE CONSTANTS EMPLOYED IN THESE FORMULAE HAVE BEEN DERIVED FROM AN ACCURATE COMPUTATION OF A LARGE NUMBER OF SIMILAR SECTIONS.



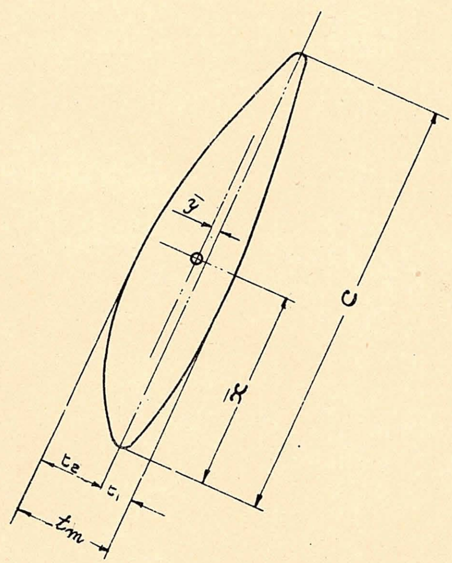
$\bar{x} = \cdot 46\, C$

$\bar{y} = \cdot 41\, t_m$

AREA OF SECTION $= \cdot 7\, C\, t_m$

$Z_C = \cdot 08\, C\, t_m^2$

$Z_t = \cdot 12\, C\, t_m^2$



$\bar{x} = \cdot 46\, C$

$\bar{y} = \cdot 41\, (t_2 - t_1)$

AREA OF SECTION $= \cdot 7\, C\, t_m$

$Z_C = \cdot 08\, C\, t_m^2$

$Z_t = K\, C / t_m^2$ THE VALUE OF K BEING OBTAINED FOR ANY VALUE OF t_1/t_2 FROM THE CURVE BELOW.

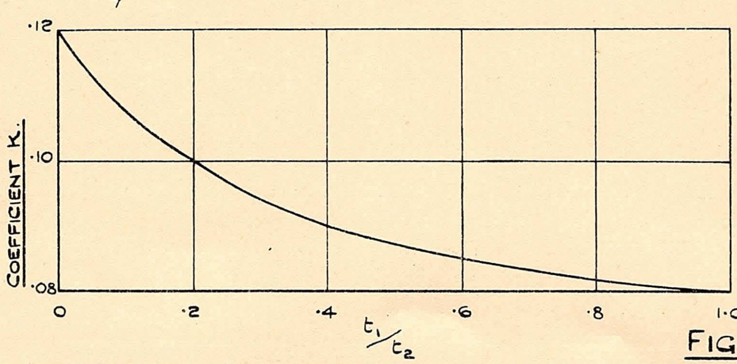


FIG. 3. CHAP. IX

AERODYNAMIC BENDING MOMENT.

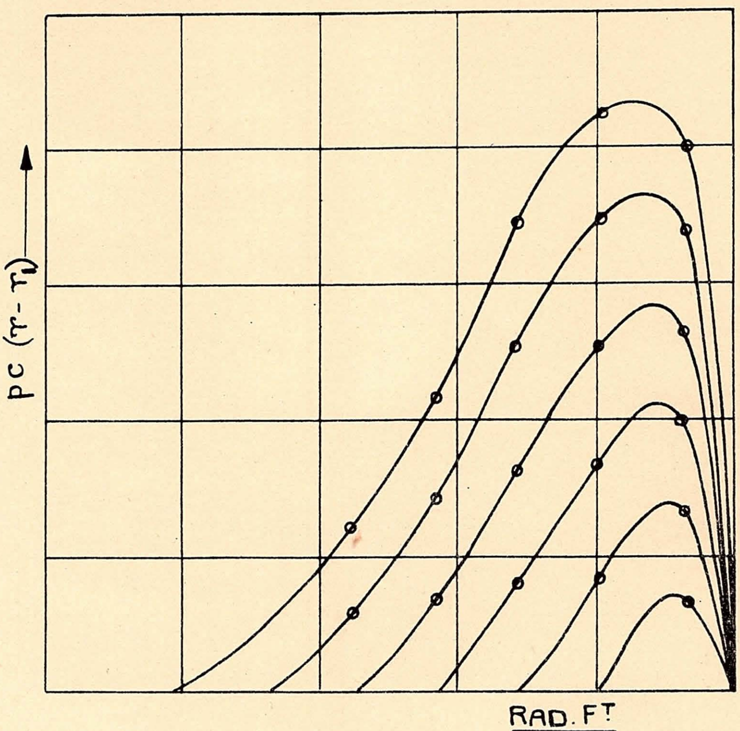


FIG. 4. CHAP. IX.

CALCULATION OF BENDING STRESSES DUE TO CENTRIFUGAL FORCE.

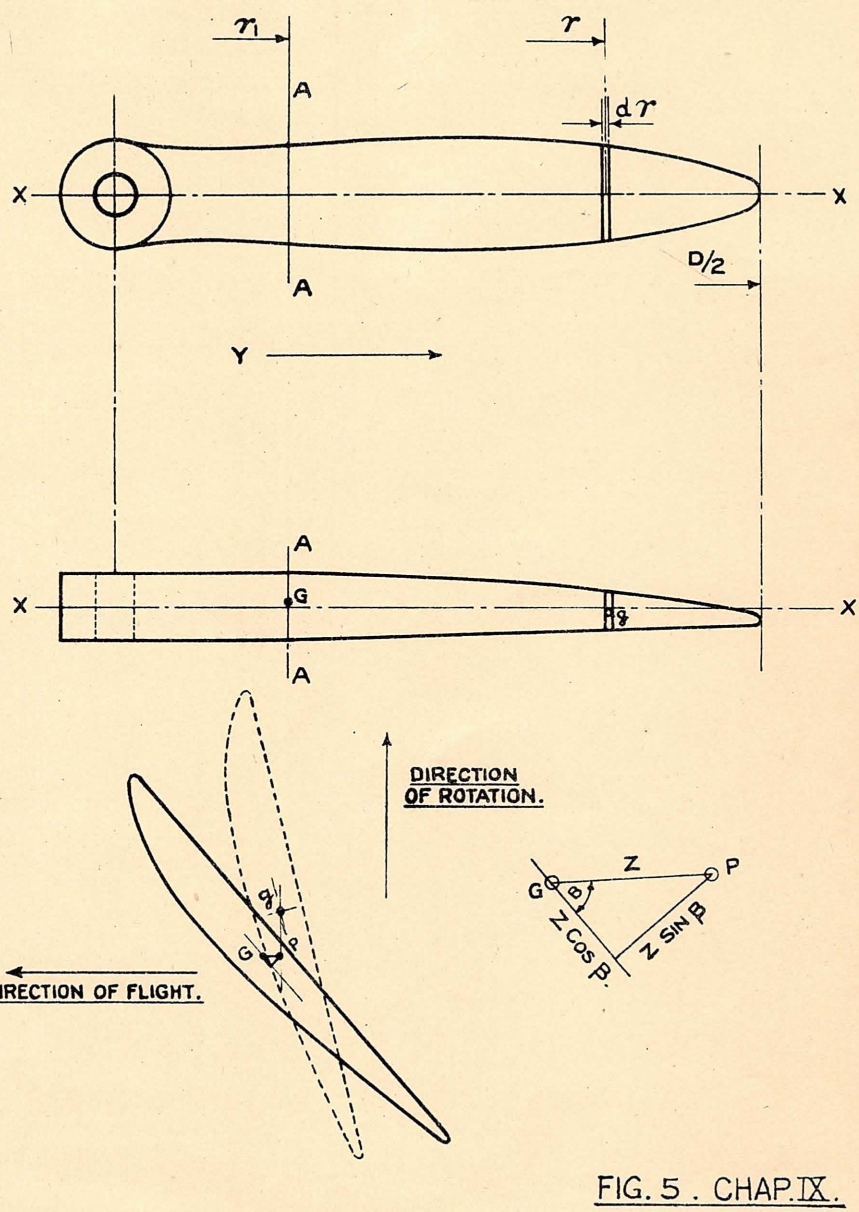


FIG. 5 . CHAP. IX.

CALCULATION OF BENDING STRESSES DUE TO CENTRIFUGAL FORCE.

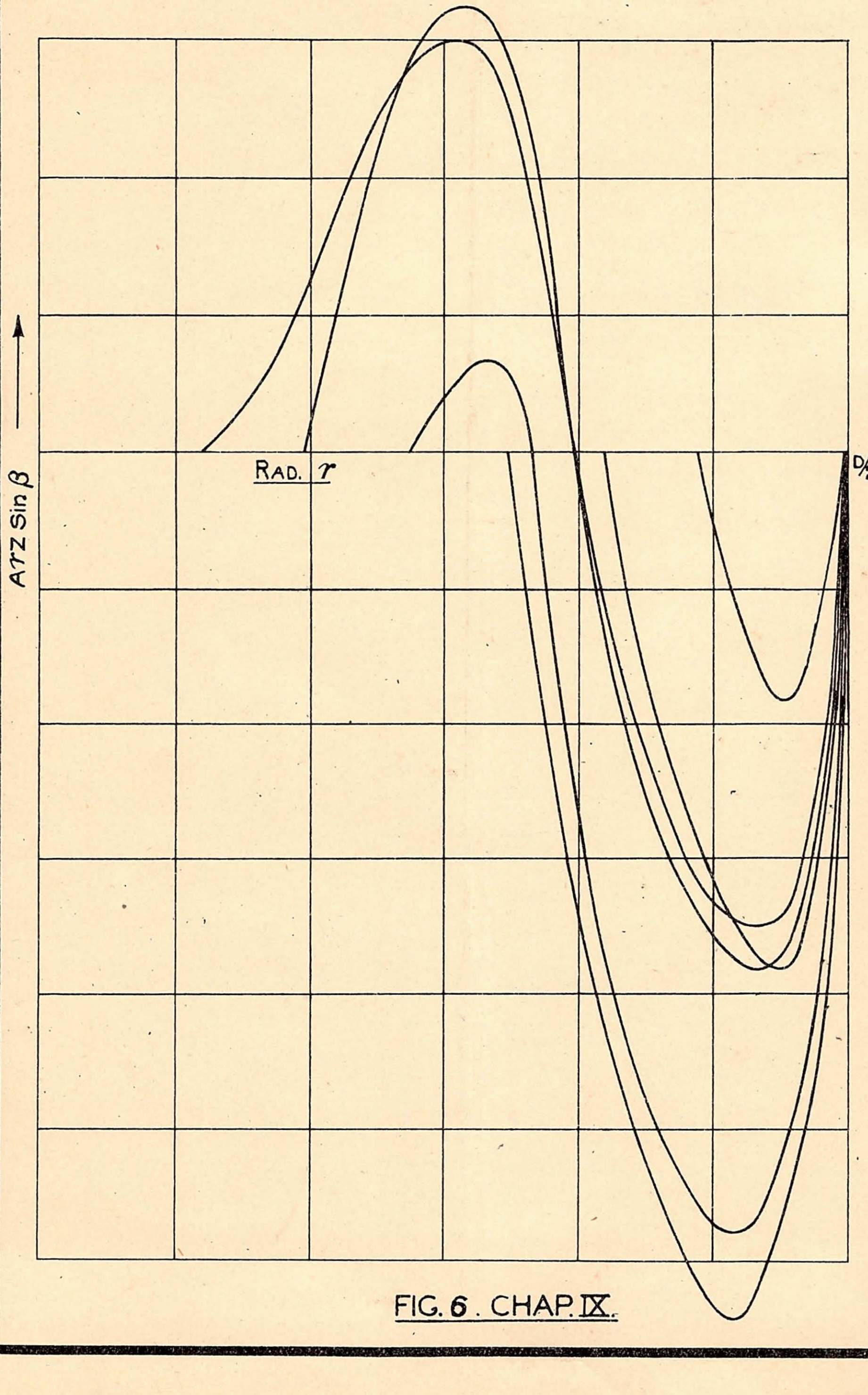


FIG. 6. CHAP. IX.

APPENDIX A TO CHAPTER IX.—AEROFOIL DATA FOR AIRSCREW DESIGN

(i) When the lift and drag coefficients of aerofoils used in airscrew design are determined from wind tunnel tests, the data must be corrected to infinite aspect ratio (rectangular wing). For any particular value of the lift coefficient, the corrections to be applied to the angle of incidence and the drag coefficient respectively are indicated in the following equations :—

$$\alpha = \alpha' - \frac{a_0 - a}{a a_0} k_L$$

$$k_D = k'_D - \frac{2}{\pi A} N k_L{}^2$$

where α and k_D refer to infinite aspect ratio and α' and k'_D to the finite aerofoil used in the wind tunnel tests of aspect ratio A.

N = factor for induced drag,

a = the slope of the lift curve for a rectangular aerofoil of aspect ratio A

and a_0 = the slope of the lift curve in two-dimensional flow.

The values of $\frac{a}{a_0}$ and N can be obtained from Table 3 of R. & M. 866 and a_0 can be assumed to equal 3·0 approximately.

(ii) Wind tunnel tests are usually made on monoplane aerofoils of aspect ratio 6, and the appropriate corrections applied at a definite value of the lift coefficient are expressed in the following equations :—

$$\alpha \text{ (degrees)} = \alpha' - 7{\cdot}15\, k_L \quad \ldots \quad (1)$$

$$k_D = k'_D - 0{\cdot}112\, k^2{}_L \quad \ldots \quad (2)$$

(iii) The following reports give wind tunnel data for a number of sections commonly employed :—

Reports and Memoranda Nos. 152, 322, 362 and 829.

The characteristics of the aerofoils of the four reports mentioned above, corrected to infinite aspect ratio (rectangular wing) by formulæ (1) and (2), are given in the curves of the accompanying diagrams, figs. 1 to 8 inclusive, which will be used to obtain values of k_L and γ (where $\gamma = \cot^{-1} \frac{k_L}{k_D}$) for various values of angle of incidence α.

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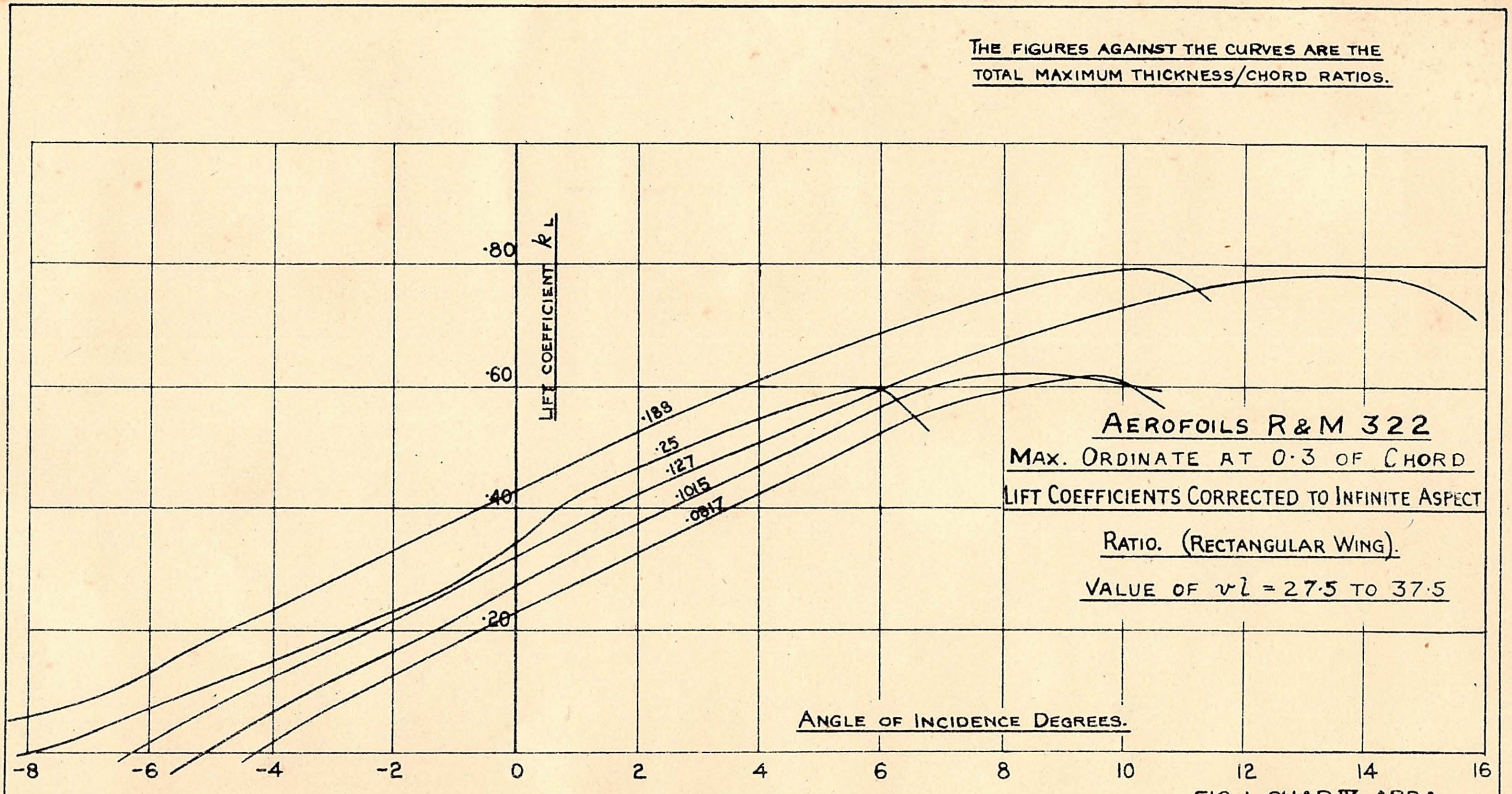

FIG. 1. CHAP. IX. APP. A.

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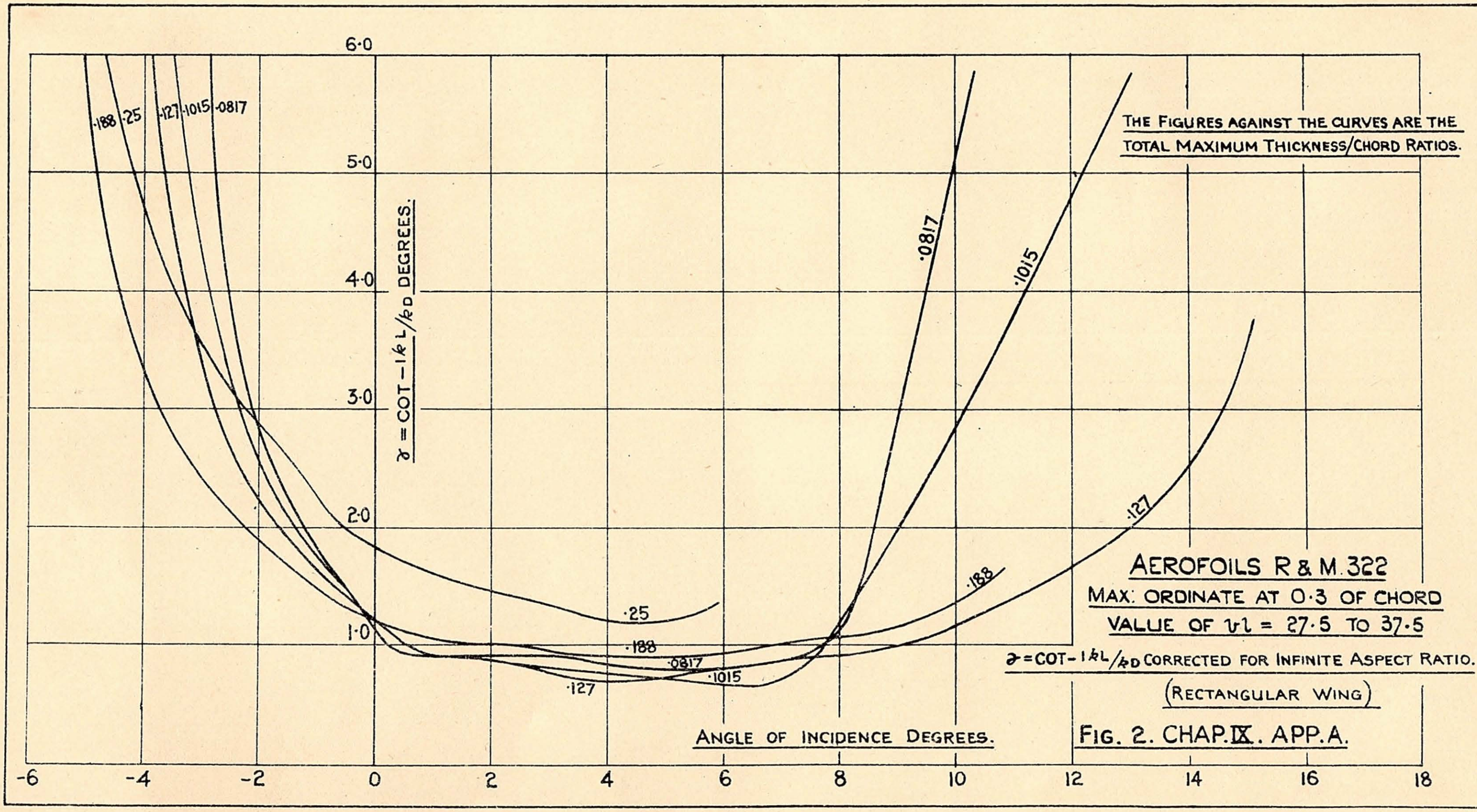


FIG. 2. CHAP. IX. APP. A.

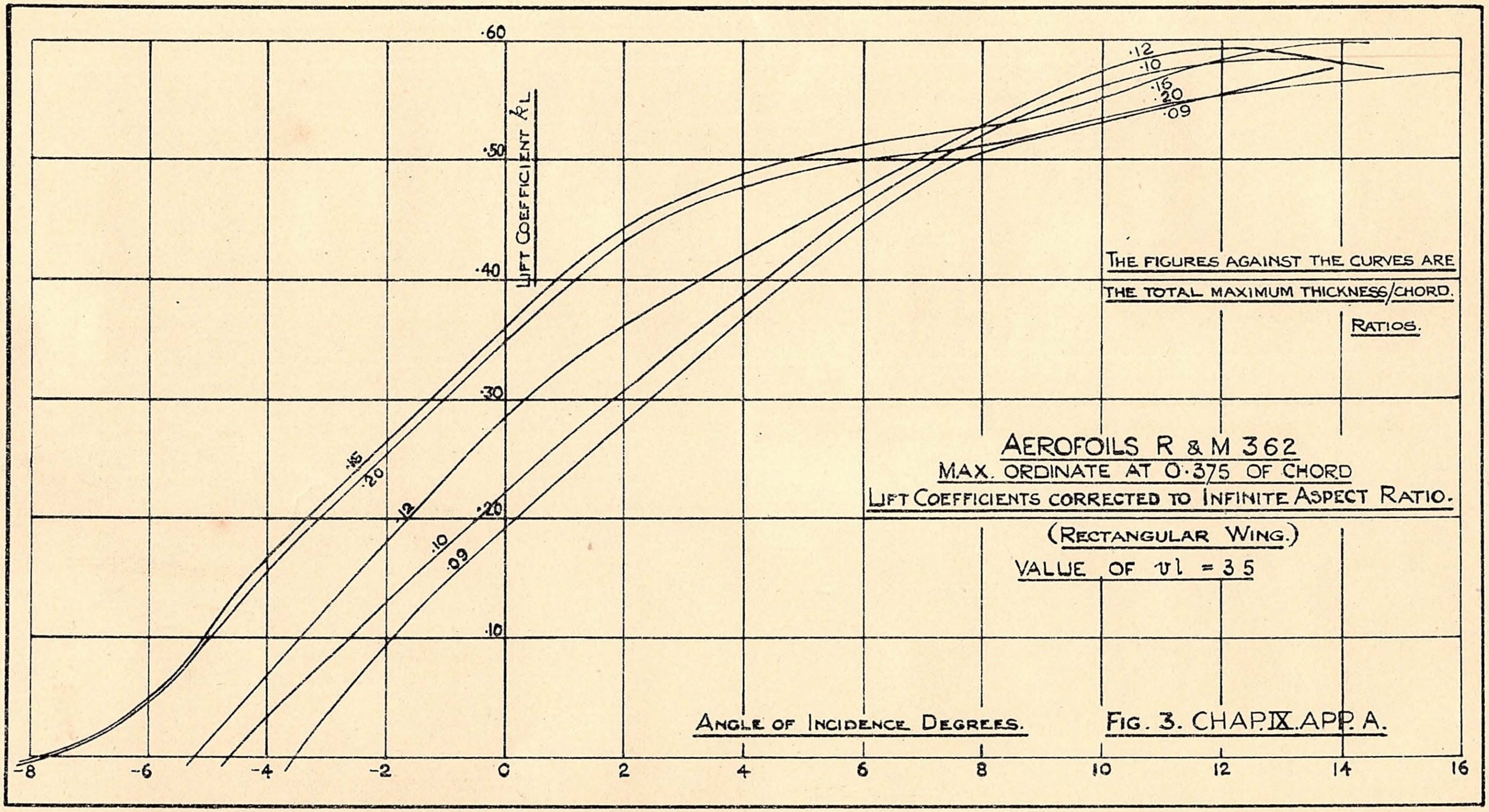
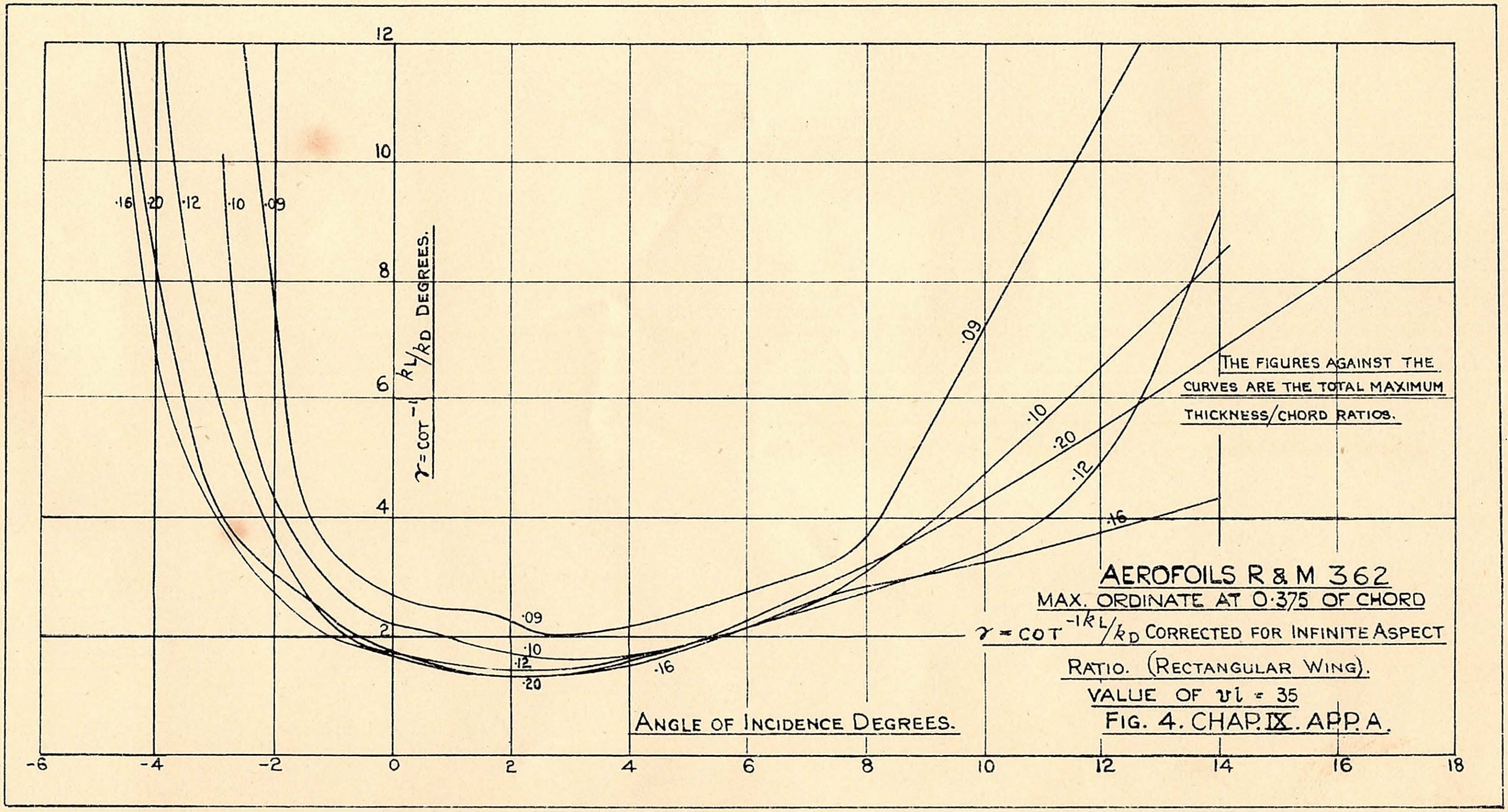


FIG. 3. CHAP. IX. APP. A.



AEROFOILS R & M 362
MAX. ORDINATE AT 0·375 OF CHORD
$\gamma = \cot^{-1} k_L/k_D$ CORRECTED FOR INFINITE ASPECT RATIO. (RECTANGULAR WING).
VALUE OF $vl = 35$
FIG. 4. CHAP. IX. APP. A.

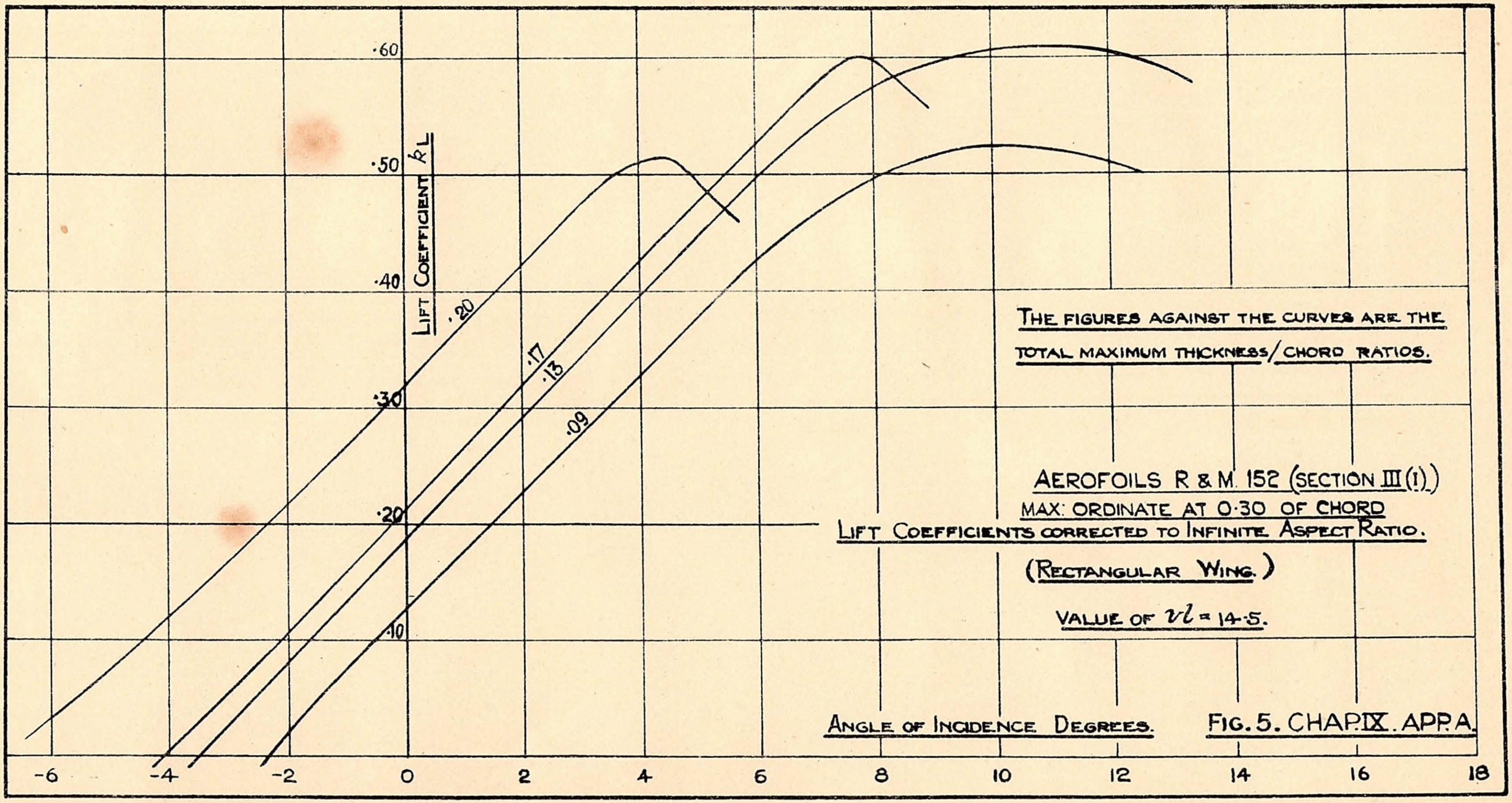
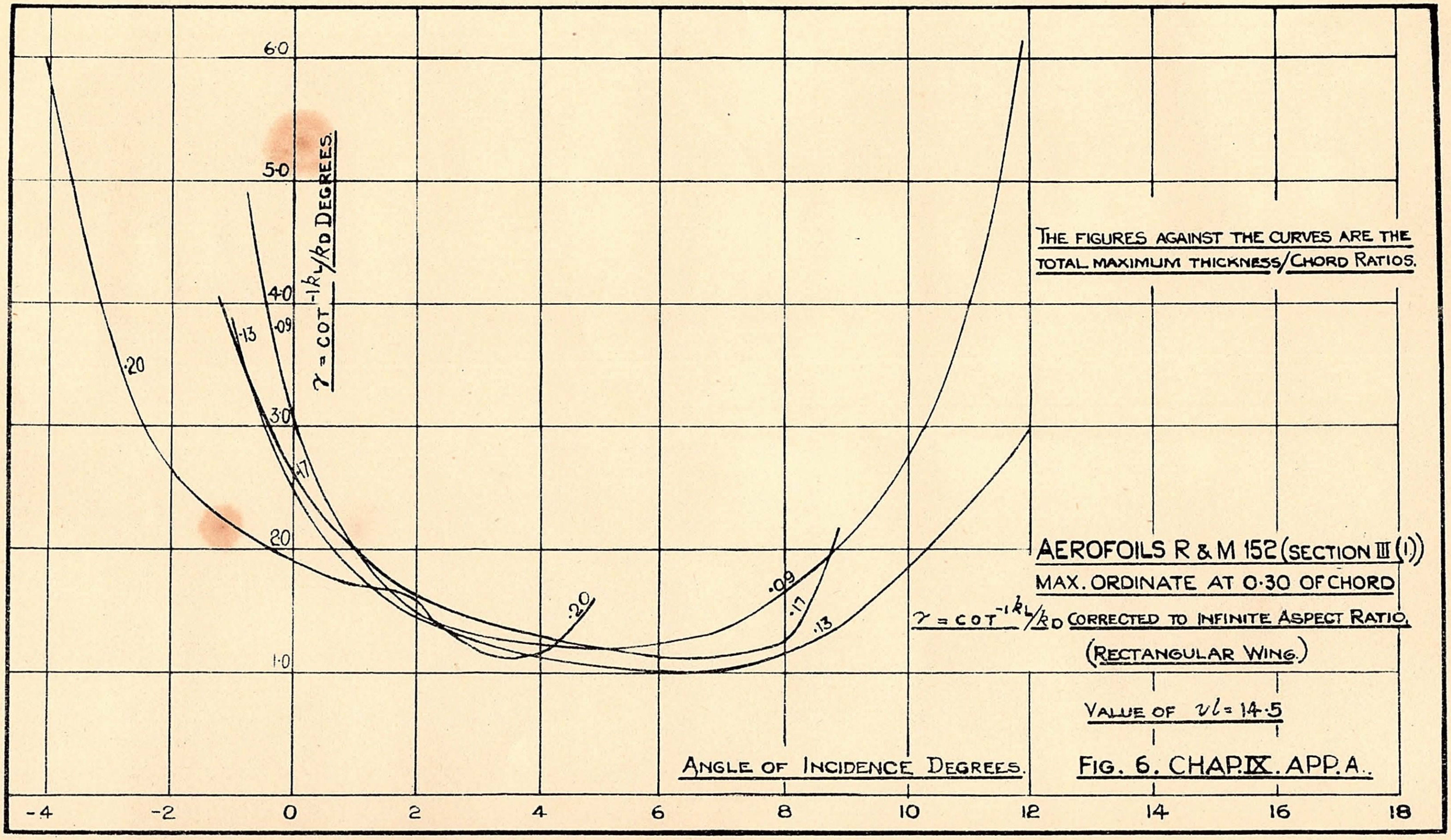


Fig. 5. Chap. IX. App. A.



AEROFOILS R & M 152 (SECTION III (1))

MAX. ORDINATE AT 0·30 OF CHORD

$\gamma = \cot^{-1} k_L/k_D$ CORRECTED TO INFINITE ASPECT RATIO.

(RECTANGULAR WING.)

VALUE OF $vl = 14·5$

FIG. 6. CHAP. IX. APP. A.

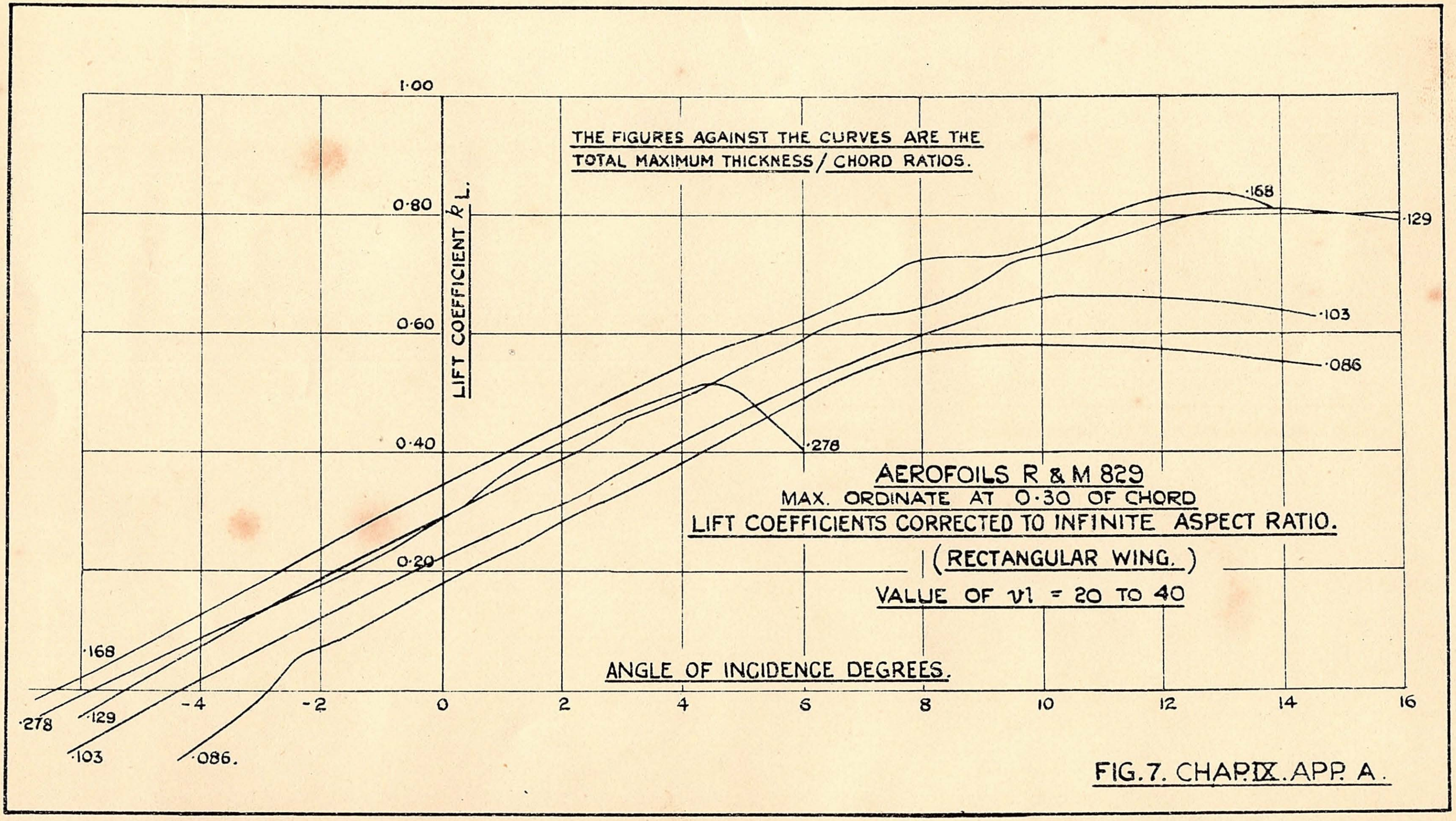


FIG. 7. CHAP. IX. APP. A.

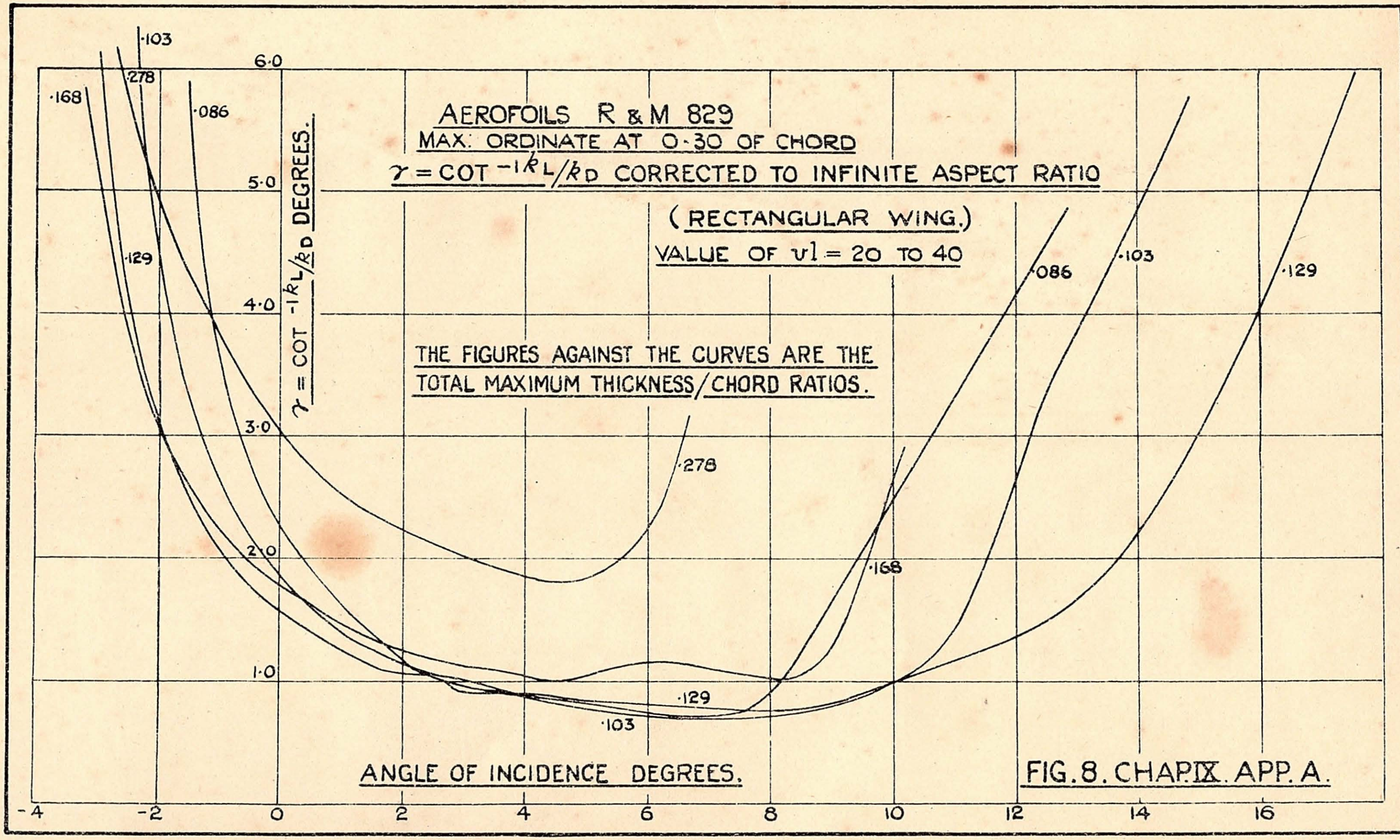


FIG. 8. CHAP. IX. APP. A.