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	$2 \cdot 2$ or $1 \cdot 5\left(\frac{T.V.}{1 \cdot 5\, V_{max.}}\right)^2$ (whichever is the greater) for experimental aeroplanes. $2 \cdot 0$ or $1 \cdot 5\left(\frac{T.V.}{1 \cdot 5 V_{max.}}\right)^2$ (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 $= \frac{550\eta H}{V \text{ (f.p.s.)}}$

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 $+ 0\cdot 1c$ *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	$0\cdot230 + 0\cdot080/C_L$	0·260	$0\cdot200 + 0\cdot080/C_L$	1·18	R. & M. 323, R. & M. 195 for contour.
R.A.F. 15	0·280	$0\cdot250 + 0\cdot032/C_L$	0·250	$0\cdot220 + 0\cdot032/C_L$	1·06	R. & Ms. 859, 872, 888, 1320, 857, 774, 816.
R.A.F. 25	0·280	$0\cdot230 + 0\cdot037/C_L$	0·250	$0\cdot200 + 0\cdot037/C_L$	0·92	R. & M. 915.
R.A.F. 26	0·305	$0\cdot250 + 0\cdot058/C_L$	0·275	$0\cdot220 + 0\cdot058/C_L$	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	$0\cdot240 + 0\cdot041/C_L$	0·260	$0\cdot210 + 0\cdot041/C_L$	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	$0\cdot230 + 0\cdot059/C_L$	0·255	$0\cdot200 + 0\cdot059/C_L$	1·26	R. & Ms. 928, 990, 1320.
R.A.F. 32	0·330	$0\cdot250 + 0\cdot135/C_L$	0·300	$0\cdot220 + 0\cdot135/C_L$	1·32	R. & Ms. 928, 1006.
R.A.F. 33	0·275	$0\cdot250 + 0\cdot014/C_L$	0·245	$0\cdot220 + 0\cdot014/C_L$	1·30	R. & M. 928.
R.A.F. 34	0·250	$0\cdot250 + 0\cdot004/C_L$	0·220	$0\cdot220 + 0\cdot004/C_L$	1·20	R. & Ms. 1071, 1146, 1635, 1706.
R.A.F. 36	0·340	$0\cdot250 + 0\cdot061/C_L$	0·310	$0\cdot220 + 0\cdot061/C_L$	—	R. & M. 1147.
R.A.F. 38	0·275	$0\cdot250 + 0\cdot036/C_L$	0·245	$0\cdot220 + 0\cdot036/C_L$	1·32	R. & Ms. 1543, 1706.
R.A.F. 48	0·275	$0\cdot230 + 0\cdot044/C_L$	0·245	$0\cdot200 + 0\cdot044/C_L$	—	R. & M. 1543, 1706.
A.D. 1	0·290	$0\cdot235 + 0\cdot062/C_L$	0·260	$0\cdot205 + 0\cdot062/C_L$	—	R. & Ms. 943, 1357.
No. 64	0·280	$0\cdot250 + 0\cdot048/C_L$	0·250	$0\cdot220 + 0\cdot048/C_L$	1·08	R. & Ms. 152, 772.
Gottingen 386	0·320	$0\cdot230 + 0\cdot098/C_L$	0·270	$0\cdot200 + 0\cdot098/C_L$	—	
Gottingen 387	0·320	$0\cdot240 + 0\cdot103/C_L$	0·290	$0\cdot210 + 0\cdot103/C_L$	—	R. & M. 1706
Gottingen 388	0·330	$0\cdot250 + 0\cdot103/C_L$	0·300	$0\cdot220 + 0\cdot103/C_L$	—	*Ergebnisse der Aerodynamischen Versuchsanstalt zu Gottingen.*
Gottingen 426	0·335	$0\cdot250 + 0\cdot095/C_L$	0·305	$0\cdot220 + 0\cdot095/C_L$	—	R. & M. 1243
Gottingen 429	0·245	$0\cdot250 + 0\cdot001/C_L$	0·215	$0\cdot220 + 0\cdot001/C_L$	—	
Gottingen 436	0·305	$0\cdot250 + 0\cdot070/C_L$	0·275	$0\cdot220 + 0\cdot070/C_L$	—	
Gottingen 449	0·335	$0\cdot235 + 0\cdot112/C_L$	0·305	$0\cdot205 + 0\cdot112/C_L$	—	
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	$0\cdot250 + 0\cdot022/C_L$	0·220	$0\cdot220 + 0\cdot022/C_L$	—	N.A.C.A. Tech. Note No. 243.
Clark Y	0·280	$0\cdot240 + 0\cdot082/C_L$	0·250	$0\cdot210 + 0\cdot082/C_L$	—	N.A.C.A. Tech. Note No. 219.
Clark YH	0·260	$0\cdot230 + 0\cdot033/C_L$	0·230	$0\cdot200 + 0\cdot033/C_L$	—	N.A.C.A. Tech. Note. No. 240, R.&M.1706.
Bristol 1a	0·265	$0\cdot250 + 0\cdot024/C_L$	0·235	$0\cdot220 + 0\cdot024/C_L$	—	Report No. B.A. 643.
B.I.R. 33a	0·315	$0\cdot240 + 0\cdot100/C_L$	0·285	$0\cdot210 + 0\cdot100/C_L$	—	R. & M. 248.
Sloane	0·295	$0\cdot240 + 0\cdot057/C_L$	0·265	$0\cdot210 + 0\cdot057/C_L$	—	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.—

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

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

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

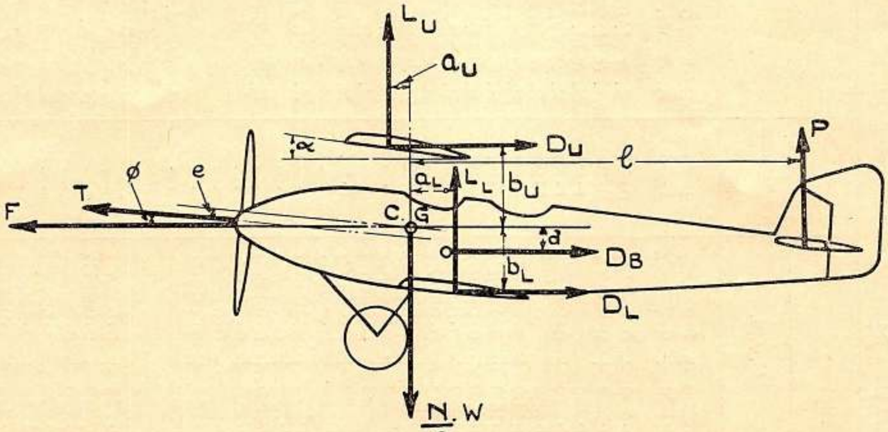

FIG. 1.—CHAP. II. C.P. Forward (engine on).

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.

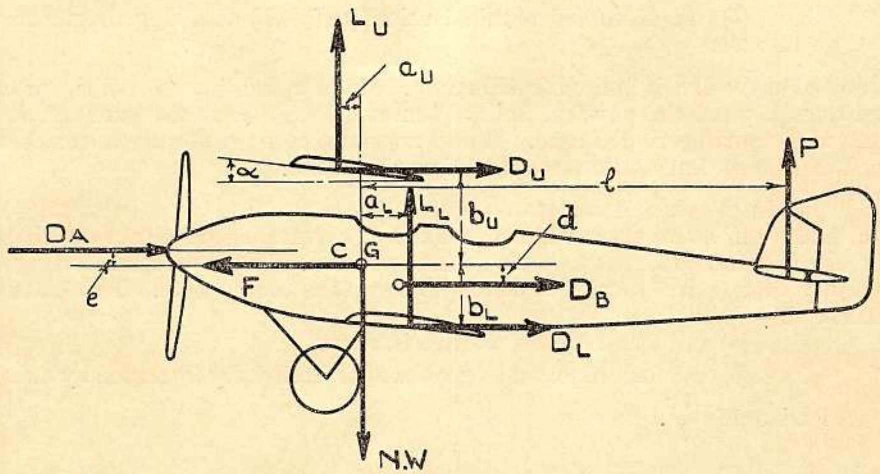

FIG. 2.—CHAP. II. C.P. Back.

(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 $0 \cdot 002378$ 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 $2 \cdot 2$ and other aeroplanes of this class a factor of $2 \cdot 0$ (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 $0 \cdot 25$ 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*) $1 \cdot 5$ 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 $2 \cdot 2$ and other aeroplanes of this class a factor of $2 \cdot 0$ (unless otherwise specified) in a fast glide at the speed defined in sub-para. (ii) below.

(ii) The speed of $1 \cdot 3$ 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 1·5 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.

* 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 $1 \cdot 5$ (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 $1 \cdot 5$ 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, $1 \cdot 5\ V_{max}$ (or $1 \cdot 3\ V_{max}$) 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 $1 \cdot 5\ V_{max}$ (or $1 \cdot 3\ V_{max}$) 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 $1 \cdot 5\ V_{max}$ (or $1 \cdot 3\ V_{max}$) 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

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

(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 $2 \cdot 0$. 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.

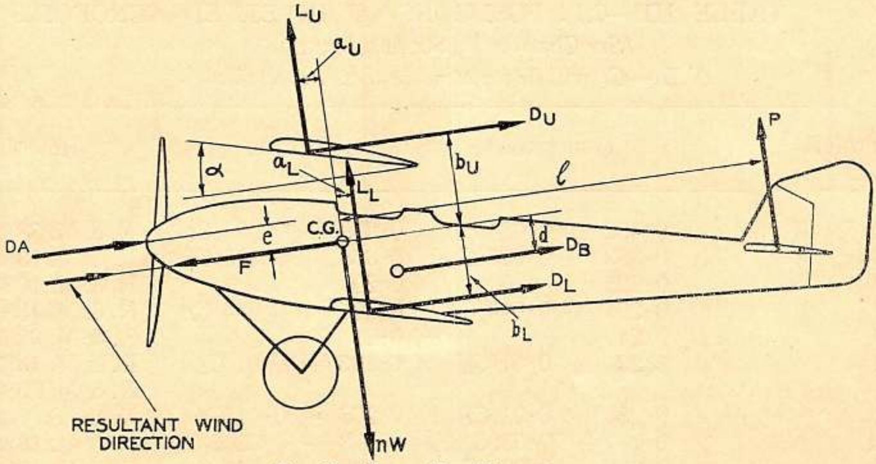

FIG. 3.—CHAP. II. Up gust case.

(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.—

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

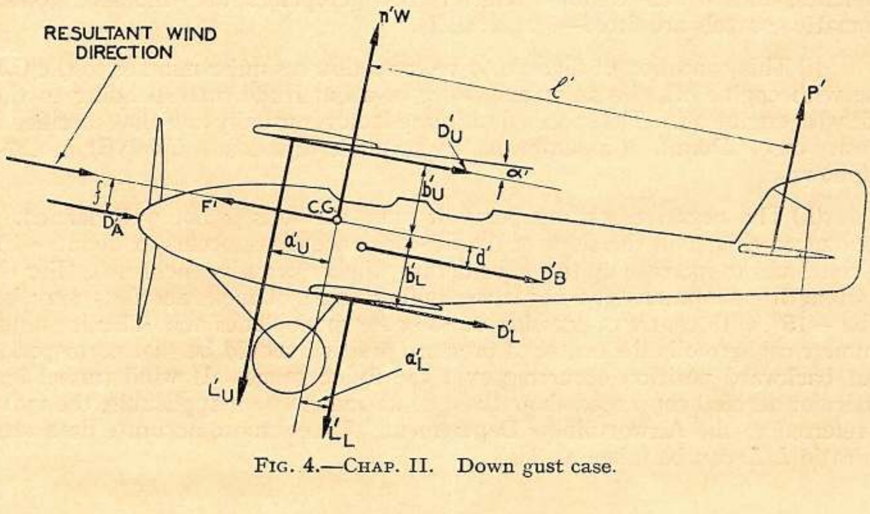

FIG. 4.—CHAP. II. Down gust case.

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

$$D'_U + D'_L + D'_B + D'_A - F' = 0 \quad \ldots \quad (5)$$
$$L'_U\, a'_U - L'_L\, a'_L + D'_L\, b'_L + D'_B\, d' + D'_A f + P'\, l' - D'_U\, b'_U = 3 \quad \ldots \quad (6)$$

(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 $V_S\sqrt{N}$, V_S being the stalling speed of the aeroplane when inverted, and N the specified ultimate factor in this high incidence inverted flight case.

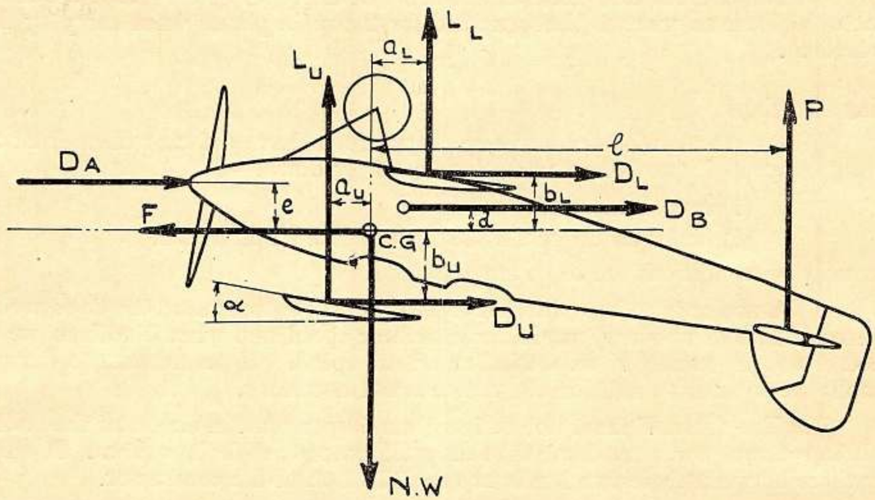

FIG. 5.—CHAP. II. Inverted flight—high negative incidence.

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 $1 \cdot 5$ 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 $2 \cdot 5$ 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 $0 \cdot 4$ 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 $\cdot 45$ 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 $1 \cdot 0$) 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 $1 \cdot 5$ 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 $0 \cdot 45$ 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 $0 \cdot 7$.

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

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 $1 \cdot 5$ 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 $c' = \text{chord of tail plane (ft.)} = \frac{\text{Area}}{\text{Span}}$ (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.—

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

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

* *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.

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

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

where V_S 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 $V_S\sqrt{\frac{N}{2}}$; 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 $V_S\sqrt{\frac{N}{2}}$.

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

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

and 2 × the gyroscopic yawing couple by

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

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 V_S 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.

* 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 1·33 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 $1 \cdot 33$ 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 $1 \cdot 5$ 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 $1/1 \cdot 5$ or $1/1 \cdot 3$ (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

$$\frac{1}{2} \times 1\cdot0 \times 0\cdot002378\ V^2{}_1 \text{ lb./sq. ft.}$$

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

$$\frac{1}{2} \times 1\cdot0 \times 0\cdot002378\ V_2^2 \text{ lb./sq. ft.}$$

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 $1\cdot0$, then this lower normal force coefficient may be taken instead of $1\cdot0$ 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 $5g$.

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.