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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

No. 480

AIRPLANE STRENGTH CALCULATIONS AND STATIC TESTS IN RUSSIA

(An Attempt at Standardization)

From L'Aéronautique, February, 1928

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS.

TECHNICAL MEMORANDUM NO. 480.

AIRPLANE STRENGTH CALCULATIONS AND STATIC TESTS IN RUSSIA.*

(An Attempt at Standardization)

We are here giving a summary of the rules established by the Theoretical Section of the Central Aerodynamic Institute of Moscow for the different calculation cases of an airplane. These rules, which have been adopted by the "Direction des Forces aeriennes militaires" of the U.R.S.S. (Union of Socialist Soviet Republics, i.e., Russia), have been in force since August 1, 1927. It will be interesting to compare them with the testing conditions required in France for the three cases of flight. It appears that the engineers of the Aerodynamic Institute considered only thick or medium profiles. For these profiles they have attempted to increase the safety when the center of pressure moves appreciably toward the trailing edge. Will such a standardization of the values of the overloads and their distribution lead to the construction of better cells with the maximum strength for the minimum wing loading? We would hesitate to affirm it, however, for investigations (like that of present pursuit airplanes), which are only typical cases. Nevertheless, this attempt at standardization is not without interest and at least has the merit of being presented in the form of concrete results.

*"Calculs de resistance des avions et essais statiques en Russie," L'Aeronautique, February, 1928, pp. 42-45.

Definitions

The <u>overload in flight</u> is defined in each instance by the ratio of the load supported by the airplane parts involved in the motion considered to the load on the same parts in uniform horizontal flight for the same angle of attack. These overloads are determined experimentally with accelerometers and also by calculation.

The <u>landing overload</u> is defined by the ratio of the load on the parts of the airplane just making contact with the ground to the load on the same parts when the airplane rests on the ground.

The <u>safety factor</u> is the ratio of the breaking load to the maximum load of elastic deformation for the given flight case at a given angle of attack. It includes the ratio of the breaking stress to the limit of elasticity and the coefficient of exploitation, which takes into account the length of service, conditions of use, etc.

From the definitions of the overload in flight and of the safety factor, it follows that the <u>static overload</u> is the ratio of the breaking load to the fatigue in uniform horizontal flight in both cases for the same angle of attack.

The <u>calculated overload</u> is the ratio of the calculated breaking load to the load in uniform horizontal flight.

The ratio of the static to the calculated overload is the perfection coefficient of the calculation. Its value indicates to the constructor the more or less precise degree of approximation to the actual fatigue conditions of the airplane.

For the calculation and the static airplane tests, the flight cases in which the most important parts support the greatest stresses have been considered, each being distinguished by a letter with a subindex for each part. The following classificatiom has been adopted.

Class 1. Commercial Airplanes

Group 1.	Full load	not exceeding	2500 kg (5512 lb.)
" 2.	"	"	2500 - 5000 kg (11023 lb.)
" 3.	"	"	5000 - 10000 " (22046 ")
" 4.	"	"	over 10000 kg

Class II. Military Airplanes

Group 12.	Single-seat	land	pursuit airplanes.*
" 11.	"	"	marine pursuit airplanes.
" "	"	"	training airplanes.
" 10.	Two-seat	land	pursuit airplanes.
" 9.	"	"	marine pursuit airplanes.
" "	"	"	training airplanes.
" 8.	Army observation airplanes.**		

*The numbers for Class II are equal to the static overload applied to the wings, as prescribed for flight case A.

**If an army observation airplane is used as a light bomber, it must satisfy the requirements of group 6.

Group	8.	Combat airplanes.
"	"	School "
"	7.	Marine observation airplanes.
"	"	Army Corps observation airplanes.
"	"	Torpedo airplanes.
"	"	Marine school airplanes.
"	6	Light torpedo and bombing airplanes.
"	5	Large bombing airplanes.
"	4	Bombers weighing over 10000 kg (22046 lb.).

I. The Wings (subindex k)

Case A_k.- Coming out of a dive at the angle of maximum lift. We take the resultant in the position corresponding to this angle according to the C_m, the inclination of the resultant to the chord being 98°. For the distribution of the load, see the following paragraphs II, III, and IV. The load itself has the form

$$P = n\ (\Pi - \Pi_{kp}) - \Pi_{kp}$$

in which n is the static overload given in the accompanying table; Π, the full load; Π_{kp}, the weight of the cell.

Case B_k.- Coming out of a dive to glide at the maximum angle of attack. The overload is the same as in the preceding case. the resultant being applied at 1/3 the distance from the trailing edge and inclined 1/3.

Case C_k.- Diving flight. The lift is assumed to be zero. The wings are subjected to a load which produces a moment of torsion and a drag.

Moment of torsion

$$M = f \frac{C_m}{C_{x_C} + C_{x_B}} \Pi \; L.$$

Drag

$$X = f \frac{C_{x_{kp}}}{C_{x_C} + C_{x_B}} \Pi$$

f, safety factor given in table;

C_m, coefficient of moment;

$C_{x_{kp}}$, " " drag of the wings;

C_{x_C}, " " " " " airplane;

C_{x_B}, " " " " " propeller;

$L = S/A$, chord of wing.

C_m and $C_{x_{kp}}$ are taken on the polar for $C_y = 0$. Moreover,

$$C_{x_B} = C_p \frac{F}{S} a$$

with

$$F = \frac{\Pi}{4}(D^2 - d^2),$$

surface swept by propeller;

S, area of wings;

$$a = \frac{Z \; b_{cp}}{D},$$

mean width of propeller blades (Z, number of blades; b_{cp}, mean width of one blade).

C_p, coefficient of the negative traction of the propeller equal to

$$0.26 \text{ for } h = 0.9 \quad \left(h = \frac{H}{D}\right)$$
$$0.24 \text{ " } h = 0.5$$

with

$$C_p = 0.26 + 0.2\,(0.9 - h)$$

as intermediate values. In the plane of the wing, the static load has the form

$$Q = X - f\,\Pi_{kp} \pm \Pi_{kp}$$

+, if the wing is fixed during the tests with the leading edge down; –, if the wing is fixed during the tests with the leading edge up; f, safety factor given in table.

Case D_k.– Curvilinear flight in inverted position. The resultant is taken at 1/4 of the chord from the leading edge and inclined –1/4. The overload is the same as in case A_k.

Case E_k.– Sudden landing. The resultant passes through the center of gravity of the wing section and the overload is given by

$$R = n\,\Pi_{kp}$$

n being given in the table.

II. Distribution of the Load Along the Wing Chord

Case A_k.– Load distributed according to Figure 1, or a loading diagram (derived from Fig. 12) is plotted by observing the condition of passage of the resultant through the center of lift. The latter method (Fig. 2) is used when

$$l \leqq 0.3125 \text{ L.}$$

Case B_k.– Uniform distribution on the half-chord, the resultant passing at 1/3 of the chord from the trailing edge (Fig. 3).

Cases C_k and E_k.– The distribution of the load depends on the way the tests are made.

Case D_k.– Figure 4.

III. Distribution of the Load Along the Span

The following rules give the distribution for the wings shown in Figure 5, with the same angle of setting of the same profile throughout the span.

Case A_k.– For elliptical wings, distribution proportional to the chords.

For trapezoidal wings, the ratio of the extreme chords being comprised between 0.4 and 0.7 and the aspect ratio between 5 and 8, we have (Fig. 6)

$$h_1 = p' \times C_1$$

$$h_2 = p' \frac{C_1 + \lambda C_2}{0.5 + \lambda}$$

with

p', calculated wing loading ($p' = \Pi_1/S$; Π_1, full load – weight of wings);

λ, aspect ratio $= l^2/S$;

C_1, chord at point of imbedding;

C_2, chord at wing tip;

$C_0 = S/l$, mean chord.

For the wing types a, b, c, d (Fig. 5), the load is distributed according to Figure 7 with

$$h = p' \frac{l}{\lambda - 0.25} .$$

Cases B_k and D_k.- For elliptical wings, distribution proportional to the chords.

For trapezoidal wings ($C_2/C_1 = 0.4 - 0.7$; $\lambda = 5 - 8$), distribution similar to Figure 6, the height of the trapezoid of the extremity being C_0 instead of $C_0/2$ with

$$h_1 = p' C_1,$$

$$h_2 = p' \frac{2 C_1 + C_2 \lambda}{1 + \lambda} .$$

For the wing shapes a, b, c, d distribution according to Figure 8, with

$$h = p' \frac{l}{\lambda - 0.5} .$$

Case C_k.- Distribution proportional to the chords.

Case E_k.- Distribution corresponding to the masses loading the wing and to the weight of the wing itself.

IV. Distribution of the Load between the Wings*

The angle of stagger is the angle with the vertical formed by the straight line joining the two points on the upper and lower chords at one-third their length from the leading edge.

The load is distributed following the angle of stagger according to Figure 9 for case A_k, Figure 10 for case B_k, Figure 11 for case E_k. On the abscissas, angles of stagger; on the ordinates, quotient of the upper C_y divided by the lower C_y.

In case C_k, the distribution is proportional to the area of the wings. In case E_k, it is proportional to the weight of the wings and to the masses.

V. The Ribs

The ribs were tested for all four cases: A_n, B_n, C_n and D_n.

Case A_n (Fig. 12):

$$\alpha = \frac{0.385 - 0.875\,\mu}{0.225\,\mu - 0.0338}$$

$$H = \frac{\Sigma}{C\,(0.225\alpha + 0.875)}$$

with

μ, distance between the center of pressure and the leading edge in % of the chord;

*These rules apply only to biplanes whose wings have the same setting.

C, chord;

Σ, load area corresponding to the load on a rib in case A, all the load being distributed thus, namely, extrados : intrados = 2 : 1.

Case B_n (Fig. 13).- The load areas are Σ_1 and Σ_2 such that $\Sigma_1/\Sigma_2 = 3$, referred to the upper part $\Sigma_1 + 0.4\ \Sigma_2$. Referred to the lower part $0.6\ \Sigma_2$, $\Sigma_1 - \Sigma_2$ corresponds to the load on a rib.

Case C_n (Fig. 14).- In this case $\Sigma_1 = \Sigma_2$: on the extrados, $\Sigma_1 + 0.3\ \Sigma_2$; on the intrados, $0.7\ \Sigma_2$.

Case D_n (Fig. 15).- On the extrados, Σ_2; on the intrados, Σ_1.

VI. The Wings

These were tested only for case B. Mean aileron loading

$$\bar{\omega}_a = 0.0525\ v^2_{max}$$

(v_{max}, maximum speed in m/s in horizontal flight near the ground) with

$$(\bar{\omega}_a)_{min} = 125\ kg/m^2.$$

The pressure is assumed to be constant along the hinge and to decrease uniformly along the rib down to a third of its value (Fig. 16).

VII. The Empennage

A.- Horizontal empennage.

Case C_h.- Static load.

$$P = k \frac{C'm}{C_{X_C} + C_{X_B}} \Pi \frac{C}{d} .$$

k, safety factor given in table;

d, distance between center of gravity of airplane and center of lift of empennage;

C'_m, coefficient of moment of airplane without empennage.

Distribution according to Figure 17.

Case K_h.-

$$P = 2 f_h \rho \Sigma_{ch} (1.4 v_{at})^2,$$

Σ_{ch}, area of horizontal empennage in m²;

f_h, coefficient of ascensional force given in table;

v_{at}, landing speed in m/s;

$\rho = 1/8$.

These two loads are compared and the greater one is taken for the tests.

B.- Vertical empennage

$$P = 2 f_h \rho \Sigma_{lv} (1.4 v_{at})^2,$$

distributed according to Figure 18.

Σ_{lv}, area of vertical empennage in m².

VIII. Landing Gear

In all the cases, the overload is

$$P = n\Pi$$

n being the coefficient given in the table.

IX. Fuselage and Engine Bed

Case E_{fb}:

$$P = n\Pi$$

n being given in the table.

Case C_{fb}.- The test is made for the rear part of the fuselage (diving), with a load corresponding to that on the horizontal empennage (case C_h).

Case $(H_{fb})_{AR}$.- The lateral load on the stern is given by the load on the vertical empennage (case K_h).

Case $(H_{fb})_{AV}$:

$$P = n\Pi_{AV}$$

Π_{AV} being the weight of the front part (see case A_{fb}).

Case A_{fb}.- Flight with a vertical acceleration corresponding to case A:

$$P = n\,\Pi_{AV}.$$

(From Technique de la Flotte Aérienne Russe, 1927, No. 1.)

T A B L E

Class	Group	Wings					Ailerons	Ribs			
		A_k	B_k	C_k	D_k	E_k	B_a	A_n	B_n	C_n	D_n
		n	n	f	n	n		n	n	f	n
I.	1	5.5	4	1.25	"	$n = 2 + \frac{V_{at}(km/h)}{20}$ (with $n \geqq 6$) *	See paragraph VI.	5.5	4	1.25	
	2	$\frac{5.5}{5}$	$\frac{4}{3.5}$	1.25	"			$\frac{5.5}{5}$	$\frac{4}{3.5}$	1.25	
	3	$\frac{5}{4}$	$\frac{3.5}{3}$	1.25	"			$\frac{5}{4}$	$\frac{3.5}{3}$	1.25	
	4	4	3	"	"			4	3	"	
II.	12	12	7.0	2.0	4.0			12	7.0	2.0	4
	11	11	6.5	1.9	3.75			11	6.5	1.9	3.75
	10	10	6.0	1.8	3.5			10	6.0	1.8	3.5
	9	9	5.5	1.75	3.25			9	5.5	1.75	3.25
	8	8	5.0	1.7	3.0			8	5.0	1.7	3.0
	7	7	4.5	1.5	2.5			7	4.5	1.5	2.5
	6	6	4.0	1.4	2.0			6	4.0	1.4	2.0
	5	5	3.25	1.25	"			5	3.25	1.25	"
	4	4	3.0	"	"			4	3.0	"	"

*If V_{at} equals or exceeds 120, the overload is given in each case by the "Commission Scientifique."

Table (Cont.)

Class	Group	Empennages: Horizontal C_h / k	Empennages: Horizontal K_h / f_h	Empennages: Vertical K_v / f_h	Landing gear E_a / n	Landing gear F_a / n	Landing gear G_a / n	Fuselage and engine bed E_{fb} / n	Fuselage and engine bed C_{fb} / n	Fuselage and engine bed $(H_{fb})_{AV}$	Fuselage and engine bed $(H_{fb})_{AR}$
I.	1	1.5									
	2	1.4									
	3	1.35	0.4	0.4			0.4	–			3
	4	"									
II.	12	2.25		0.6							
	11	2.15		0.6							
	10	2.10		0.6							
	9	2.0		0.6							
	8	1.90	0.4	0.6	$n = 2 + \frac{V_{at}}{20}$ $(n \geq 6)$*	$n = \frac{V_{at}}{100}$ $(n \geq 0.8)$*	0.4	$n = 3 + \frac{V_{at}}{20}$ $(n \geq 6)$*	See horizontal empennage	See vertical empennage	4
	7	1.70		0.5							
	6	1.50		0.5							
	5	1.30		0.4							
	4	"		0.4							

*If V_{at} equals or exceeds 120, the overload is given in each case by the "Commission Scientifique."

P. G.

Translation by Dwight M. Miner,
National Advisory Committee
for Aeronautics.

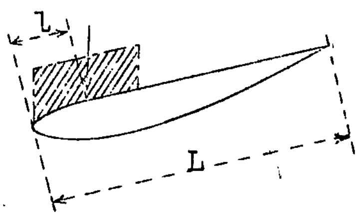


Fig.1

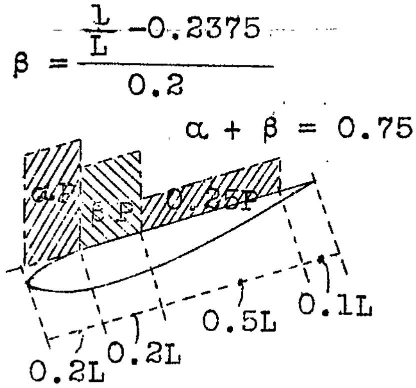


Fig.2

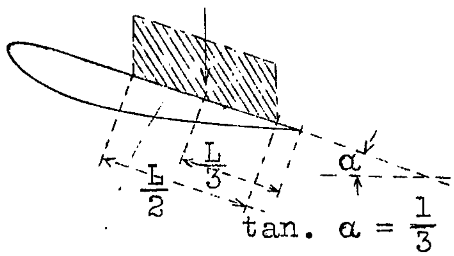
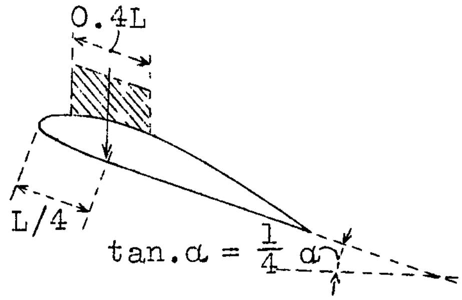


Fig.3



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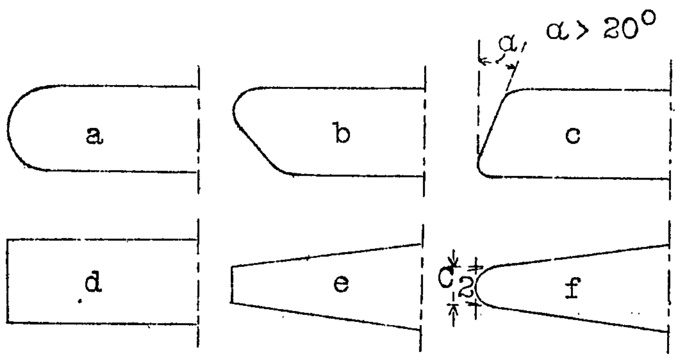

Fig.5

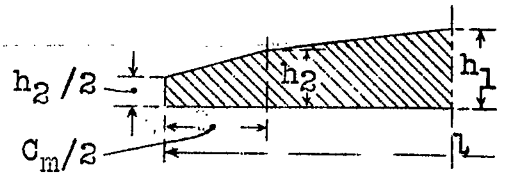

Fig.6

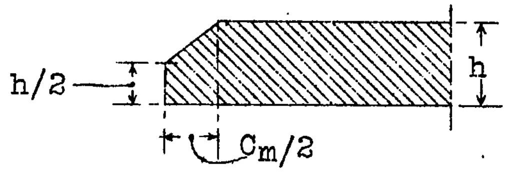

Fig.7

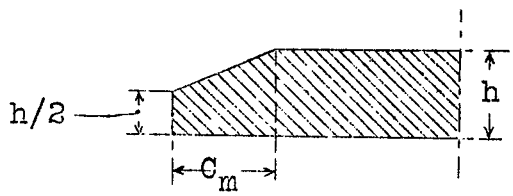

Fig.8

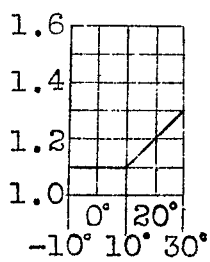

Fig.9

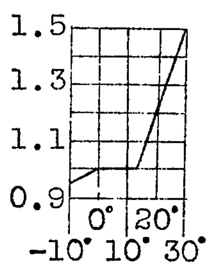

Fig.10

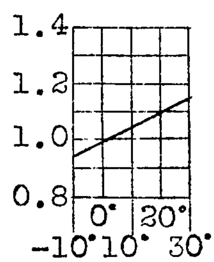

Fig.11

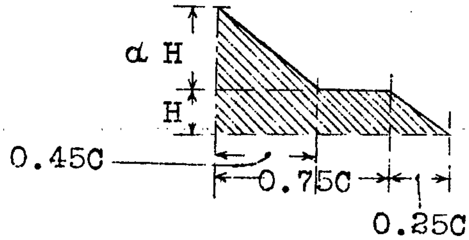

Fig.12

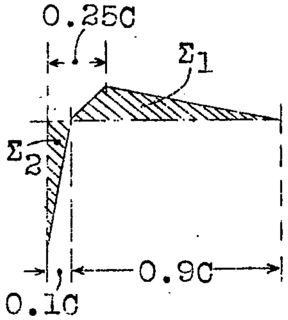

Fig.13

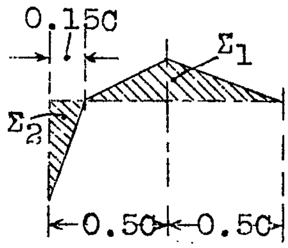

Fig.14

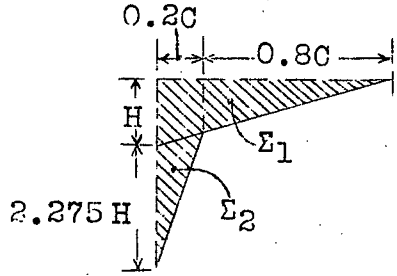

Fig.15

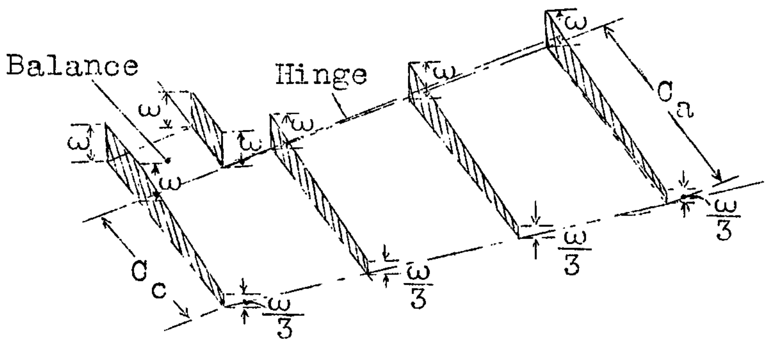

Fig.16

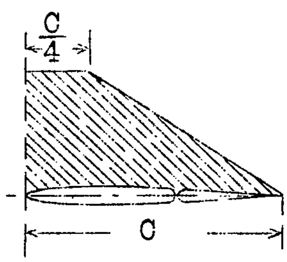

Fig.17

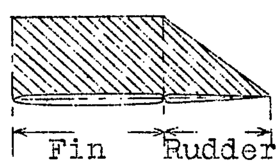

Fig.18

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