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RESEARCHES ON AILERONS AND ESPECIALLY ON THE TEST LOADS TO WHICH THEY SHOULD BE SUBJECTED

By J. Sabatier

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RESEARCHES ON AILERONS AND ESPECIALLY ON THE TEST LOADS TO WHICH THEY SHOULD BE SUBJECTED.*

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Aileron calculations have hitherto given greatly differing results according to the different authors. It seems to be the general opinion that it is only necessary to give the ailerons such dimensions that the airplane can maneuver well, that the stresses they must undergo are relatively small, and that they are strong enough if their framework is of the same order of strength as the wings to which they are attached. This article will show that the problem is really quite complex and that it should receive more attention.

What are the strength requirements for ailerons in the different countries?

In France the ailerons are included in the static tests of the wing and as if they formed an integral portion of it. Moreover, the technical specifications of 1925 stipulate that the ailerons must be able to support, before breaking, a load of at least 200 kg/m² (41 lb./sq.ft.) distributed triangularly, the base of the triangle being at the leading edge.

*"Recherches sur les ailerons et notamment sur les charges d'épreuve auxquelles ils doivent etre soumis." From "La Technique Aéronautique" for November 15 and December 15, 1926.

The German specifications of 1918 and the Italian ones of 1924 require constant loads of respectively, 200 and 250 kg/m² (41 and 51 lb./sq.ft.).

The American military specifications of June 22, 1926, require breaking-test loads of 73-170 kg/m² (15-35 lb./sq.ft.), according to the type of airplane, the heaviest loads being for the lightest airplanes. The requirements of the U. S. Navy (July 25, 1926) are similar, though less severe, since they scale the loads from 73 to 145 kg/m² (15 to 30 lb./sq.ft.).

Most of the above specifications do not directly involve the aerodynamic characteristics of the airplane, notably its speed. In England, however, an attempt has been made to take account of these characteristics by adopting, for the aileron test-load, an expression of the form $\lambda\, c_z\, V^2{}_s$, in which λ is the numerical factor, c_z the maximum lift of the wing, and V_s the minimum sustentation speed of the airplane.

One can understand why V_s comes into the expression, since it is, in fact, at its minimum speed that an airplane has the greatest need of its means of control and requires the maximum action of its ailerons. It must be remembered, however, that the stresses exerted by the air on the ailerons are not necessarily maximum, since these stresses also vary as the square of the speed of the airplane. The factor λ must therefore be chosen so as to take account of this condition.

Lastly, attempts have been made to determine the strength of the ailerons and of their controls by the maximum effort which the pilot must exert on the steering wheel or control stick. For this reason certain Belgian contracts stipulate that the aileron controls must be able to withstand a force of 160 kg (353 lb.) exerted on the control stick. Although the effort of the pilot is an important consideration, it is obviously not sufficient to determine completely the strength of the ailerons, especially if they are balanced, because the stress they then exert on their controls is only a fraction of the stresses actually supported by their surfaces.

In short, the rules now followed in different countries differ greatly from one another both in principle and in results. The following table facilitates the comparison of the test loads (in kg/m²), as deduced from these rules, for various types of airplanes.

	Germany	U.S.	Italy	Load resulting from a force of 160 kg on control stick
Nieuport 29	200	170	250	495 (ailerons not balanced)
Potez 15	200	122	250	180 (ailerons balanced)
Breguet 19B2	200	122	250	310 (ailerons not balanced)
Goliath	200	98	250	125 (" " "
Farman Bn 4	200	73	250	95 (" " ")

The loads deduced from the force exerted on the control stick were calculated on the assumption that the action of the air on the aileron is distributed triangularly, beginning at the leading edge. The value indicated for the balanced ailerons of the Potez 15 is only approximate, due to the relative uncertainty of the hypotheses concerning the division of the pressure between the portion of the aileron located behind the wing and the balancing portion located outside the wing.

In view of the above divergencies, it is desirable to take up the problem again at the beginning and to investigate experimentally the stresses to which the ailerons are exposed in flight. In order to simplify this investigation, we will only consider the case of rectangular ailerons located entirely behind the wings (that is, not balanced). For balanced ailerons with a portion of their surface outside the wing, it would be necessary to investigate separately the action of the air on the portion behind the wing and on the outside portion which the air attacks more directly. When the ailerons are in the neutral position, i.e., when they simply form a normal prolongation of the wing, they may be considered as an integral portion of the wing. The stresses they support are then the same as the air exerts on the trailing portion of the wing.

The present rules of the C.I.N.A. (International Committee on Aerial Navigation) for the calculation and the static testing of airplane cells provide for the four principal and for various accessory cases.* Of these cases the most interesting from our present viewpoint are:

* The 1925 French rules for general technical conditions are similar, the values of the factors f being only a little higher, which fact does not otherwise affect the sense of the resulting conclusions.

Flight at the extreme forward position of the center of lift;

Horizontal flight at high speed;

Vertical dive at the speed limit.

The mean factors by which the stresses corresponding to each of these three cases must be multiplied, in order to obtain the test loads which the wings of normal airplanes must withstand before breaking, are as follows:

Case 1, $f_1 = 6$;

Case 2, $f_2 = \frac{3}{4} f_1 = 4.5$;

Case 3, $f_3 = 1.5$.

A certain angle of flight and a special distribution of the air pressures along the chord correspond to each of these three cases for any given wing profile. The test loads must be distributed in conformity with this plan, both on the wing and on the aileron.

For horizontal flight, if we designate the lift of the wing and that of the aileron respectively, by C_z and c_z and if the corresponding load of the cell is P/S, it is easily seen that the breaking strength will be given by an expression of the form $q = \frac{P}{S} f \times \frac{c_z}{C_z}$.

In order to investigate the variations of this expression, the S.T.Aé. (French Technical Section of Aeronautics) recently determined, in the wind tunnel of the Eiffel Laboratory, the aerodynamic-pressure curves of various wing profiles commonly employed on airplanes.

The following results were obtained:

Angle of attack with reference to zero-lift	Angle of attack with reference to chord tan. to lower side of wing	100 C_z (wing)	100 C_z (aileron at 40%)	f	f $\frac{c_z}{C_z}$
Profile 430					
degrees	degrees				
15	6.9	108	43	6	2.4
9	1.1	67.5	48	4.5	3.2
6	-1.9	45	43	4.5	4.3
0.4	-7.7	3	34	1.5	-
Profile 389					
15	10	102	38	6	2.3
9	4.2	63	33	4.5	2.3
6	1.2	42.5	28	4.5	3
0.4	-4.6	3	16	1.5	-
Profile 387					
15	8.8	109	49	6	2.7
9	2	63	35	4.5	2.5
6	-1	42	32	4.5	3.4
0.4	-6.8	2.5	21	1.5	-
Profile 382					
15	5.6	95	30	6	1.9
9	-0.2	61	28.5	4.5	2.1
6	-3.2	39	26	4.5	3
0.4	-9	1.5	25	1.5	-
Breguet Profile					
15	11.3	99.5	33	6	2
9	5.5	64	29	4.5	2
6	2.5	44	27	4.5	2.7
0.4	-3.3	2	16	1.5	-
Halbronn Profile					
15	11.5	99	25	6	1.5
9	5.7	69	30	4.5	1.9
6	2.7	48	22	4.5	2.6
0.4	-3.1	6.2	16	1.5	-

It is seen that the expression $f \frac{c_z}{C_z}$ is about 50% larger for the case of flight at high speed than for flight at large angles of attack. The former consideration is therefore the one to be taken into account in determining the test load for the ailerons.

If we pass to the case of diving with small lift, we find that the corresponding load factor is 1.5, as against 4.5 for the case of high speed. The load factor for the former case is therefore given by the expression

$$\frac{Q_p}{Q_h} = \frac{1.5}{4.5} \frac{C_{zp}}{C_{zh}} \frac{V^2_p}{V^2_h}$$

in which the indices p relate to the diving and the indices h relate to the high speed.

According to the above figures, the ratio C_{zp}/C_{zh} has a mean value of 0.7. By starting with this value, we find that the test load deduced from the diving does not exceed that deduced from the maximum horizontal speed, unless the diving speed appreciably exceeds the double (2.24) of the former. Under these conditions we can generally abide by the case of high speed.

The above conclusions were deduced from the results obtained for ailerons with a chord equal to 40% of the chord of the corresponding wing. Similar conclusions were reached for relatively narrower ailerons (20%, for example). Since the relative chords of the ailerons now in use are between 16 and 40%, these conclusions can be accepted in a general way.

The results with the 20% ailerons, for the profiles tested, are as follows:

Angle of attack with reference to zero lift	100 c_z	$f \frac{c_z}{C_z}$
	Profile 430	
15	17.5	0.96
9	27	1.8
6	24.5	2.4
	Profile 389	
15	22.5	1.3
9	18.5	1.3
6	17	1.8
	Profile 387	
15	29.5	1.6
9	18	1.3
6	16	1.7
	Profile 382	
15	12.7	0.8
9	12.5	0.9
6	11.5	1.3
	Breguet Profile	
15	16	0.96
9	14.8	1
6	15	1.5
	Halbronn Profile	
15	6.6	0.4
9	20	1.3
6	11.6	1.1

The first conclusion to be drawn from these results is that it is not always sufficient to load the ailerons as simple portions of the wing, as in the ordinary static tests, because these tests are generally made under conditions of flight at large angles of attack, i.e., according to what we have just seen, under conditions which are not the most important from the standpoint of the ailerons.

The above values, as also the corresponding pressure curves (Fig. 1) show that, at high speed, the distribution of the air forces on the ailerons is very nearly triangular. Under these conditions, for the same profile and for a wing equally loaded per unit area, the mean value of the test load is proportional to the chord of the aileron. Moreover, the load must be so distributed that its center of gravity will be at a distance of 1/3 of the chord of the aileron back of its hinge edge.

For the purpose of establishing a general formula, utilizable in the case of profiles for which the distribution of the pressures is not accurately known, we can take for the ratio c_z/C_z*, the mean value corresponding to the angle of 6° with reference to the line of zero lift, a value deduced from the preceding data. For a given chord of the aileron we finally obtain the following expression for the test load.

*The mean value of c_z/C_z is 0.6 for ailerons with a relative chord of 40% and 0.33 for ailerons with a relative chord of 20% of the wing chord.

$$q = \frac{Pf_2}{S} \frac{c_z}{C} = \frac{Pf_2}{S} \times 1.8\ p$$

p being the ratio of the chord of the aileron to that of the wing.

Moreover, it may be convenient to employ in this expression the load factor f_1 corresponding to the first flight case, which is the most used. In this case, since $f_2 = \frac{3}{4} f_1$, we have the following definitive formula:

$$q = \frac{P}{S} f_1 \times 1.35.$$

It must be remembered that, in all that precedes, we have assumed the ailerons to be in the neutral undeflected position.

It may be further noted that the above formula is very similar to the one indicated by certain English writers (Pippard and Pritchard), who state, in fact, that the normal load supported by the ailerons in flight has the form $\frac{P}{S} p$, which is equivalent to admitting that the distribution of the pressure is triangular over the whole wing, the base of the triangle being at the leading edge and its apex at the trailing edge. The curves in Fig. 1 show that this hypothesis is insufficient and would generally give too small aileron loads. However that may be, the above formula gives the following test loads for the five military airplanes already taken as examples.

	Total weight (kg)	Wing area (m^2)	$\frac{P}{S}$ (kg/m^2)	f_1 (1926 spec.)	p	Test load $1.35 \frac{P}{S} f_1$ p (kg/m^2)
Nieuport 29	1,180	27	43	12.8	0.19	140
Potez 15	1,815	45	40	8.6	0.29	135
Breguet 19 B 2	2,347	48	49	8.3	0.21	115
Goliath-Renault	5,145	162	31	6	0.36	90
Farman Bn 4	11,650	268	43	6	0.16	55

The same formula gives the following results for certain well-known commercial airplanes:

	Total weight (kg)	Wing area (m^2)	$\frac{P}{S}$ (kg/m^2)	f_1 (1925 spec.)	p	Test load (kg/m^2)
Bernard	1,170	11	107	6	0.18	155
Farman Sport	410	19.5	21	8	0.32	70
Breguet 14 T	1,908	49	39	7.5	0.41	160
De Havilland 34	3,000	55	55	7	0.20	105
Jabiru	5,220	31	64.5	6	0.155	80

It is now fitting to examine the case of ailerons when they are deflected upward or downward from their neutral position. Most of the researches made on this subject were in connection with experiments on small models conducted as follows:

The two ailerons were given equal and opposite deflections. The stabilizing couple for the model was then measured. It was assumed that the total force exerted on each aileron was equal to the quotient of the moment of the couple divided by the distance between the centers of the two ailerons. Unfortunately, this method can not give very accurate results for the following reasons:

1. For the same angles of deflection, one up and the other down, the forces exerted on the two ailerons are very far from equal, as we shall see farther on.

2. The stabilizing couple exerted on the airplane is not produced alone by forces applied directly to the ailerons, but also by dissymmetries in the aerodynamic properties of the wings, created by their deflection.

3. This method disregards the fact that the ailerons are under load when they are in the neutral position, i.e., when the stabilizing couple is zero.

In order to elucidate the problem, it would therefore be necessary to avoid the intermediation of the stabilizing couple and measure directly the stresses on the ailerons themselves. This has recently been done in the large wind tunnel of the S.T.Aé. at Issy-les-Moulineaux. The dimensions of this tunnel render it possible to experiment with models so large that the relatively small forces under investigation can be measured with sufficient accuracy. Owing to these conditions, the results obtained at the S.T.Aé. are worth examining.

The method of testing is shown by Fig. 2. Two profiles (the 430 and the I.A. of the S.T.Aé., or the Halbronn profile) were successively tested. In the two cases the wing and aileron had the same respective dimensions. The chord of the wing was 1.6 m (63 in.) and that of the aileron 0.25 m (9.84 in.) (ratio p = 0.16). The span of the wing was 2.3 m (7.55 ft.) and that of the aileron 1.6 m (5.25 ft.). The wing was mounted rigidly in the tunnel, the aileron alone transmitting to the balance the forces to which it was subjected. The air velocities varied between 50 and 53 m (164 and 174 ft.) per second. The clearance between the wing and aileron, the importance of which will be evident farther on, was 11 mm (0.43 in.) for the Halbronn wing and 9.5 mm (0.37 in.) for the 430. The measurements made under these conditions gave the following results:

Halbronn Profile*

Inclination of wing with reference to				
	zero lift...	3.5°	8.5°	13.5°
	tangent chord..	0°	5°	10°
Resultant normal to lower surface of aileron when deflected from its neutral position	-12°	-28.5	-26	-26
	- 6°	- 8.5	- 7.8	- 5.6
	0°	17.6	19	24.6
	6°	36.6	39	44.6
	12°	53.3	57.4	66.2
	18°	62.6	--	--

Profile 430

Inclination of wing with reference to				
	zero lift...	2.9°	4.9°	7.9°
	tangent chord..	-5°	-3°	0°
Resultant normal to lower surface of aileron when deflected from its neutral position	-12°	-17.2	-14.5	-14.1
	- 8°	- 1.6	1.05	1.6
	- 4°	+14.1	16.8	17.8
	0°	28.4	31.2	33.8
	4°	41.8	44.7	48.4
	8°	55	58.3	63
	12°	66	69.5	75.3

* The values of the normal resultants were calculated from the values of the c_x and c_z measured directly on the aileron. The negative values correspond to a downward resultant. The negative angles correspond to the upward deflection of the aileron. The zero position corresponds to the undeflected ailerons.

Before analyzing these data in detail, we should recall how, as already mentioned, the experiments performed at St. Cyr and especially the pressure tests made with small wing models, without hinged aileron, enabled the valuation of the forces supported by the ailerons in their neutral position.

For the two profiles tested at the same angles of attack and with the same ratio of $p = 0.16$ between the chord of the aileron and that of the wing, the data obtained from the Issy and Eiffel experiments give the following comparative results:

Halbronn Profile

		Values of 100 C_z	
		Deduced from Eiffel data	Measured at Issy
Inclination with reference to tangent chord	0°	7.5	17.6
	5°	13	18.8
	10°	10	24.2

Profile 430

Inclination with reference to tangent chord	-5	17	28.4
	-3	18	31.3
	0	20	33.4

Thus the values of C_z obtained from direct measurements on the two profiles are 1.5 to 2 times those deduced from the experiments with a small model. What is the reason for these differences and which of the two series of values is more accurate? An analysis indicates two causes, one secondary and the other more important.

The secondary cause is the difference in the nature of the air flows past the small model tested in the Eiffel tunnel and past the wing fragment of large size but small relative span tested at Issy.

The principal cause is the effect on the aerodynamic properties of the aileron produced by the clearance or slot between it and the wing. This effect was demonstrated by the following experiment, performed at Issy, on the same wing model (430) as before. The inclination of the wing being 0° with reference to the chord tangent to the lower side of the wing, and the aileron being in its neutral position, the values of the aerodynamic pressures, supported by the model at different points on the same wing section, were determined. The measurements were made successively for two sections, located at about the middle of each half of the aileron, and their mean value was taken. These mean values were also found for the case of zero clearance (slot closed by paper); for a slot 4 mm (0.157 in.) wide; and for a slot 9 mm (0.354 in.) wide. The results thus obtained are as follows:

a) The presence or absence of a slot does not appreciably affect the pressure on the lower side.

b) On the contrary, the presence of a slot at the articulation greatly increases the negative pressure on the lower side near the leading edge of the aileron. If d_0 is the negative pressure measured near the leading edge, with the slot closed, the negative pressure d_9 measured with a 9 mm (0.354 in.) slot can attain a value of $4\ d_0$.

c) When the width of the slot is changed, the value of the negative pressure changes in the same direction, but in a much smaller relative degree. Thus the negative pressure $d_9 = 4\ d_0$ still remains in the vicinity of $3\ d_0$, when the width of the slot is reduced from 9 to 4 mm.

The measurements made do not yet render it possible to determine accurately the effect which the increase in the negative pressure created by the slot at the leading edge will have in all cases on the forces supported by a full-size aileron. It is probable that this effect is a function of the aileron chord and of its span as compared with that of the corresponding wing. However this may be, if we abide by the results obtained on the 430 model, Fig. 3 shows that the presence of a 9 mm (0.354 in.) slot can cause for this model an increase of about 50% between the stresses measured on the aileron with the slot open and with it closed.. This increase is of the same order as that of the differences indicated by the above comparative table of the values of 100 C_z and renders it possible thus to explain their origin.

We can conclude that the test loads previously calculated according to the formula $q = \frac{P}{S} f_1 \times 1.35\ p$ hold good only for the case when the slot is very narrow, in order to have only a negligible effect on the stresses undergone by the aileron during flight. Otherwise one must increase these loads by an amount which may attain about 50%, that is, replace the original formula by the formula $q = \frac{P}{S} f_1 \times 2 p$.

On referring to the results obtained with the two wing models I A and 430 (Fig. 4) tested at Issy with their ailerons, it is obvious that we can draw the following conclusions.

1. The normal stress undergone by an aileron deflected downward is very much greater than that supported by the same aileron deflected upward the same amount. For the profile 430 with a deflection of 10° in each direction, the aileron deflected downward is stressed about 9 times as much as the aileron deflected upward. This result confirms the objections already stated against the methods for calculating the strength of ailerons based on a too exclusive consideration of the evolution couple they produce.

2. The value of the normal stress N varies but little for the same wing and for the angle of deflection of an aileron, when the angle of attack of the wing varies only within the customary limits.

3. The normal stress N is very nearly proportional to the angle of deflection. The inclination of the straight line is very nearly the same for the two profiles tested and leads to the formula $100\ N_\alpha = 100\ N_0 + 3.2\,\alpha$ (α being expressed in degrees).

Under these conditions, the surcharge produced by the deflection will be given for a full-size aileron by an expression of the form $3.2\,\alpha\,\frac{a V^2}{2g}$, that is, $0.002\ V^2\alpha$, α being expressed in degrees or by $0.115\ V^2\alpha$, α being expressed in length of arc.

The latter formula can be compared with the one proposed by several writers, who have advocated the adoption of the formula of Joessel for the calculation of ailerons. Joessel gives, for the force supported by an elevator which makes the angle β with the direction of the air current, the expression $\frac{0.08\ V^2\beta}{0.4 + 0.6\beta}$ which can be written $0.17\ V^2\beta$ for angles between 0 and 10°.

The experiments performed in the Issy tunnel on the two profiles considered, prove therefore that Joessel's formula is not directly applicable to the calculation of ailerons, but that we can utilize it to a certain extent by replacing the angle β, between the surface and the direction of the air current, by the angle α of the deflection of the aileron and by adding, to the value thus calculated, the mean force supported by the aileron in the neutral position.

Lastly, if we adopt the expression $0.002\ V^2\alpha$ as the first approximation for calculating the supplementary force produced by the deflection, we must know by what safety factor it is to be multiplied in order to determine the static test load to which the aileron should be subjected.

We have already seen that, for the aileron and its wing to have a mechanically homogeneous strength, it is necessary to apply to the force supported by the aileron in the neutral position a load factor of $f_2 = \frac{3}{4} f_1$. The same factor would seem to need to be applied to the supplement of the force produced by the deflection. Anyway it is necessary to take account of the fact that the ailerons are never deflected suddenly at high speed and that the speed of the airplane will gradually diminish as the angle of deflection increases. This result can be obtained by conserving for V, in the formula $0.002\ V^2\alpha$, the value of the maximum speed and by limiting α to 10°.

Finally we obtain, for the test load of the ailerons, an expression of the form

$$Q = f_1 \left(\frac{P}{S} \times 2\ p + \frac{1.5}{1000}\ V^2{}_h \right),$$

Q and P/S being expressed in kg/m², and V_h^2 in m/sec. The following table gives the test loads resulting from the above formula for the airplanes already mentioned.

	V_h		f_1	$\frac{1.5\ f_1}{1000} V_h^2$	Test load of aileron at 0	Total test load (kg/m²)
	km	m/sec				
Nieuport 29	230	64	12.8	79	210	290
Potez 15	190	53	8.6	36	200	235
Breguet 19	220	61	8.3	46	170	215
Goliath Renault	166	46	6	19	135	155
Farman Bn 4	183	51	6	23	85	105
Bernard	448	125	6	140	230	370
Farman Sport	143	40	8	19	105	125
Breguet 14 T	170	47	7.5	25	240	265
D.H. 34	170	47	7	23	150	175
Jabiru	204	57	6	29	120	150

Thus the surcharge due to deflection is generally $\frac{1}{4} - \frac{1}{3}$ of the load in the neutral position. It may be much larger, however, especially for swift airplanes. It is not negligible therefore, as assumed by some writers. In order to supplement this result, it is important to determine the position of the center of lift on the chord of the aileron. The measurements made from this viewpoint gave the following results:

Profile 430

Inclination of wing with reference to	zero lift...	2.9°	4.9°	7.9°
	tangent chord..	-5°	-3°	0°
Positions of center of lift on aileron chord for angles of deflection with reference to neutral position	-12°	0.27	0.27	0.27
	- 4°	0.34	0.34	0.34
	0°	0.32	0.32	0.31
	4°	0.33	0.33	0.33
	12°	0.39	0.39	0.38

Halbronn Profile

Inclination of wing with reference to	zero lift...	3.5°	8.5°	13.5°
	tangent chord..	0°	5°	10°
Positions of center of lift on aileron chord for angles of deflection with reference to neutral position.	-12°	0.27	0.27	0.30
	- 6°	0.27	0.20	0.20
	0°	0.27	0.35	0.37
	6°	0.28	0.40	0.43
	12°	0.33	0.50	0.53

For a deflection of 10° downward, which has already been considered, they show that the center of lift is at 40-50% of the chord. This result confirms the grounds for the rule followed in the United States (1922 specifications) which stipulated that, in the static tests of ailerons, the center of gravity of the test load should be located at 5/12 of the chord. This distribution is nearly that which would be represented by a right trapezoid having the side perpendicular to its bases on the leading edge and the smaller base equal to about half the larger.

Hitherto attention has been given only to normal stresses on the lower side, which are, in fact, much the most important. Nevertheless, it is well to determine the stresses of drag with the variations they undergo. The measurements made on the two profiles, tested under the same conditions as above, gave the following results:

Halbronn Profile

Inclination of wing with reference to	zero lift...	3.5°	8.5°	13.5°
	tangent chord..	0°	5°	10°
Drags in the plane of the aileron when deflected with reference to its neutral position	-12°	-6.7	-6.3	-7.5
	- 6°	-2.4	-2.4	-4
	0°	0.3	1.5	0.2
	6	0.9	5.3	3.2
	12	3.3	10.8	6.4

Profile 430

Inclination of wing with reference to	zero lift...	2.9°	4.9°	7.9°
	tangent chord	5°	3°	0°
Drags in the plane of the aileron when deflected with reference to its neutral position	-12	-3.5	-3	-2.5
	- 8	-0.2	0.2	-0.8
	- 4	-1.4	2.3	3
	0	3.5	3.2	4.2
	4	3.9	3.7	4.4
	8	4.5	3.8	5
	12	5.4	4.4	-

These figures show that the forces in the plane of the ailerons are, at their maximum, only about 0.1 of the forces perpendicular to this plane. Moreover, it is obvious that the drags for upward deflections can be negative, that is, can support the aileron on its hinges, instead of tending to separate it from the wing. The change in the lift occurs at the moment the action normal to the plane of the aileron becomes zero.

These results render it possible to account for a few interesting details in the functioning of the ailerons during flight.

We have, in fact, already seen that the force supported by an aileron can be put in the form $(a + b\,\alpha)\,V^2$, α being the angle of deflection. This expression also gives the value of the tension of the controls between the pilot and the aileron. When the pilot operates the control stick, he must overcome the difference between the tension of the control of the aileron deflected downward and that of the aileron deflected upward, that is, $[a + b\,\alpha - (a - b\,\alpha)\;V^2]$ or $2\,b\,\alpha\,V^2$.

It will be first noted that the tensions of the two controls are equal only when α is zero, that is, when the ailerons are in the neutral position. When, on the contrary, the value of α is such that $a - b\,\alpha$ is near zero*, we have a common organ actuating two controls, one being under high tension and the other under low tension. The effect of the former will evidently predominate as regards the reflexes and maneuvers of the pilot. If the control terminating at the aileron deflected upward has any play, nothing can prevent this aileron from flapping about its neutral position like a flag in the wind. The flapping is all the more liable to occur when the value of α is relatively small. The flapping is more vigorous in proportion to the strength of the couple of recoil. Both these circumstances correspond to high values of b, that is, to the factor of in-

* We have already seen that this angle is 4.5° for the Halbronn and 8° for the 430 profile.

crease of the forces in terms of the deflection. Of course the flapping, if due to the above causes, will increase in violence with increasing speed of the airplane. Two accessory phenomena may still further increase the flapping. We have already seen that, in the neighborhood of the deflection corresponding to zero stress, the aileron changes its support on its hinges. This change may occur, if the controls have any play, at a flapping of the aileron parallel to its plane. It can, moreover, periodically change the width of the slot or clearance between the aileron and its wing.

We have already seen the effect this slot has on the stresses to which the aileron is subjected during flight. The periodical opening and closing of the slot, produced by the oscillations of the aileron will then be accentuated by the sudden rupture of equilibrium entailed by the flapping of the aileron about its axis.

The above remarks explain, at least in part, the vibration of the ailerons, which are often manifest on airplanes, especially at very high speeds (either horizontal or diving). These vibrations may be strong enough to dislocate the ailerons and even the neighboring ribs of the wing. They are therefore very dangerous.

If the above explanations are correct, it is obvious that, in order to avoid the vibrations, it would be necessary to eliminate all play in the controls and reduce to a minimum the clearance between the ailerons and the wing.

The experimental device employed at Issy can give indications not only concerning the forces acting directly on the ailerons, but also regarding their effect on the aerodynamic properties of the wing. For this purpose we measure the positive and negative pressures exerted at various points of the same profile for several angles of deflection of the ailerons. If we operate in particular on the sections B and C of the 430 model and plot the mean results, we obtain the curves shown in Fig. 5.

The curves first show the effect of the clearance slot in the case of a deflected aileron. They also show that the progressive couple created by the deflections of the two conjugated ailerons is due especially to the modifications they introduce in the aerodynamic properties of the wing. These modifications affect the larger portion of the wing chord.

In brief, the preceding considerations show:

The importance of knowing the maximum stress undergone by the ailerons when much deflected, as well as in the neutral position. These stresses are greater than generally supposed.

The presence of an appreciable clearance or slot between the aileron and the wing increases these stresses.

If we wish to resort to a mean formula for determining the load an aileron should withstand, in order to afford guaranties similar to the ones required of wings, the formula

$$q = f_1 \left(2 \frac{P}{S} p + \frac{1.5 V^2}{1000}\right)$$

would seem to serve the purpose. It would seem advisable, however, to try experiments on other models than the ones hitherto used, in order to determine whether this formula is general enough.

At the same time the pressure data already obtained should be supplemented in order to determine the aerodynamic conditions for the functioning of ailerons as stabilizing organs.

It would be desirable to verify these data by a few direct flight tests, especially of the stresses undergone by the controls, at various engine speeds and for various angles of deflection.

Translation by Dwight M. Miner,
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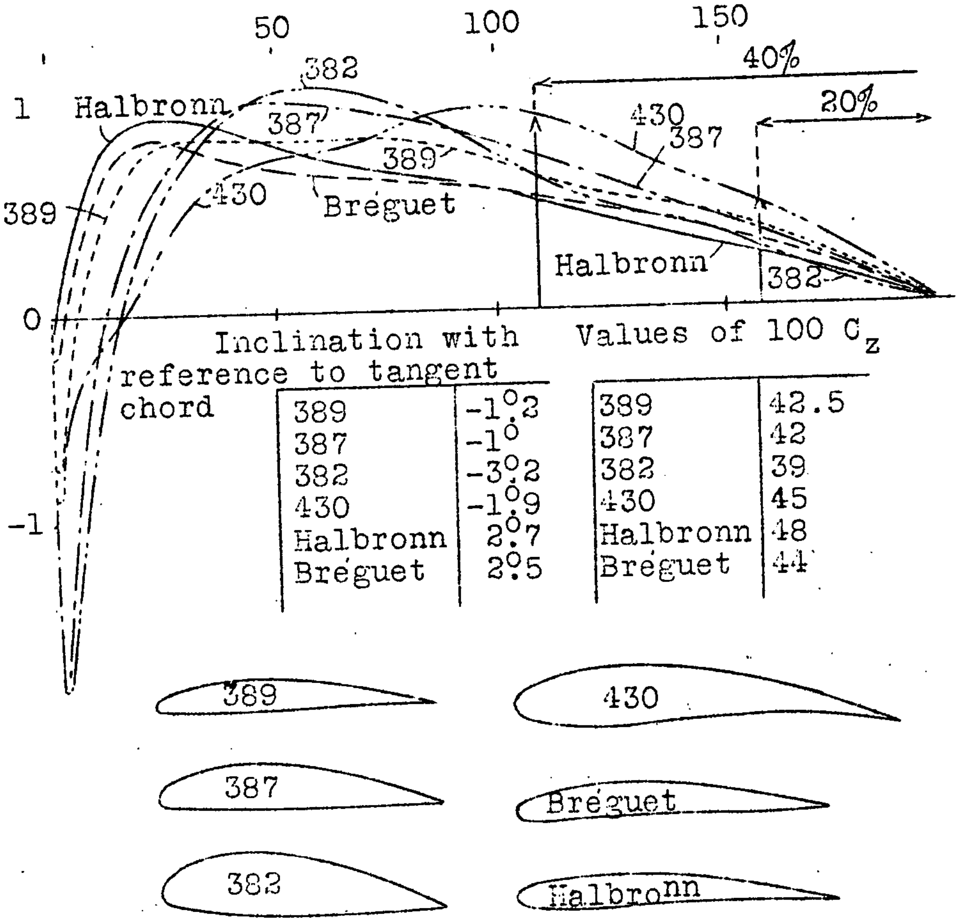


Fig.1 Distribution of loads on ribs for an inclination of 6° with reference to chord of zero lift.

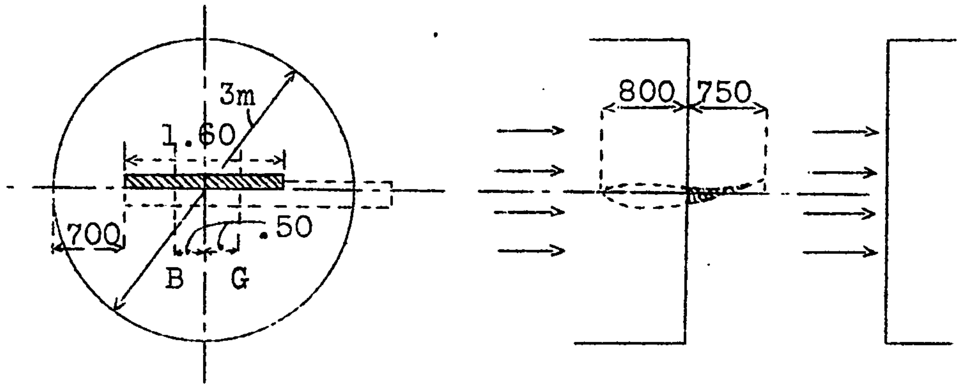


Fig.2 Position of model tested in the Issy wind tunnel.

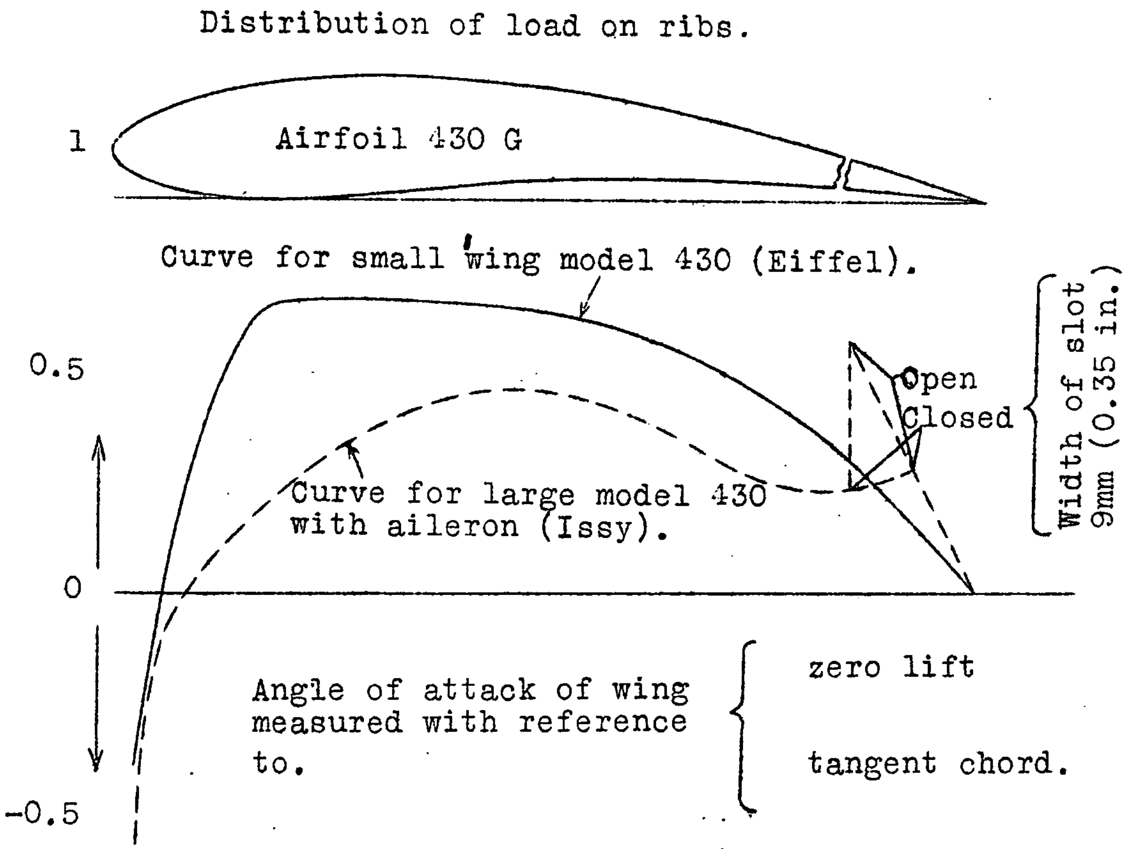
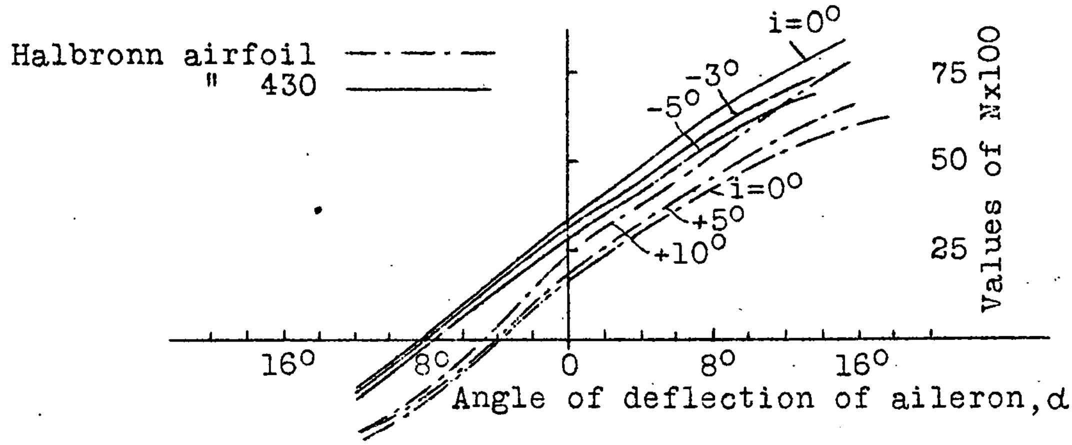


Fig.3



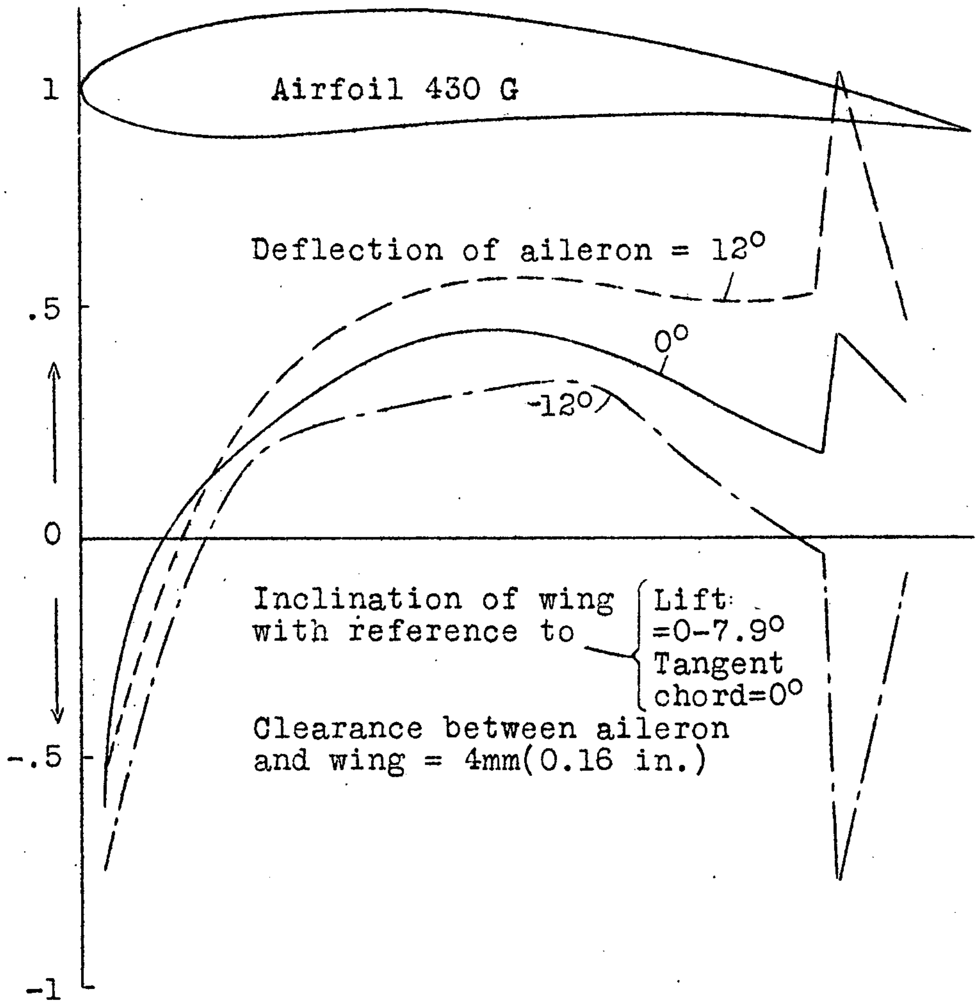


Fig.5 Distribution of loads on ribs. Curves obtained at Issy on large model with aileron.