TECHNICAL NOTES.

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS.

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LOADS AND CALCULATIONS OF ARMY AIRPLANES.

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Translated from
Technische Berichte, Vol. III, Section 6,
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Office of Naval Intelligence, U.S.N.

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A military airplane must fulfill two conditions:

1. It must be aerodynamically capable of good performance.

2. It must also be structurally strong enough to maneuver without risk of failure.

It may seem superfluous to note yet it can not be overemphasized that an airplane of good flying qualities is useless if it is not sufficiently strong and fails at the critical moment. A justified lack of confidence on the part of the pilot in the strength of his airplane will prevent him from taking complete advantage of its possible merits. Therefore a constructor should not be criticised for carefully investigating the strength characteristics of his designs.

What then, must be the requirements for strength so that a pilot will have at his disposal a powerful but not too heavy airplane. The answer depends on the magnitude of the air forces or accelerations to which the moving airplane is subjected. Investigations of these forces have not been carried far enough so that they can always be estimated, and, where there is apparently a clear understanding of the effect of the air forces, mathematical methods for expressing these effects in useful form are lacking. It is not even possible to cover all possible cases by full flight investigations of these phenomena, since such maneuvers as an airplane pilot ventures or instinctively carries out in moments of great danger can not be imitated inten-

* Checked by D. L. Bacon, Assistant Physicist, Aeronautical Laboratory, N.A.C.A.

tionally. It is in just such cases that the machine is subjected to the most dangerous loads.

Because of these difficulties it is useful and necessary to be governed by experience. That is, if one airplane of known strength has resisted all the usual and even the extreme air loads to which it can be subjected and if a second airplane having less strength than the first was found, under like conditions to be insufficiently strong, it is obvious that the strength required is greater than that of the second airplane but not greater than that of the first. If, as is now actually the case, strength factors of different airplanes of the same class do not differ to any great extent, then the permissible values are fairly well defined.

On the basis of such experience, aided by scientific investigations, standards have been developed which are satisfactory for the calculation of airplane structures.

I. Loads on the Wing Truss.

Loads on the wing truss are expressed as multiples of the weight of the airplane, the air forces which keep the airplane in equilibrium being proportional to its weight. If, furthermore, the admissible and simplifying assumption is made that the air forces which counteract the forces of acceleration on the wings of the moving airplane are equally distributed along the wings, then the air forces on the wings may be considered as being proportional to the quantity: Total weight minus weight of wings.

Hence the weight of the wings themselves is not to be included when calculating the stresses in the wing truss. The wings are supported by the sustaining air pressure; and that portion of the air forces that balances the forces of acceleration on the wings, being equal and opposite and acting at the same points, can not give rise to any moments in the structure.

The numbers by which the quantity Total weight - weight of wings are to be multiplied in order to obtain the applied load are called "Load Factors." These vary according to the conditions of flight and have, for convenience, been grouped under four representative conditions or "load cases."

1. Case A - Taking off (or large angle of incidence).
2. Case B - Gliding at 30° to the horizontal.
3. Case C - Nearly vertical dive.
4. Case D - Upside down flying.

However, it would not be justifiable to use the same load factors for all types of airplanes even under the same flight conditions. As already stated, the air forces on the wings correspond to the acceleration forces on the moving airplane. Since the acceleration forces depend on the rate of change of velocity they are therefore greater the more rapid and maneuverable an airplane is. Since usually the speed and maneuvering ability depend on the weight or useful load it is evident that types of airplanes can be divided according to weight or effective load into different "calculation groups." According to present practice, airplanes may be divided into five such groups:

<u>Calculation group I</u>. Airplanes of any type of construction having a flying weight greater than five tons; i.e., giant planes that are required to cover great distances with the maximum possible useful load, abundant fuel and bombs, but without great speed.

<u>Calculation group II</u>. Airplanes of two and one-half to five tons flying weight and one to two tons useful load; this includes small and short range bombers.

<u>Calculation group III</u>. Airplanes of two and one-half to four tons flying weight and 1700 to 3300 lbs. useful load. This includes those larger flying machines which are distinguished from those of groups I and II by greater fighting value and consequently greater speed and mobility.

<u>Calculation group IV</u>. Airplanes of 2600 to 5500 lbs. flying weight and 900 to 1800 lbs. useful load. These include two-seaters for photographic, reconnaissance, battle and scouting purposes.

<u>Calculation group V</u>. All airplanes of less than 2600 lbs. (up to 900 lbs. useful load), hence single seaters and light two-seaters of any kind.

Of course no such division into classes can be correct in all details if it is intended to be general. In order that the classification be pliable and afford room for new types it has been decided that the Imperial Airplane Department be authorized to decide in which group a new type should be placed.

The load factors required can thus be originally determined from the flight conditions and calculation groups. The accuracy of calculated factors depends on the exactness with which the distribution of loads among the structural members can be determined.

It is obvious that stress analysis may be avoided and the strength of the finished airplane demonstrated by sand loading. This was customary in the early days of airplane construction; new types were developed by trial, the machine was completely built and subjected to the sand load. This method no longer suffices, for unavoidable errors in construction caused great losses in time, labor, and raw materials. Today science gives the constructor means to calculate the required dimensions of the structural members from the beginning of the design. Furthermore, today it also furnishes methods by which the best form and most suitable construction consistent with least weight may be determined. In spite of this progress it is not possible, owing to the lack of trained workers and dependable structural materials to entirely do away with sand loading.

Although the strength of a flying machine may have been carefully calculated, it should still be required that its structural strength be demonstrated by a sand load.

The mathematical calculations of the strength of individual members of the wing truss do not exactly coincide with the results of the actual load tests. In the mathematical analysis the truss is considered as a structure built of separate and individually considered members. Actually, however, the wings do not act as mere jointed frameworks of spars and compression struts, but, because of the added stiffness of ribs and fabric, act somewhat like separate rigid bodies.

In the case of test loading these influences have an effect whereas in calculation they are neglected. Accordingly two requirements must be satisfied under conditions A, B, and D, one for purposes of calculation, using low load factors, and the other for sand testing with high load factors. In case C a double requirement is not necessary for a high head resistance is indicated even by calculation and because the relative stiffness of the wings is smaller in proportion to the torsional moment while diving.

Tables I and II give the load factors compiled according to the above considerations, for stress analysis and for sand testing. The loads specified are breaking loads; i.e., under these loads the members should be stressed nearly to their ultimate strength. This does not mean that the air forces encountered are actually as great as the sand loads specified. These factors include a certain factor of safety to allow for the nature of the load and for the characteristics of the structural materials.

The following effects have been considered in this connection:

1. The loading during flight is not constant but changes suddenly in such a manner that the strength characteristics of the materials, especially their stiffness, can not be fully developed. This includes both suddenly changing air loads and the effect of motor vibrations.

2. Some members are unduly stressed due to high and indeterminate initial tensions in the stays.

3. Wooden members, such as spars, ribs, and struts deteriorate through the action of time and weather.

4. Experience shows that airplanes in quantity production generally are somewhat heavier than the original model machine, due to subsequent changes and additions.

The foregoing affords a basis for the statement that the values given in the table are for ultimate static loads. It would therefore be incorrect to assume a factor of safety when any structural member appears to be loaded in excess of the safe limit of the material.

The increases in magnitude of the load factors in Table II over those in Table I are the result of experience. A comparison between calculated strength and actual breaking strengths has shown no differences amounting to as much as 30%. These differences however decrease as the size of airplane increases, and a corresponding allowance has been made in the tables.

It seems superfluous to discuss in detail the prescribed loads indicated in Fig. 1.

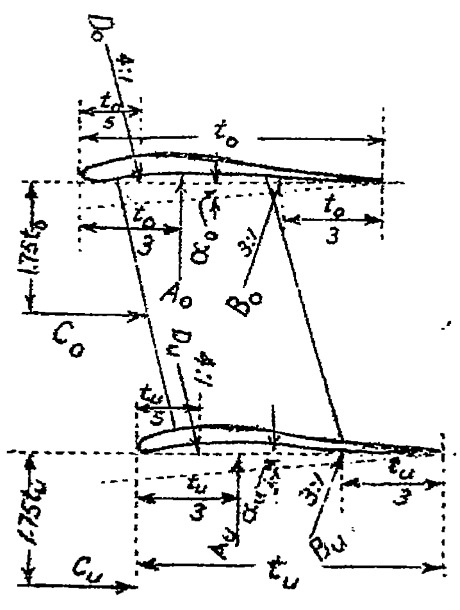

Table I. Load Factors for use in Stress Analysis.

Group	Type		Case A	B	C	D
No.	Flying wgt. lbs	Useful load lbs	take off	glid-ing	div-ing*	upside down
I	over 11,000		3.5	2.5	1.2	----
II	5,500 - 11,000	2,200 - 4,400	4.0	2.5	1.5	----
III	5,500 - 9,000	1,800 - 3,300	4.5	3.0	1.75	2.5
IV	2,500 - 5,500	900 - 1,800	4.5	3.0	2.0	2.5
V	less than 2,500	less than 900	5.0	3.5	2.0	3.0

Table II. Load Factors Required in Sand Testing.

Group	Type		Case A	B	C	D
No.	Flying wgt.lbs.	Useful load,lbs	take off	glid-ing	div-ing*	upside down
I	over 11,000		4.0	2.5	1.2	----
II	5,500 - 11,000	2,200 - 4,400	4.8	2.6	1.5	----
III	5,500 - 9,000	1,800 - 3,300	5.5	3.2	1.75	2.8
IV	2,500 - 5,500	900 - 1,800	5.7	3.3	2.0	2.8
V	less than 2,500	less than 900	6.5	4.0	2.0	3.5

* The factors for Case C apply to drag forces but not to diving moments.

As long as no new developments prove the insufficiency of these assumed loads we may consider them to represent the facts. An explanation is needed however, for the increase in the diving moments in Case C (Vertical Dive). During a dive the air forces act so that the front part of the wing experiences a pressure from above and the after part a pressure from below. Thus a couple is set up, producing a moment about an axis parallel to the spars. According to recent tests in the Göttingen wind tunnel and according to theoretical aerodynamic investigations the moments formerly assumed have been too small. If the moment about a wing be expressed by

Constant x Chord x (Drag Load) where

Drag on upper wing + Drag on lower wing = Total weight - weight of wings,

then the constant must be so chosen that when multiplied by the chord it will give the proper moment arm.

From a number of experiments the following average values were found:

$$M \text{ upper wing} = 1.75\, R^x_{\text{upper}} \text{ chord, instead of } \frac{2}{3} R^x_{\text{upper}} \text{ chord.}$$

$$M \text{ lower wing} = 1.75\, R^x_{\text{lower}} \text{ chord, instead of } \frac{2}{3} R^x_{\text{lower}} \text{ chord.}$$

Experience shows that we have enough measurements with the hitherto existing moments and load factors - except for internal stresses in the lower wing - so that we would not be justified in increasing the lever arm two and a half times to increase the moments proportionately.

Returning, then, to known moments, load factors for the moments in Case C are omitted. In order, however, that too small internal stresses shall not be obtained, the partial forces acting as drag on the wing surfaces are multiplied by the corresponding load factors.

This requirement is not contradictory. It takes account of the fact that the spars must be strong enough to suit other loading conditions, while the internal stresses should be particularly investigated for diving conditions.

It should also be realized that in diving a large portion of the air force is taken up upon the body and its appendages, which is not taken account of either in the calculations or in sand testing, thus, after experience with the old requirements, the new ones must also suffice to guard against insufficient dimensions.

Proportions of the Load Assumed by Upper and Lower Wings. Lacking more complete information on the subject all airplanes are calculated on the basis of a 11 : 9 ratio of load distribution between upper and lower wings. Although this is doubtless a good average value, it is not to be overlooked that a closer approximation to actual values would be desirable. The division of load between upper and lower wings depends on both gap and stagger, accordingly diagrams are given in the 1918 edition Neuauflage der Bau - und Liefervorschriften showing this relation in terms of gap and stagger. In Case C (Vertical dive) the division of load is independent of stagger.

Distribution of Air Forces Along the Span. We have hitherto assumed the air pressure to be evenly distributed along the wing span. This is not entirely correct as the pressure diminishes near the wing tips and within the area of slip stream. For purposes of calculation the pressure per unit length a is assumed to decrease from P at a point one chord length distant from the wing tip to $\frac{P}{2}$ at the extreme tip. For calculating the span itself the full pressure P is assumed to extend to the wing tip.

Unsymmetrical Loading. All previous considerations have reference to a load symmetrically distributed on both sides of the central axis. In curved flight however the load is no longer symmetrical. Nevertheless, no special load case has been introduced for unsymmetrical loads because the cabane and the body longerons supporting it are the only parts unusually stressed.

II. Loads on Control Surfaces.

In determining the loads on control surfaces the same difficulties are encountered as with the main supporting surfaces. As far as the control surfaces themselves are concerned the loading is unimportant as they are so small that a high factor might easily be obtained with negligible increase in weight. The control surface loadings are important however in proportioning the members of the body which support them. Because of its rigid construction the body is usually to be considered as a single unit, and changes in the position of the center of pressure on the tail cause only a slight variation in the moments on the body, so that only the magnitude of these forces is of importance. The task is thus essentially simplified, it only remains to discover what are the maximum possible loads on the tail surfaces.

The tail surface load may be expressed by

$$q = \mu . \frac{\gamma \; . \; V^2 max}{2g}$$

and is principally influenced by the speed and the constant. The maximum speeds for any type of machine are known approximately or are specified. The maximum value of q depends on the angle of incidence of the fin and the displacement of the rudder. As these are limited by the construction of the machine it will suffice to consider the least favorable case. The Research Laboratory at Göttingen has carried out a great many investigations of control surfaces* which have furnished reliable data for calculation. The following loads are specified for the various type classifications, which furnish a basis of comparison regardless of speed.

Table III Specified Tail Plane Loads.

Group Number	I	II	III	IV	V
Average Load lbs/sq.ft.	24.6	24.6	30.7	36.9	41
Average Load Kg/m^2	120	120	150	180	200

The values given in this table agree closely with recent practice. Because of possible damage to surfaces during shipment and handling, fins on airplanes of Group I are calculated on a basis of 200 Kg/m^2 and of Groups II to V inclusive on a basis of 300 Kg/m^2. Rudders are likewise calculated at 200 Kg/m^2. Experience has shown that rudders so designed are sufficiently stiff and strong.

III Loads on Wing Ribs.

The nature of rib loads has been previously considered in Technische Berichte, Vol. I, No.3, p. 81. The same load factors are used as for the wing truss in Cases A, B, and D. In Case C the moment specified for the truss should be increased 50%. The strength of wing ribs must always be demonstrated by trial loading for, because of the minute di-

* Technische Berichte, Vol. I, No.5, p.168.

mensions of the component parts, mere calculation is insufficient.

IV Loads on Landing Gear.

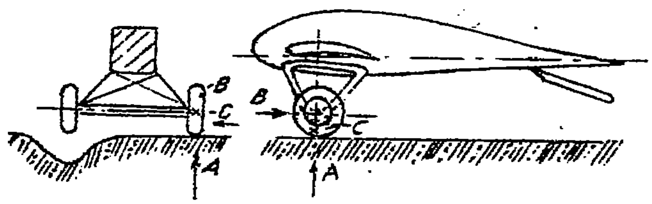

Fig. 2.

Three conditions of loading are to be considered (see Fig. 2). A load directly under one wheel (force A), a retarding force (force B) and a side load (force C).

Forces A and B or A and C act simultaneously and amount at least to the following multiples of the static loads per wheel (one half total weight for two wheeled airplanes).

Force	Multiple
A	6
B	4
C	0.6

V Calculation.

Static investigations are carried out according to established methods. For load distribution between members of the wing truss it is immaterial what loading is used, for the stresses are proportional to the load and can subsequently be found for any load factor. In contrast to this are the fiber stresses in members, such as wing beams, under compression and bending loads where the stresses are not directly proportional to the load factor but to an exponential function thereof.

For what load factor, should the calculations then be carried out? The static analysis would be most exact under loads which do not stress the members above their elastic limits. This procedure is, however, subject to objections.

1. For wood, the most common structural material, the elastic limit is difficult or impossible to define, and varies so much under different conditions that it does not furnish a good basis for calculation. The ultimate strength, however, for any cross section and type of loading may be determined with sufficient accuracy.

2. As already stated, the specified load factors do not represent the actually experienced air loads but also include effects of vibration, deterioration of material, etc. It is important however that the constructor know the behavior of the members under breaking load. As no simple relation between load and strain exist he would be unable to tell, when calculating with light loads, whether the deflections for ultimate load exceeded the limits of safety.

Therefore, if it is not practicable to carry through calculations for several different load values, it is advisable to make the static analysis for the ultimate load.

In calculating wing spars the application of loads must be considered with especial care. Euler's formula does not apply to short struts and the strength of these should be found by experiment as Tetmajer's formula does not hold for struts of hollow section.

VI Rigidity of Materials.

The accuracy of static analysis depends on the use of correct values for the elasticity of the structural parts. For both wood and duralumin, the materials most used, the coefficients of elasticity and elastic limits are so dependent on the size and shape of the member that in some circumstances large errors would be introduced by the use of their average values. This is also true for steel cables. It is therefore essential that the elasticity be measured in each individual case on specimens which check exactly with the required dimensions, and which are subjected as nearly as possible to the desired load. The elastic curve of cables is therefore to be obtained by the use of actual lengths and cross sections, and with due regard to splices and thimbles, in at least three tests.

The determination of elasticity in spars under exact flight conditions, i.e., under compression and bending loads, is not usually convenient because of the lack of time and of laboratory facilities. However, it is considered satisfactory to obtain the elasticity of a beam eccentrically loaded as shown in Fig. 3.

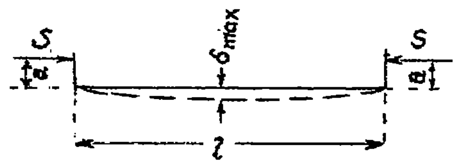

Fig. 3.

The length of the specimen must be equal to the distance between panel points on the wing, and the eccentricity $\underline{a}$ must be so chosen that the maximum bending moment produced will be equal to that on the actual spar.

In order to estimate the distance $\underline{a}$ a first approximation E' is assumed, and $\underline{a}$ is calculated from

$$\frac{S \cdot a}{\cos\left(\frac{l}{2}\sqrt{\frac{S}{E' \cdot J}}\right)} = \text{Maximum bending load on spar.}$$

The test can then be carried out by gradually increasing S to the breaking load, measuring the deflection in the center S after each increase in load. The value of S_{max} immediately before rupture is of the greatest importance, therefore especial care is necessary at this point or the experiment will be worthless. Using S_{max} the correct coefficient of elasticity E, for use in calculating breaking loads is obtained from

$$\cos\left(\frac{l}{2} \cdot \sqrt{\frac{S}{E \cdot J}}\right) = \frac{1}{1 + \frac{\delta_{max}}{a}}$$