TECHNICAL NOTES.

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS.

No. 6.

STATIC TESTING AND PROPOSED STANDARD SPECIFICATIONS.

by

E. P. Warner, Chief Physicist,
Aerodynamical Laboratory, N.A.C.A.,
Langley Field, Va.

Washington,
July, 1920.

NOTES ON STATIC TESTING OF CELLULES.

By E. P. Warner.

The use of static testing, or, as it is more commonly called, sand loading, has been very much extended during the war, and some countries now require a sand load of every new type of machine before accepting it for general use by their air services. A static test admittedly cannot replace a stress analysis, especially if, as is usually the case, sand loading is only to be carried out for one condition, but it is useful as a check and as a means of detecting faulty workmanship and unsuspected weaknesses in fittings and other small parts not readily susceptible of analysis. The sand load should be considered, therefore, as a check to be applied only after the stress analysis has been carried through with all possible refinement of method, and not at all as a substitute for the calculations or as an excuse for shirking them.

Static tests fall into two groups, the first of which is designed to load all the members of the structure approximately in accordance with the worst loads which they carry in flight, while the second is directed to the testing of certain specific members which are suspected of weakness and which are difficult to analyze mathematically. The nature of the loading in the second type is different for every different test, but the purpose of the first is clearly enough defined to permit of the adoption of some standard set of loading specifications, at least for airplanes of normal design. In the pages which follow, an attempt is made to carry through an analysis leading to such a standard, the goal being the determination of a load which will simultaneously impose on every member of the airplane structure a stress equal to the worst that it ever carries in flight. It is manifestly impossible literally to attain this object, as the conditions encountered in the air are various enough to cause reversals of stress in many members, particularly in the front lift truss, and it is obvious that no single sand load, however ingeniously devised, can set up tensile and compressive stresses in the same member at the same time. However, some approximation to the worst loads in most of the members can be secured.

TESTING OF CELLULES.

For a normal test, the airplane is inverted, the fuselage being supported at points close enough to the points of attachment of the wing hinges to insure against any risk of failure of the fuselage. The load is applied by means of quilted bags filled with dry sand, and is supported by jacks at each panel point during the application of each increment of load, in order that there may be no local stresses set up by application of a load to one part of the wings before the corresponding loads are applied elsewhere. After the application of each increment of load the vertical deflection should be measured at every point of support and half-way between each two points of support along each spar. The deflection should also be measured at corresponding points along each trailing edge and at points midway between the spars as an index of the change of form of the section. At the wing-tips the horizontal deflection should be measured, as well as the vertical. Tests are sometimes made with the cellule right-side-up, representing the condition of upside-down flight, but these are less frequent and less important than those for the normal condition.

DISTRIBUTION OF LOAD ALONG THE CHORD.

It is necessary to adopt some arbitrary convention for this distribution; first because it would be impossible to distribute the load to exactly duplicate the rather irregular pressure distribution on the wings at any given angle, and secondly because an airplane has to contend with an almost infinite variety of conditions, each of which corresponds to a different distribution of air load.

The ideal method would be to do as the British have done with a series of Thomas biplanes, testing a number of identical machines under different conditions of loading. One, for example, could be tested with the center of gravity of the load far back, corresponding to high-speed conditions, one with the c.g. in its farthest forward position, and still another with the longitudinal axis vertical, representing the conditions in a vertical dive where there is no lift and a large part of the weight of the airplane is balanced by the drag of the wings. Since it is undesirable and usually quite impracticable to break up more than one airplane of a given type, some other solution has to be adopted as a compromise between counsels of perfection and those of expediency.

The customary method has been to mark the wing chord off into two or three sections and to assume the unit loading to be uniform within each of those sections, the unit loadings in the several sections being so proportioned that the center of gravity of the load is from .35 to .4 of the chord back of the leading edge. An attempt will be made here, as already noted, to build up from fundamental data a method of sand-loading which will correspond as nearly as possible to all of the worst conditions experienced in flight. During the derivation and elaboration of that method the present practice will be entirely disregarded, in order that there may be an independent check on the empirical rules for loading which were devised by the French technical authorities and which are now generally used in America.

The two worst load conditions to which a cellule is submitted are:

(a) the case of coming out of a dive or performing any violent manoeuvre in which the angle of attack passes close to the angle of maximum lift coefficient while the speed still remains in the neighborhood of the limiting speed in a vertical dive. Under this condition the resultant force on the complete cellule is usually inclined slightly forward of the perpendicular to the wing chord, and the load on the drag truss is therefore negative unless there is a large amount of stagger.

(b) the case of a vertical, or nearly vertical, dive at limiting speed. In this instance the weight of the airplane is completely balanced by its drag, the propeller thrust being ineffective at such high speed. In most airplanes, the wing drag and the parasite resistance of the interplane bracing make up from 50% to 65% of the total resistance at the angle of zero lift. The figure will be taken as 60% in this analysis. Since the angle of zero lift does not coincide with that of zero wing moment there is a considerable diving moment on the wings during a vertical descent. This is balanced by a downward force on the tail, and gives rise to an upward load on the rear wing truss, a downward load on the front one.

As a first step, it is necessary that some estimate be made of the maximum stresses which the airplane will be required to undergo in flight under the two critical conditions just described. Any load in excess of the figures thus determined which is sustained on sand-load can then be considered as a material factor, or true factor of safety. In (a) the maximum loading can be determined by measuring the normal accelerations in flight. A large number of tests on pursuit planes, made at the Royal Aircraft Establishment have failed to show any accelerations in excess of 4.2 g, and it may fairly be assumed that this value would not be exceeded in any manoeuvres likely to be carried out by the ordinary pilot, although the theoretical limit of the dynamic load factor when the elevator is pulled up hard very suddenly is much higher than 4.2, ranging from 7 to 13 in different machines and by different methods of computation. The center of pressure is close to its farthest forward position at the instant when the dynamic load attains its maximum value, the angle of attack then being close to the burble point, and this position may be taken as .3 of the chord from the leading edge, this being the mean figure for a number of commonly used wings. The front lift truss therefore carries the larger part of the lift load. The vector of resultant force may be considered to be inclined 2° forward of the perpendicular to the chord of the wings. The component of load in the plane of the wings is then directed forward, and is carried by the anti-drag wires, unless there is more than 2° positive stagger. Since most airplanes built at the present time have some positive stagger, the forward thrust on the wings at large angles need seldom occasion any concern. The only condition under which there is a forward thrust on a wing in a machine with 20% positive stagger, and it is then due to the forward components of the stagger wire tensions which distribute the vertical loads between the two lift trusses, and appears only in the upper wing.

In a dive at limiting velocity the drag acting directly on the wing truss may, as already noted, be taken as 60% of the weight of the airplane, this 60% being distributed approximately equally between the drag of the wings themselves and the parasite resistance of the interplane bracing. These figures apply only to biplanes and triplanes in which the interplane bracing is of normal type, and they should be materially modified for machines in which there is little or no external bracing of the cellules and also for those in which there is a multiplicity of bodies or nacelles.

The moment coefficient of a biplane, referred to the leading edge of the mean chord, may be taken as .03 (in absolute units) at the angle where there is no normal force on the combination, this value of the coefficient seldom being exceeded under the conditions specified. The drag coefficient for this angle was similarly found to be .02. Since, as noted above, the wing drag is about one-third of the total drag, the coefficient of total drag may be taken as .06. The equation of motion along the longitudinal axis of the airplane and the equation for moments about the front spar of each wing (assuming the moment coefficients to be the same for the upper and lower wings) may then be written:

$$\rho D_C^1 AV^2 = W = .06 \times .00238AV^2 = .00014AV^2$$

$$M = M_C \times C \times \rho AV^2 = .03C \times .00238AV^2 = .00007CAV^2$$

where D_c^1 is the drag coefficient for the airplane as a whole, A the total area of supporting surface, V the limiting speed of dive, C the chord of the wings, and W the weight of the airplane. Solving for AV^2 from the first of these equations and substituting the value thus obtained in the second,

$$AV^2 = \frac{W}{.00014} \qquad M = \frac{.00007}{.00014} \times C \times W = .5\ CW$$

This is the equation for moment about the leading edge, but, since there is no normal force, the moments about all points lying in the chord of the wing will be equal, and the equation just given can equally well be taken as representing the moment about the front spar. If this is done the loads and reactions in the front truss will have no effect on the moment about this axis, and the total moment can be equated to the product of the total load carried by the rear truss by the distance between the spars. If b be taken as representing the distance between spars and F is the total load in the rear lift truss

$$F = .5\ W \times \frac{c}{b}$$

Since there is no resultant normal force on the wing cellule except the very small one (about .15W) required to counterbalance the down load on the tail the downward load on the front truss is nearly equal in magnitude to the upward load just found for the rear system.

The maximum upward load on the rear lift truss may not occur exactly at the angle of zero normal force, but it is usually very close to it. It can readily be shown that, if a vector diagram be plotted for the cellule alone, if a line be drawn connecting the front spars in the two wings, and if a point be found on that line which divides it into segments whose lengths vary inversely as the areas of the wings to which they are adjacent, the force in the rear truss is greatest, assuming the speed always to be that of steady motion, for that angle whose vector is at the greatest perpendicular distance from the point so found. This condition is generally satisfied by an angle about one-half degree larger than that of zero normal force, but the difference between the maximum load and that found by the formula already given for the angle of zero normal force is only a few percent - not enough to take into account. If, on the other hand, the speed remains constant, the maximum force in the rear truss would be found at an angle near the burble point.

In actual fact, the speed neither remains constant nor does it vary in such a manner as always to correspond to the speed of steady flight. As the angle of attack increases in flattening out of a dive the speed falls off, slowly at first and then, as the drag coefficient runs up with increasing angle, more and more rapidly. For example, a theoretical analysis, made by the Airplane Engineering Department, U.S.A., of the motion of a JN in flattening out showed that the speed decreased only from 179.95 ft. per sec. to 179.89 while the angle of attack rose from -3° to $+3^\circ$. A further increase of angle to 9° was accompanied by a drop of speed to 179.00, and, by the time the angle reached 15°, the speed had fallen to 174.2 ft. per sec. The actual changes in speed would undoubtedly be more rapid than this, the assumptions on which the computation was based being quite different from the conditions which normally exist in flight, but the proportional variation of the rate of change of velocity is probably substantially correct. The worst load in the rear truss therefore comes shortly after the flattening out begins, when there has been a distinct change of angle from the diving attitude but very little drop of speed. It would not be easy to compute the magnitude of the maximum force in the rear truss or the angle at which it occurs, and an approximate allowance must be made.

Since the spars are usually separated by about 55 percent of the chord length the maximum load carried by the rear truss during a vertical dive is approximately equal to .9W, where W is the total weight of the airplane, in most types. For the reasons stated in the preceding paragraph it is best to allow as an absolute maximum that the lift load in the rear truss may rise to two and a half times that given by the formula, but any such rise will be accompanied by a reduction in the load in the drag truss. The latter will never exceed the value which it has in the limiting vertical dive, and which has been taken as .6W. This figure is still further reduced by the fact that the weight of the wing truss acts in direct opposition to the drag when diving. Taking the weight of the cellule as 15% of the total weight of the machine, the net longitudinal force is reduced to .45W.

With the normal spacing of the spars (front spar at about .12, rear one at about .65, of the chord from the leading edge) the front lift truss will carry 66% of the total load when the c.p. is .3 of the way back on the chord. If the maximum total load in that condition is 4.2W the load on the front truss will be 2.8W. To get the net loads in the lift trusses must be reduced by 15% to allow for the weight of the cellule.

Summarizing, then, the maximum net loads to be encountered are:

2.4 W in the front truss.

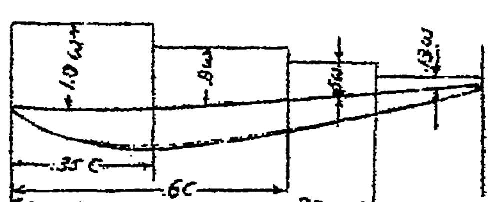
2.2W in the rear lift truss.

0.45W in the drag truss (plus any loads which result from the inclination of the lift bracing in a staggered machine). 0.45 is strictly correct only for an orthogonal cellule.

If a single sand-load test is to be made the load should be so distributed that all three of these truss systems will reach their maximum normal loads at the same time, and any additional load which the structure carries after reaching this point is a true margin of protection in all the trusses against deterioration, non-homogeneity, or initially inferior quality of some of the members entering into the structure or of the manner of their assembly. The imposition of the maximum load in all trusses at the same time is a very severe test, as the actual tendency, when one system of trussing is heavily stressed and another is not, is for part of the load to be transferred from the heavily to the lightly loaded portions of the structure through the stagger wires and other secondary bracing members. It is, however, the best that can be done if only a single test is to be made, and it is better that the conditions of a sand-load test should err on the side of severity rather than on that of leniency. The only other serious error which arises from this attempt to combine all loads on a single test lies in the combination of a load in the drag truss with the maximum force on the front lift members, a combination which never occurs in practise. The result is that the stresses imposed on the lower front spar are greater, those on the upper front spar less, than they should be, but this is a fault which is inherent in the very nature of a sand-load test and which cannot be entirely avoided. The most that can be done is to bear in mind the nature of the error and to be particularly watchful for any signs of yielding of the upper front spar, since any weakness which that member may manifest in the sand-load test will be accentuated in flight. A tearing out of the lower front wing hinge at slightly less than the desired load factor may, on the other hand, be regarded with some leniency, as the force at that point during the sand load is greater than it ever would be in flight.

Having decided what combination of loads is to be simultaneously imposed on the various trusses, it remains only to determine in detail the distribution of the sand along the chord to accomplish the desired end.

If the spars are assumed, as before, to be at .12 and .65 of the chord, the center of gravity of the applied load must lie at .37 of the chord from the leading edge. This position of the center of gravity is given by the load distribution shown in Fig. 1, a distribution approximating the mean pressure distribution over the wings.



ω is a factor determined to give the correct total load per running foot.

So far, the load distribution to which this reasoning leads is closely similar to that which has been most commonly used.

It is necessary to consider next the angle at which the cellule must be inclined during the test in order that the maximum load specified for the drag truss may occur at the same time as the maxima in the lift trusses.

The total net load to be applied to the lift trusses is 4.6 W, assuming that the cellule weight is 15% of the total weight, while that in the drag truss, neglecting the effects of stagger on the drag bracing, is 0.45 W. The cellule should therefore be inclined at an angle whose tangent is 0.45. This angle is 5°.6. This is materially less than that sanctioned by present practise, which usually, almost universally, embodies an angle of 15°. The choice of 15° is based on the assumption of an L/D of 3.8 for the cellule at the angle of maximum horizontal speed. The assumption is substantially correct, but the conclusion as to the necessary inclination is false, as it must be remembered that, while the drag load in horizontal flight at maximum speed is very near to the maximum which can ever be encountered, the lift load on both front and rear trusses can be very much increased over the value then experienced. In fact, as has already been pointed out, if the pilot could flatten out of a dive so abruptly that the limiting velocity was maintained undiminished while the whole range of angles of incidence was covered the maximum loads on the front and rear lift trusses would occur almost simultaneously and very nearly at the angle of attack giving the maximum lift coefficient. It is recommended, in view of the results of this analysis, that all cellules be set up for testing with the mean wing chord at an angle of 6° to the horizontal, the trailing edges of the wings being lower than the leading edges. In machines which are entirely internally braced this angle may be reduced to 4^{o}, as the drag of the wing assembly in such a case is a smaller proportion of the total drag than when there is interplane bracing.

In applying the sand, deduction has to be made to allow for the fact that the lift of the wings in flight is partially balanced directly by the weight of the cellule, and that the "apparent weight" of the cellule increases, in general, at just the same rate as does the lift. When the dynamic load is 4.2 W the cellule will be pressed downward with a force equal to 4.2 times its own weight, and this load will always be distributed between the spars in the same proportion, whatever ma y be the angle of attack. The assumption has thus far been made that the cellule weight is 15% of the total, but this assumption should be abandoned and the actual weight of the cellule be taken into account in computing the amount of sand to be used . Since a sand load of 5.4 W, distributed in the manner which has been specified, has the same effect on the front truss as has a dynamic lift load of 4.4 W at a large angle of incidence, the effect of the dynamic load on the cellule itself can be simulated by reducing the applied sand load by the amount 4.2 W', where W' is the weight of the cellule. This method rests on the assumption, which always approximates closely to the truth, that the C.G. of the cellule is at the same distance back from the leading edge (37%) as is the C.G. of the sand load. Since $\frac{5.4}{4.2}$ equals 1.29, the formula for the amount of sand to be added to give loads in all trusses which will be equal to those corresponding to a dynamic factor of one is:

$$W_s = 1.29W - W'$$

where W_s is the total weight of sand used. In applying the first load a further reduction of W' is made in the amount of sand, since the inverted cellule itself throws a load on the lift bracing. These figures, as already noted, apply primarily to fast manoeuverable single-seaters.

In general, the procedure in deciding on the amount of load which an airplane of any particular type must carry on static test is, first to decide on the maximum dynamic load which the machine is likely to be called on to withstand, taking due account of the probable uses and conditions of operation, whether or not stunting will be necessary, etc., then to increase this by a proper factor of safety, material factor, or depreciation factor, the magnitude of which depends on the nature of the structure and the conditions under which it is likely to be used, and finally to apply the formula given above to find the actual weight of sand which must be used.

DISTRIBUTION OF LOAD ALONG THE SPAN.

The method which has sometimes been used, of carrying the full load out to near the tips of the wings and then removing the load entirely from the tips, is unsatisfactory in that it reduces the bending moments in the overhanging portions of the spars below their proper values, although the shears at the last panel point are probably approximately correct. If, on the other hand, the load is adjusted to make the bending moments right the shears are wrong. It is more accurate to make a moderate reduction of the load at some distance from the wing-tips. It is recommended, in view of experiments which the British have conducted on the pressure distribution over the wings of models, that the unit loading in each portion of the chord should be reduced by one-third on a portion of each wing extending one-half chord length in from the extreme tip.

DISTRIBUTION OF LOAD BETWEEN SUPERPOSED SURFACES.

In biplanes, the unit loading on the lower wing (lower in normal flight, upper when in position for sand-loading) should be 80% of that on the upper. In triplanes, the unit loading on the lower wing should be 80% of that on the upper, that on the middle wing should be 75% of that on the upper. These ratios are based on the experiments carried out at Mass. Inst. Tech., by Comdr. Hunsaker, and on some more recent work at the National Physical Laboratory (R.& M. 196, T.1127), and are the mean ratios of the lift coefficients of the several wings at the angle of maximum lift.

ALLOWANCE FOR AILERON LOADS.

In order to allow for the effect of the use of ailerons when manoeuvering the mean loading on the ailerons and, on the portions of the wings immediately in front of them, clear to the leading edge, should be 20% greater than it would be if the ailerons were not there. The aileron effect at high angles is more than this when coming out of vertical banks, but this need not be taken into account, as the aileron load falls almost entirely on the rear truss, and the rear truss is worst stressed during very steep dives, when aileron movements and loads would be small.

TREATMENT OF TAPERED WINGS.

Each element of a tapered wing should be treated as if the wing had a constant chord equal to the chord at that element. That is the wing should be divided into strips by lines which are everywhere a distance from the leading edge equal to a certain fixed proportion (.35, .60, etc.) of the chord at that point. In making the reduction of load for one-half chord length at the tips the chord length should be taken as that at the inner end of the strip on which the loading is reduced. The area of reduced loading should therefore be trapezoidal in form if the wing-tips are square, and should have an altitude equal to half the length of its longer base.

TREATMENT OF MULTI-ENGINED MACHINES OR THOSE WITH MORE THAN ONE FUSELAGE OR NACELLE.

Wherever there is a concentrated weight outside the fuselage, as at an engine nacelle, an upward load, proportional in magnitude to the product of this concentrated weight by the load factor, must be applied by passing a rope from the point of application of the concentrated load up over a pulley and attaching weights at the end of the rope. These weights must, of course, be increased by a properly proportioned increment as each new load is added on the wings. The alternative method sometimes used, of supporting a twin-engined machine at the two nacelles and applying an upward load by a rope attached to the center fuselage is not correct, as the nacelles, standing by themselves, must have all loads applied at their centers of gravity. The resultant force on the fuselage, on the other hand, may come wherever the supporting reactions bring it, as any torsional moment applied to the fuselage by the wings while in flight is resisted by the tail attached to the fuselage. If the machine has three fuselages, or two fuselages and a nacelle, with the tail carried by the outer two, these outer two should be rigidly supported during the sand-load test and the upward force applied to the central one.

The load applied at the points of concentrated weight should not act vertically, but should be inclined to allow for the longitudinal force acting on the nacelles or fuselages. In the case of nacelles which carry engines, the direction of this longitudinal force is always the same as the direction of flight. When flying horizontally the propeller thrust furnishes the forward component. During a dive, this component is provided by the weight of the nacelle and its contents, the weight always being greater than the drag. The resistance of a typical engine nacelle at limiting speed is very nearly 30% of the weight of the nacelle and power plant, so that the residuary longitudinal force is 70% of that weight. Since the maximum normal force to which the nacelle may be supposed to be subjected is 4.2 times its own weight, the rope through which the upward load is applied when making a static test should be inclined forward of the perpendicular to the chord by an angle which has the tangent $\frac{0.7}{4.2}$. This angle is 9°5. Since the chord is inclined 6°, the rope should lead upward and forward at an angle of 3°5 from the center of gravity of the nacelle.

As in the case of the weight of the cellule proper, the upward load applied to allow for the dynamic load on the nacelles should be only 80% of the actual product of their weight by the dynamic load factor, since that factor is increased 25% to allow for the abnormally even distribution of the load.