COMPARISON OF THE STRUCTURAL DESIGN REQUIREMENTS FOR AIRPLANES WITH THE LOADS OBTAINED IN FULL SCALE PRESSURE DISTRIBUTION TESTS

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Comparison of the Structural Design Requirements for Airplanes with the Loads Observed in Full Scale Pressure Distribution Tests

[Signature]
Rear Admiral Chamberlain
Chief of Staff

[Date]
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# TABLE OF CONTENTS

<table>
<thead>
<tr>
<th>Section</th>
<th>Page</th>
</tr>
</thead>
<tbody>
<tr>
<td>I. Introduction</td>
<td>1-2</td>
</tr>
<tr>
<td>II. High angle of attack</td>
<td>3-8</td>
</tr>
<tr>
<td>III. Low angle of attack</td>
<td>9-10</td>
</tr>
<tr>
<td>IV. Inverted flight</td>
<td>10-11</td>
</tr>
<tr>
<td>V. Rolls</td>
<td>11-12</td>
</tr>
<tr>
<td>VI. Dive</td>
<td>13-15</td>
</tr>
<tr>
<td>VII. General discussion</td>
<td>15-18</td>
</tr>
<tr>
<td>VIII. Recommendations</td>
<td>19</td>
</tr>
<tr>
<td>IX. Bibliography</td>
<td>19-20</td>
</tr>
<tr>
<td>X. Appendix</td>
<td>21-35</td>
</tr>
<tr>
<td>Specified design loads</td>
<td>21-28</td>
</tr>
<tr>
<td>High angle of attack</td>
<td>28-31</td>
</tr>
<tr>
<td>Inverted flight</td>
<td>31</td>
</tr>
<tr>
<td>Rolls</td>
<td>31-32</td>
</tr>
<tr>
<td>Dive</td>
<td>32-35</td>
</tr>
</tbody>
</table>

(iii)
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(IV)
COMPARISON OF THE STRUCTURAL DESIGN REQUIREMENTS FOR AIRPLANES WITH THE LOADS OBTAINED IN FULL SCALE PRESSURE DISTRIBUTION TESTS

(Prepared by Mendel N. Pack and Capt. Howard Z. Bogert, A. C., Matériel Division, Air Corps, Wright Field, Dayton, Ohio, May 27, 1931)

I. INTRODUCTION

At present, there appear to be two schools of thought in the airplane structural field, at least in this country; the first of which believes more aerodynamic data have been accumulated and are available than have ever been efficiently utilized by the structural engineer; the second of which believes there is little or no really exact knowledge of either the magnitudes or distribution of the loads on an airplane; and which feels that to introduce excessive refinements of analysis as has frequently been recommended by the first group, will be futile; and "lead to an illusory degree of accuracy." (Reference 34, p. 191.)

The authors of the present report belong to the latter school, and feel that an adequate rejoinder to those of the first group is to point out that it is really the province of the aerodynamics experts to inform the structural engineer of the loads upon the structure, so that he can then intelligently proportion the structure to carry these loads. The fact that the same ancient chart for relative distribution of load on the wings of a biplane is still in use by the military services (and one but little better by the Department of Commerce) indicates that the aerodynamics experts have themselves made scant use of this "vast" fund of knowledge.

It appears beyond question that there is a real need for further pressure distribution work, to determine both total loads, and distribution of these total loads to the component parts of the structure. Accelerometer work alone is not sufficient, for the accelerations must be "tied in" with air speed and center of pressure position if they are to be of real value.

The general form of presentation of the data of Report No. 364 (reference 1) indicates that the author believes the experimental results are valuable principally in throwing light upon the load distribution. However, it is felt that such an extensive program, covering, as it does, such a wide range of maneuvers and speeds, will be invaluable in indicating the order of magnitude of the total loads to be expected in the ordinary tactical maneuvers, although not necessarily the worst loads possible of attainment. The present report will deal, therefore, both with the consideration of total loads, and with the distribution of these loads over the structure of the airplane.

Practically all of the pressure distribution work to date has been seriously limited in value; mainly because of inherent inaccuracies of the recording instruments, failure to obtain sufficient data, or an attempt to investigate too many variables at one time. These limitations are admirably pointed out in reference 4, for all such tests made up to the time of publication of that report, and accordingly, only brief comment will be made here regarding the tests mentioned therein, together with some comment on other tests conducted since the publication of reference 4.

In reviewing the pressure distribution work to date, it is desirable to emphasize, at the outset, the necessity of adopting an attitude of wholesale skepticism; which, when once acquired, will effectively guard against the danger of drawing too general conclusions from any of the various reports, or from accepting all of the statements made as gospel truth. The art and science of aviation is progressing so rapidly that the latest experimental data is often obsolete by the time it has been released for general distribution.

As an illustration of this point, some of the earlier pressure distribution reports arrived at conclusions and made recommendations which appear almost laughable in the light of present-day knowledge. In reference 17, page 1, it was found that the tail load always acted up; that the center of gravity position was the most important factor in determining the magnitude of the tail loads, and that the (then) design tail loads were too high and could be reduced; thus saving weight in both the tail surfaces and fuselage. (Reference 17, p. 25.) Although all of these statements have since been disproved, nevertheless, a great deal was learned from these early tests, and some of the other conclusions have since been corroborated by recent more refined researches.

Some of the salient points of the MB-3 tests (reference 9), which are believed to be the most complete set of pressure distribution data obtained before the PW-9 tests, will be briefly indicated. The rearward movement of the center of pressure, along the span, as the tip is approached, is very much in evidence. The lower wing center of pressure is almost always farther back than the upper wing center of pressure. In high speed level flight (145 miles per hour in this particular test), the relative loading exceeds a value of 2.0; it is about 1.5 for level flight at 70 miles per hour and about 1.33 for pulling out of a dive and in a vertically banked turn. One of the conclusions of the report is that the design load factor may be reduced 10 per cent, due to the fact that the wings are not supporting the entire load during longitudinal maneuvers. All of these results are not necessarily correct, but are here offered for observation. Some
of them have been checked at least qualitatively by the PW-9 tests; others, like the 10 per cent reduction in load factor, are at present not given much credence.

The more recent investigations on the pressure distribution over the tail surfaces of the F6C-4 (reference 15) and the PW-9 (reference 1) in various maneuvers have been invaluable in clarifying the knowledge of tail loads and in furnishing a check upon these loads as calculated from a theoretical basis.

The PW-9 study was later extended to include the wings as well, and the whole investigation published as Report No. 364, National Advisory Committee for Aeronautics, 1930, The Pressure Distribution Over the Wings and Tail Surfaces of a PW-9 Pursuit Airplane in Flight, by Richard V. Rhode (reference 1). This is the most complete work on pressure distribution in flight released thus far and the one with which the present report is chiefly concerned. As constant reference will be made to it throughout what follows, it will hereafter be referred to simply as Report No. 364.

A study of the loads obtained in Report No. 364 will be made, and these loads compared with the latest structural design rules, in order to obtain some insight as to the validity of the present rules, or suggestions for their revision.

Any such detailed study and comparison is valuable, for even though it does not lead to definite recommendations for changes in the present design rules, it may at least give some insight as to the sufficiency of the data obtained and its method of presentation, which will be helpful in the conduct of future tests of this nature.

Although Report No. 364 was not released until early in 1931, because of the time consumed in making the large number of flight tests, working up the data, and presenting it in final form, it may now be considered almost as obsolescent, and the same cautionary remarks applied to previous pressure distribution work, as mentioned above, will also apply to the conclusions and recommendations in Report No. 364, even though it is still the latest data available.

It may be appropriate to mention here, that, to the knowledge of the authors, no attempt has been made, by the activity originally requesting the research on the PW-9, to give Report No. 364 the thorough study and analysis it deserves; nor to make available, to the testing agency, suggestions or constructive criticisms which might be of assistance in the prosecution of future work of a similar nature. This may be partially due to changes in personnel occurring in the interval between the initiation of the tests and the publication of the final report; the elapsed time, in this case, being about seven or eight years. One would naturally suppose, from a consideration of the pressure originally exerted to have these tests undertaken, and the continual interest in the rate of progress being made; that the results would have been given precedence over routine work, and been subjected to a most complete and thorough study, by the initiating agency. But for some reason, which is not immediately apparent, no such study has been undertaken.

A full description of the PW-9 airplane, including drawings and photographs, is contained in Report No. 364, and since it is obviously undesirable to repeat all of the data contained in that report, only a 3-view drawing, Figure 14, and a list of major characteristics of the airplane, Table 1, are given in the appendix, on page 21 of this report. For this reason, it is essential that anyone desiring fully to understand the subject matter of the present report, should have at hand, ready for constant reference, a copy of Report No. 364.

As mentioned briefly above, the design tail loads for airplanes have been recently rationalized and standardized (see references 1, 3, 6, 8, 9, 15), so that this report will be found to deal mainly with the loads on the wings. The fuselage is generally designed by conditions other than the aerodynamic loads, but where data are given bearing upon fuselage loads, mention will be made of it in this report, and a brief discussion of the results will be included.

In digesting the data giving the loads obtained in flight, the procedure naturally resolves itself into first computing the required loads from the latest design specifications. This is done for all the major flight conditions; and the design loads thus obtained are compared with the actual flight loads under the corresponding conditions. The major conditions treated in the order in which they will be presented are: (1) High angle of attack, (2) low angle of attack, (3) inverted flight, (4) roll, (5) nose dive, and (6) general and miscellaneous conditions.

Throughout this report, whenever design loads are mentioned, it will be understood, unless otherwise stated, that this design load is equal to the specified "ultimate design load" divided by the factor of safety of 1.5. It is believed that present design requirements are based on an anticipated load factor, plus a margin of 50 per cent to allow for possible imperfections in material, approximations of analysis, and general lack of exact knowledge of the loads. The results of accelerometer tests (reference 7) made several years ago, showed that accelerations slightly under 8 were obtained in fast pull outs of pursuit airplanes. This indicates that, with a high angle of attack load factor of 12 (to which the pursuit requirements had been raised at about the time the tests were made), the factor of safety was therefore 1.5. Accordingly, if the high angle of attack flight load exceeds 12 or 8, that in low angle of attack exceeds a load factor of 6.5 or 4.33, or that in inverted flight exceeds a load factor of 4.0 or 2.67, it can be concluded that the actual flight load has exceeded the "expected" design load. The specified design load factors of 12.0, 6.5, and 4.0, for high angle of attack, low angle of attack, and inverted flight, respectively, are those as given in reference 5, page 8. In all cases, where comparisons between actual flight loads and calculated loads are made the gross span load, or gross beam load, as the case may be, is used (dead weight of wings not deducted), since the actual air loads which are given in Report No. 364 are gross loads and not net loads.
II. HIGH ANGLE OF ATTACK

The high angle of attack condition may be considered as the most important major design condition, as it results in the highest total load on the wing structure, and also since the other design load factors have generally been based upon some fraction of the high angle of attack load factor. For this reason, as well as because this attitude was easy to attain in flight, and could easily be duplicated if check runs were desired, there are more data pertaining to the high angle of attack condition in Report No. 364 than for any of the other conditions investigated.

The critical high angle of attack condition occurs in the later stages of the recovery from a dive, and since it is characterized by the maximum forward position of the center of pressure of the wing, and by extremely high loads, it is usually critical for the front lift truss. This attitude was attained, in report No. 364, by "pull outs" or "pull ups" from dives at varying speeds, the pull outs themselves varying in abruptness, and the speeds varying from close to stalling speed, up to 181 miles per hour. These pull ups were made both power on and power off; and in their entirety, represent a most complete presentation of experimental data; and one in which, because of the simplicity of the maneuver and its ease of duplication, the data are in excellent agreement. This is shown by the fact that, for the medium and high speed pull ups, an examination of the pressure plots and the time history curves, indicates that the maximum forward position of the center of pressure is the same, within the experimental error, in every case (see text, p. 19, and figs. 30 to 55, Report No. 364), and that this average value agrees with the wind-tunnel tests on the PW-9 model cellule.

From a detailed examination of all the pull ups presented (which include all those for which satisfactory records were obtained in the flight tests, figs. 30 to 55 of Report No. 364), it appears that there are six runs which result in overloads in the high angle of attack condition; namely: Run No. 137, abrupt power-on pull up at 181 miles per hour; run No. 196, abrupt power-off pull up at 181 miles per hour; run No. 138, abrupt power-on pull up at 172 miles per hour; run No. 195, abrupt power-off pull up at 159 miles per hour; run No. 222, right roll at 167 miles per hour; and run No. 225, left roll at 183 miles per hour.

These runs were selected from a consideration of the maximum acceleration occurring in the maneuver, coupled with the center of pressure position, at the instant of maximum load, as determined from the time histories. The runs will be taken up individually, and each one discussed in some detail, in the order in which they are listed in the previous paragraph.

In run No. 137, the highest acceleration was attained of any presented in Report No. 364—a value of 9.0 g, at 0.92 seconds after the start of the maneuver. The actual flight spar loads, as calculated from the span load curve (fig. 29, p. 29 of Report No. 364) and the center of pressure positions (fig. 28d p. 26 of Report No. 364) are plotted in Figure 1, of this report and compared with the design load curve, based on gross spar loads obtained, using an expected load factor of 8, as previously explained.
It will be noted from an examination of Figure 1 that the distribution of load along the span, in the case of the upper front spar, does not check very closely the design distribution. However, the overloading of the tip section (from the outer strut point outboard to the tip) will serve to decrease the mid-span bending moment, and as most of the remaining overload occurs on the center section or closely adjacent to it, the actual overload is not as great as would be guessed from a consideration of the maximum acceleration alone.

In the case of the lower front spar, the flight load is well within the design curve, and follows roughly the same form. The relative loading, as read from the time-history curve (fig. 41, p. 35 of Report No. 364) at 0.92 seconds, is 1.32, as compared with the design efficiency of 1.183. This explains the lack of overload on the front lower spar.

The rear-spar loads are plotted also, on Figure 1, although the front-spar loads are the critical ones for this condition. The increase in load at the tip on the upper rear spar serves to emphasize the fact that the center of pressure is farther back at the tip than it is at sections farther inboard.

Run No. 196 is one of three for which insufficient data were included in the report to permit the calculation of gross spar loads. The data for the three runs were later furnished through the courtesy of Mr. Richard V. Rhode.

In this run, the noteworthy point is the severe overload of the upper front spar (see fig. 2 of this report), considerably greater than that occurring in run No. 137, even though the maximum acceleration was only 7.8 g; just within the expected load factor. This brings out, without the necessity of added emphasis, the importance of the center of pressure position, and the absolute necessity of knowing the center of pressure corresponding to a recorded acceleration or total load, as was mentioned in the introduction.

In this run, again, the lower front spar flight load is well within the design load; even though, due to the irregularity of form of the flight curve, there is a slight peak rising above the design curve. The upper rear spar load, as in the case of run No. 137, shows a peak at the tip, though of less severity.

The relative loading, in this run, as read from the time history (p. 41, fig. 50 of Report No. 364), is 1.37. This would account for a small part of the excess of overload of the upper front spar over that of run No. 137 (where the efficiency is 1.32) but the really impor-

![Figure 2](image-url)

*Figure 2*
loads (see Table 9 of the appendix), it was found that the flight loads were well within the design loads, so the curves were not plotted. The relative efficiency for this run, at the time of maximum load, was 1.27. (Fig. 49, p. 49, of Report No. 364.)

The next two runs considered were a left and a right roll, at 163 miles per hour, and 167 miles per hour, respectively. The reason they were included here is because the manner in which the maneuver is executed results in a high acceleration and a well-forward position of the center of pressure, at the instant of maximum load.

These two runs then, cover what are generally known as "snap rolls"; a maneuver which is entered at fairly high speed, followed by an abrupt use of the elevator to attain the position of maximum lift, with a left, and 0.88 seconds for the right roll, the form of the pressure curves is that of the high angle of attack condition, and the effect of the rolling motion of the wings has not as yet made itself apparent.

The curves for the left roll at 163 miles per hour, run No. 225, are plotted on Figure 4 of this report, and can be seen to be similar in form to those discussed previously. The front upper spar shows an appreciable overload, and the rear upper spar has a very pronounced tip peak.

The front lower spar is practically within the design load, although in this case the load at the tip is in excess of the design load. This local excess is not critical, since it tends to decrease the mid-span bending moment. In this run, the relative efficiency at the time of maximum load was 1.34. (Fig. 71, p. 61 of Report No. 364.)

The curves for the right barrel roll at 167 miles per hour, run No. 222, are plotted on Figure 5, of this report. The difference in form of the upper wing curves of the two runs is probably due partly to the fact that in run No. 225, the loads are those on the up-going wing, and in run No. 222, the loads are those on the down-going wing. A similar consideration applies to the lower-wing load curves, although the difference in shape is less pronounced. (It will be noticed throughout Report No. 364, that the differences in the flight curves for various maneuvers, or for variations of the same maneuver, are always more pronounced for the upper than for the lower wing.)

The upper front spar shows a greater peak on the tip section than for the left roll, as would be expected, but the resulting overload is less. There is scarcely any peak load on the tip of the upper rear spar.

Contemporaneous or slightly later use of full aileron and rudder. It is really a horizontal spin, and the airplane is first stalled in order to give the pilot enough power to execute the maneuver quickly, that is, to utilize the negative aerodynamic damping moment due to roll, which operates above the stall. (Reference 25 describes this maneuver as well as the aileron roll in considerable detail.)

In the case of a slow roll, or aileron roll, the maneuver is usually executed at a much lower speed; the rolling moment being obtained largely with the ailerons, with some help from the rudder at the start and finish, but with the rudder used mainly to maintain direction.

To return to the snap roll, as exemplified by the two runs mentioned, it will be noted from an examination of Figures 66a to 67h, pages 54 to 58, of Report No. 364, that at the time of maximum load; 0.88 seconds for the

![Figure 3](image-url)

**Figure 3**

The curves for the right barrel roll at 167 miles per hour, run No. 222, are plotted on Figure 5, of this report. The difference in form of the upper wing curves of the two runs is probably due partly to the fact that in run No. 225, the loads are those on the up-going wing, and in run No. 222, the loads are those on the down-going wing. A similar consideration applies to the lower-wing load curves, although the difference in shape is less pronounced. (It will be noticed throughout Report No. 364, that the differences in the flight curves for various maneuvers, or for variations of the same maneuver, are always more pronounced for the upper than for the lower wing.)

The upper front spar shows a greater peak on the tip section than for the left roll, as would be expected, but the resulting overload is less. There is scarcely any peak load on the tip of the upper rear spar.
FIGURE 4

Gross Spar Loads
High angle of attack
Pan No. 223
Max. Acc. 73 g.
Left roll @ 163 m.p.h.

FIGURE 5

Gross Spar Loads
High angle of attack
Pan No. 222
Max. Acc. 73 g.
Right roll @ 167 M.P.H.

Design rules ————
Flight ————
**Figure 6**

Gross front spar loads
High angle of attack

Design C.P. 26% c=150
Design C.P. 31% c=183
Flight run No. 126

**Figure 7**

Gross spar loads
High angle of attack
Run No. 132
Max. Acc. 5.0 g.

Design rules
Flight run No. 132
Max. Acc. 5.0 g.

Front spar

Rear spar
The lower front spar curve is very similar to that of the left roll. The relative efficiency, at the time of maximum load, is 1.35. (Fig. 70, p. 60 of Report No. 364.)

The dissymmetry of load resulting from the snap rolls, and its effect upon the structure, will be discussed fully under the heading, Rolls, pages 11-12.

After considering the various high angle of attack runs individually, a study was made of the effect of an arbitrary 3 per cent forward movement of the design center of pressure. This would bring it to 28 per cent of the chord, instead of 31 per cent, as is the design location for the Gottingen 436 airfoil. (P. 17, reference 5.) In addition, new design curves were computed based on a relative efficiency of 1.5, instead of 1.183 as used in the previous calculations. These new design curves, based on 28 per cent center of pressure location, and a relative efficiency of 1.5 are plotted against the original design curves (center of pressure 31 per cent; relative efficiency 1.183), and the flight load curves from run No. 196 (fig. 2 of this report). The three curves are shown for comparison on Figure 6 of this report. A rather large increase in the relative loading was used; since this particular value (1.5) gave an increase in load roughly equal to that caused by the 3 per cent forward movement of the center of pressure, approximately 0 per cent in each case. An increase in relative loading from 1.183 to 1.5 on this airplane, everything else remaining the same, would be equivalent to increasing the stagger from 6° to 37°. (Extrapolating the relative efficiency chart given on p. 12, reference 5.)

Since a cursory examination of Figures 60 and 61, of Report No. 364, would cause the reader to view with alarm, the excess of flight loads over design loads, and the flight bending moment which is 135 per cent of the design bending moment, it was thought worth while to plot these loads and bending moments against actual design loads and bending moments, based on a load factor of 8. Accordingly, this is done in Figures 7 and 8 of this report, based on the same assumption of a pin joint at the cabane attachment which in itself would indicate a larger mid-span bending moment than actually occurs in a continuous beam. Obviously, the actual design curves and bending moments are merely ½ or 160 per cent of the design curves and bending moments given in Report No. 364, so that the actual design bending moment is still 25 per cent larger than the flight bending moment.

Some comment regarding the method of calculating the results presented in Figures 60 and 61, page 50, of Report No. 364, is believed necessary. Since these results as given, may be misleading to the casual reader. The flight spar loads in run No. 132 were calculated, and then compared with the design load curve based on the acceleration recorded in flight (in this case a load factor of 5). The comparison is thus seen to be purely a comparison as regards distribution of load, and not as regards total load acting. For comparison of load distribution, it is believed it might have been better to have taken the average of several high angle of attack spar load curves, reduced them to equal areas under the curves, and then compared the resulting average load curve to the calculated design curve also based on the same total load.
III. LOW ANGLE OF ATTACK

The low angle of attack condition is that occurring in the early stage of the pull-out from a dive, and consequently is distinguished by a much lower acceleration, and hence much lower total load than is the case in the high angle of attack condition. The center of pressure is well back, usually taken in the neighborhood of 50 per cent for the common airfoils in use to-day. (See p. 17, reference 5.) For the reasons mentioned, the ultimate design load factor for this condition is usually taken as some fraction of that for the high angle of attack condition. The present Army requirements for pursuit airplanes are for a load factor of 6.5 in low angle of attack, and 12.0 in high angle of attack, as mentioned previously.

![Graph of Spar Loads](image)

A careful study of all the pull-ups recorded, Figures 30 to 55, inclusive, pages 30 to 46, inclusive, of Report No. 364, indicates that none of these maneuvers is critical for the low angle of attack condition, since in each case where the total loads (as measured by the maximum acceleration), are of any appreciable magnitude, the center of pressure has moved considerably forward of 50 per cent (the design center of pressure for the Gottingen 436 airfoil) before the loads come anywhere near the expected load factor of 6.5 or 4.33.

Having eliminated all the pull-ups from consideration this leaves only one run, No. 226, a dive at 260 miles per hour, to be considered. It is understood that a large number of high speed dives were made, but that the records were so erratic, due to the high speed, with the consequent magnification of bumps due to rough, bumpy air; as well as to the inherent imperfections of the recording apparatus; that only this one run was good enough to be included in the final report.

It will be noted from the text (p. 63 of Report No. 364), that "the pull out was normally executed; that is, it was made cautiously with due regard to the speed at which the airplane was traveling, and that it in no wise represents a special test condition." This statement is important, since no time history is given, from which the control movement against time could be judged. The point chosen was that of maximum acceleration (3.6 g) which occurred early in the pull out, and is a fair indication that the pull out was much more gentle than those made at 181 miles per hour, for example.

The spar loads for this run are plotted against the design load curves, based on an expected load factor of 4.33, on Figure 9. It will be seen that the upper rear spar flight load is well under the design load, and that the only overload (since this condition is not the critical one for the front spars) is the lower rear spar. It should be emphasized that, if the design relative loading had been about 1.35 (the average relative loading occurring in flight in the high angle of attack condition) the overload would have been even more severe. The relative loading at the point chosen, is given in the text as 1.02, which is quite different from
a value of 1.35. This serves to focus attention upon the need for a different relative loading for low angle of attack than for high angle of attack. Probably a relative loading of unity would be conservative for the lower rear spar, but the reverse would be true for the upper rear spar.

It is interesting to speculate as to what the overload would have been, had the airplane been pulled out with the same abruptness as is used in Army pursuit tactics and in the Navy diving bomber tactics; or if the air speed had been “stepped up” to terminal speed, and the pull outs still made with the same degree of abruptness. It is common knowledge that the Navy is now requiring dives to terminal velocity and pull outs with a prescribed minimum loss of altitude, before acceptance of diving bomber and fighter types; and that the Army pursuit tactics require power dives and abrupt pull outs as a part of the regular combat training. The upper echelons in Army pursuit maneuvers usually fly 5,000 feet above the next lower echelon, and when the lower echelon is engaged, the upper comes down power on; and it is reasonable to assume the dives approach terminal speed before the pull out is begun.

There is apparently an error in the text on page 63, Report No. 364, where the statement is made, “the derived curves are based on a load factor of 3.25, which is the low angle of attack load factor divided by the intended factor of safety of 2”. A comparison of the curves of Figure 92, page 79 of Report No. 364, with those of Figure 9, of this report (which are based on a load factor of 4.33) indicates such a close similarity, that it is believed the design curves of Figure 92 were based on this same expected load factor of 4.33.

This conclusion is substantiated by an examination of Figure 18 of the appendix of this report, where the N. A. C. A. load curves and those calculated in this report are compared. On Figure 18, curve “a” is obtained by dividing the ordinates of the corresponding curve of Figure 92 by 3.25, and curve “b,” by dividing these ordinates by 4.33. On the lower spar, only the curves obtained by dividing the ordinates of Figure 92 by 4.33 are shown. From these, it can be seen that the N. A. C. A. curves of Figure 92 were actually calculated on the basis of an expected load factor of 4.33, and not 3.25 as stated.

It is unfortunate that such a wide gap exists between the fastest pull ups (181 miles per hour) and the terminal speed (about 320 miles per hour for the PW-9) with only the one dive at 260 miles per hour in between. However, there is probably some good reason for the lack of data in this range; and it must be remembered that the interest in fast dives and abrupt pull outs had not developed to its present degree at the time the PW-9 flight tests were run.

### IV. INVERTED FLIGHT

The inverted flight design condition is one provided to cover the contingency of the airplane flying upside down, a push down, or a poorly executed normal loop, an outside loop, a “down bump,” or for any other case in flight where a reverse load on the airplane might be experienced. The necessity for its inclusion in the design requirements for military airplanes is generally admitted as a matter of course; but it is difficult for some commercial manufacturers to see why it should be included for transport airplanes.

However, when some thought is given as to the conditions under which a reverse load may be experienced, the necessity for its inclusion is readily apparent. A sudden bump or gust may cause a reverse load on the wings as high as 1.5 g (reference 44) or a sudden, sharp push down on the control stick while in level flight will accomplish the same result.

In Report No. 364, page 63, this latter method is resorted to, because of the difficulty of holding the airplane on its back. It is believed some peculiarity of the loading, or of the load distribution, on this particular airplane, as flown in the tests, may have contributed to this difficulty, as the service pilots have never had any trouble in flying this type upside down.

It is believed that attainment of the inverted attitude by means of a half roll might have been conducive to better results.

The only data given in Report No. 364 for this condition are those for run No. 211, inverted flight at 79 miles per hour; and run No. 215, a push down at 180 miles per hour. Since this latter run shows the pressure distribution to be more closely characteristic of a dive than of inverted flight, it was not considered as belonging in the inverted flight classification.

The flight spar load curves for run No. 211 are plotted and compared with the design load curves, based on a load factor of 1.0, rather than upon the anticipated load factor of 1.5 or 2.67. This was done because the flight loads were of such small magnitude, that, to have plotted them against the anticipated load factor of 2.67 would have made them appear almost insignificant. The only point of interest here is that the lower rear spar load is positive, while the loads on the other three spars are negative, as would naturally be expected.

No other conclusions can be drawn from the curves, because of the low air speed (79 miles per hour) and the absence of any applied acceleration; the curves being included simply to complete the record. (See fig. 10, of this report.)

It is most regrettable that more data were not procured and recorded for this condition, since the recent epidemic of outside loops and inverted aerobatics throws grave doubt upon the adequacy of the present design load factors for the inverted flight condition.

The Navy has recently completed a rather extensive accelerometer study of loads in inverted flight, and although it is understood the results are to be held confidential, the very fact that such a study was initiated is indication that the present load factors are viewed with distrust. It is known that the Army Air Corps is somewhat concerned about this condition also. Here again, the time lag is apparent, since at the time the PW-9 tests were started, outside loops had not been attempted, and upside-down flying was confined mainly to rolling over on one’s back, gliding for a short space (since carburetors for upside-down flying had not been perfected, and the engine always promptly cut out) and then rolling over right side up again.
There is need for considerable research along this line, at the same time that sober thought is given as to the justification, from the purely military standpoint, for some of the inverted maneuvers recently attempted.

![Diagram of roll loads](image)

**V. ROLLS**

In the early stages of this investigation, it was hoped that one of its useful outcomes would be a more rational design requirement for the unsymmetrical roll analysis, or a justification of the present design rules. Several years ago, the design requirements for this condition, which is generally critical for the cabane structure and some of the fuselage members, called for an analysis with 75 per cent of the design load acting on one wing and 60 per cent of the design load acting on the other, in both the high and low angle of attack conditions. After the failure of the wings of an Army observation airplane in a roll, the design requirements were arbitrarily increased to 100 per cent of the design load on one side and 70 per cent of the design load on the other side; the analysis to be performed for the high angle of attack, low angle of attack, and also the inverted flight conditions. It is felt that the inclusion of the inverted flight condition was a step in the right direction, but the boosting of the unsymmetrical load difference from 15 to 30 per cent was perhaps too severe and may have been due to the extreme pressure always exerted upon the Materiel Division, following a structural failure in flight. Since there is no evidence that the mishap was not due to flaws in the material on this particular airplane, or to improper detail design, or careless maintenance, the severity of the design loads will perhaps justify reconsideration.

There are essentially three types of rolls; viz, slow rolls or aileron rolls, ordinary snap rolls, and inverted snap rolls. The first type are generally so mild that the loads incurred are covered under other conditions, and consequently, do not merit especial consideration here. The third type were, until recently, not often performed, but are believed to be severe on the cabane structure and fuselage cross brace members, since they will cause compressive stresses in the cabane which will be augmented by the compressive stresses induced by the side load due to dissymmetry. It is unfortunate that Report No. 364 does not contain data on some inverted rolls, since it is believed that this is a critical maneuver.

Report No. 364 contains comprehensive data on four ordinary snap rolls, viz, run No. 222, right roll at 167 miles per hour; run No. 225, left roll at 163 miles per hour; run No. 220, right roll at 126 miles per hour; and run No. 223, left roll at 120.5 miles per hour. An investigation of the last two runs disclosed that both the total loads and the unsymmetrical loads were much lower than those obtained in the first two; therefore, the discussion will be confined to an analysis of runs Nos. 222 and 225. The total loads in these two runs are of considerable magnitude and have already been discussed under the high angle of attack condition, page 5.
The span load curves which were obtained in flight are plotted on Figure 11 of this report and have merely been copied from Figure 72 of Report No. 364. In Figure 11 the right and left rolls have been combined (with slight corrections for the difference in total load), so that the combination simulates a roll to the right. It can be seen from Figure 15 of this report, that the airplane contained orifices over the entire semispans of the right upper and left lower wings only. As a result of this, a complete actual span distribution is impossible and resort must be had to a synthetic distribution obtained by combining the two rolls. In recapitulation for the sake of emphasis, it will be stated that in the right roll pictured on Figure 11, the loads on the right upper and left lower wings are actual loads; the loads on the left upper and right lower wings are fictitious loads, having really been obtained in a left roll. The use of this synthetic roll as a basis of analysis may be open to question, but a detailed scrutiny of the time histories of both these rolls (given in figs. 70 and 71, Report No. 364) will indicate that the control movements, air speeds, accelerations, etc., are so nearly similar for similar intervals of time, that the results are probably not far from correct. Incidentally, the analysis is based on loads recorded 1.0 second after the start of the maneuver (the time of maximum dissymmetry), and it is also interesting to note that the moment of the right lower wing exceeds that of the left lower wing and thus apparently "opposes the roll."

The design curves represented on Figure 11 of this report, are gross loads based on a factor of 8.0 for the of the weight of the wings to the weight of the airplane in order to obtain the design "net" moment. These two net moments are then directly comparable. The detailed computations are given in the appendix on page 31.

The results of the computations show that the air moment minus the inertia moment is only 29.5 per cent of the design net moment. Therefore, an unsymmetrical design moment of 8.8 per cent difference instead of 30 per cent, would have caused the two net moments to coincide. Incidentally, the latest design rules (reference 6, p. 204) have been cut, for Army bomber and cargo airplanes, to 100 per cent—80 per cent, while the requirements for all other airplanes remain at 100 per cent—70 per cent. The Navy has always retained its 75 per cent—60 per cent ruling, and their airplanes seem to withstand rolls just as well.
The dive condition is characterized by a comparatively low resultant force on the wings (since the airfoil is at or near zero lift) but since the center of pressure is at some distance off the wing, a large moment is produced, resulting in a large down load on the front spar and a correspondingly large up load on the rear spar. However, since this is an equilibrium condition, the airplane traveling at uniform speed and hence without any acceleration or applied load factor, the actual loads are susceptible of more accurate calculation than for any of the other conditions.

The intensive study and research given the problem of tail loads in the last two years, has led to a clarification of the dive condition, so that it can be stated with assurance that the knowledge of dive loads is much greater now, than when the PW-9 tests were undertaken, and that this phase of the problem is, at present, not causing as much concern as some of the other design conditions.

There is only one run, No. 226, presented in the text of Report No. 364, bearing upon this condition, a dive at 260 miles per hour. The flight spar loads for this run are plotted on Figure 12 of this report, and compared with the design curves obtained in several different ways.

The first comparison is based on the old method (reference 5) in which the design ultimate load in the inverted flight condition is assumed acting on the front spar, and similarly distributed; and the up load necessary for equilibrium, found by taking moments about the tail post, is assumed acting on the rear spar.

On Figures 12a and 12b, in order to make the design curves consistent with all the others, the front spar load was calculated on the basis of the anticipated load factor of 4.0 or 2.67, and then the rear spar load necessary for equilibrium was computed. The detailed calculations for these and the other design curves are given in the appendix.

Secondly, the spar loads necessary to equilibrate the maximum normal tail load of 1,480 pounds as given in reference 8, were calculated and distributed as outlined briefly in that report. The design tail load, without the material factor, was used in order to make the results comparable with the other design curves.

Finally, the new sixth edition of the Army Air Corps Handbook was received as the report was nearing completion, so the dive design loads were computed a third time by the revised method given therein (reference 6, p. 174) and these results also plotted on Figures 12a and 12b. This was done after the spar loads and the tail load had been computed, and had been plotted against angle of attack, as shown on Figure 13 of this report.

The smaller curve in the upper left-hand corner of Figure 13, shows tail load against air speed, from which a tail load of 1,000 pounds was determined for an air speed of 260 miles per hour. This is compared with the load of 916 pounds in actual flight test, as given on
FIGURE 13

Gross Spar Loads

Dive at 260 M.P.H.
Run No. 226
Lower Wing

New Army A.C. Method
Method of A.M.A.
Old design rules
Flight

FIGURE 13
page 103, Table 6, of Report No. 364, for run No. 226, and appears as a reasonably close check. However, when the small difference in distance from center of gravity to center of pressure of the horizontal surfaces, between that used in the dive calculations and that determined by the actual center of pressure location for the particular run, is taken into account; the flight moment is 14,350, and the calculated moment is 14,830, which based on the flight moment as the datum, gives a difference of only 3.34 per cent. The detailed computations are given in the appendix.

To return to the flight loads, as given by run No. 226, the same remarks apply as were made for this run under the low angle of attack condition, since it is obvious that, had the airplane been dived to terminal velocity, the resulting moment, and consequently, the resulting spar loads, would have been much higher. This increase can be closely estimated by examining Figure 13 of this report, since because of the close check at 290 miles per hour, it may be assumed the flight loads at terminal velocity would correspond closely with the maximum load plotted on these curves.

However, this may not be an absolutely valid assumption, since it has been pointed out (reference 21) that the calculated tail loads in a dive will generally be in excess of the flight loads, due to the deflection of the wing under the torsional load, which changes the airfoil form and consequently the moment characteristics.

VII. GENERAL DISCUSSION

Having reviewed the major flight design conditions in some detail, it is believed a brief discussion of the question of load factors, and factors of safety, would not be irrelevant; especially because of the confusion as to the latter term, even among experienced aeronautical engineers.

For example, throughout Report No. 364, repeated mention is made of the factor of safety as 2.0, whereas in the present report, the factor of safety used is 1.5, and it is believed the fact is well established that the actual factor of safety is considerably less than 2.0, although in some cases its exact value may be unknown.

In reference 26 will be found the statement, “Accelerometer tests show that in practice the load factors commonly used in designs are about 20 to 40 per cent higher than the maximum recorded loading.” Therefore, in this case, the factor of safety would be 1.2 to 1.4.

Again, at the time the Army Air Corps revised its pursuit high angle of attack load factors upward to 12.0, it was known that accelerations of 7.8 g, (reference 7), had been attained in a pursuit plane, the maneuverability of which, even at that time, was exceeded by other experimental types then in existence. This would give a factor of safety, using the even figure of 8, of 1.5 for the high angle of attack condition.

In reference 35, page 2, the statement is made, “The best that can be done therefore, is to estimate the maximum loads which can possibly occur in all the structural members and so design those members that a small margin of strength is in hand under the worst conditions. Even this small margin of safety is not always obtainable, and especially in the smaller and more easily maneuverable airplanes there are certain evolutions which would cause structural failure if attempted.” And again, in the same reference, page 42, “a small scout airplane may well have a load factor of seven, but a factor of safety of under one; that is to say, it would be possible by clumsy flying to break the airplane in the air.”

In reference 44, page 202, it is pointed out that the “maximum theoretical load factor for most modern airplanes varies from 17 to 24”; and in reference 29, in discussing the impossibility of attempting to design for the “theoretical maximum load,” the author points out (p. 232) that “it is not a good policy to depend entirely on the discretion of the pilot for avoiding excessive loads on the airplane.”

It has been pointed out, in other parts of the references mentioned above, that the safest guide is to design the airplane strong enough so that the load (and accompanying acceleration) necessary to break the airplane would exceed the physiological resistance of the pilot, and in even so recent a publication as reference 6, page 170, that very statement is used as explaining the basis for the design of pursuit airplanes.

In reference 25, the author of the article states that the civilian acrobatic category load factors (7.5 maximum) include a factor of safety of two (a statement which is open to question) and that “it seems reasonable for single-seater fighters to be designed to the same ultimate acceleration that the pilot himself can withstand; this may mean a factor as high as 10.0.”
It is well known that British accelerometer statistical records consistently show lower accelerations for similar maneuvers than do those of this country, and the quotation does not make it clear just what, if any, factor of safety this figure of 10.0 includes. If it be assumed that the author is including a factor of 2.0 here, as he mentions for civilian acrobatic types, this would bring down the "maximum acceleration which a pilot can withstand," to 5.0 g, which of course is absurd. Accelerations as high as 10.5 g have been recorded (reference 15) and, as long ago as March, 1924, accelerations up to 7.8 g (reference 7) were attained. The important point to note, as far as the present discussion is concerned, is that here again the statement is made that the resistance of the pilot should be used as the basis for the design of pursuit types. (It should be mentioned that this article was printed under date of January 31, 1930, and probably written some time previously, and hence does not necessarily represent the present views of the author.)

Such a basis was believed to be a perfectly safe one a few years ago, but in reference 15, the pilot experienced a maximum of 10.5 g, and in Report No. 364, 9 g was repeatedly attained. The increased fund of accelerometer data recently accumulated conclusively shows that pilots' reactions to accelerations are variable over a wide range, not only as between different pilots, but also as regards the limiting acceleration which will make any one pilot "go black," since this limit depends upon the amount of accelerometer work this particular pilot has done. In other words, it is believed that, with practice, can be trained to stand higher and higher accelerations, the upper limit of which is not known at present, but which is certainly high enough to cause failure of any airplane yet constructed.

In tests made with both recording accelerometers, and with indicating accelerometers on the instrument boards of the planes, pilots have found that if they begin to "pass out," at say, 6 g and automatically tend to ease off the pull-out, when the record of the recording instrument is examined (and checked by the "maximum" hand on the indicating instrument) they find they have overshot to a value of 7.5 or 8.0 g, or possibly more, depending on the original speed of the maneuver, and the quickness of the pilot's reaction.

It is believed the foregoing statements are sufficient to show the futility of attempting to base the design load factors upon any such indeterminate datum as the "physiological resistance of the pilot," and, therefore, some other rational method must be adopted. Since it is impossible to design for the "maximum theoretical load," and not desirable to limit drastically the elevator control power for military types, it is believed the only solution is to increase the pilot's knowledge of the strength of the airplane, and his ability to calibrate his physical reactions in terms of the punishment he is inflicting upon the structure.

More accelerometer data should be accumulated, not to find out what is the worst possible load which can be obtained (since it has been demonstrated conclusively that this is very close to the theoretical maximum) but rather to learn the probable range of loads which will be experienced in performing the essential military maneuvers peculiar to the particular type of airplane under consideration. From such a study it will be possible to obtain an "envelope" including the extreme values recorded over a period of time, and this envelope will be the basis for a determination of the design ultimate load factor, after a reasonable factor of safety has been introduced, to take care of material imperfections, etc. It will then be necessary to instruct the pilot as to the structural limitations of the particular type, and require him to fly the plane accordingly.

This may make a good many service pilots, especially of the pursuit school, throw up their hands in despair; but the fact remains that the Army Air Corps has already done this with all but pursuit and training planes, by limiting their diving speed and the maneuvers they are permitted to perform; and that any airplane, thus far constructed, regardless of type, has been susceptible of destruction in the air by improper handling, even though the majority of the pilots never realized this fact.

In addition, the effect on performance of increasing the design load factor must be considered. If an attempt were to be made to design present service pursuit airplanes to take care of the highest acceleration so far observed, 10.5 g (reference 15), and still maintain a factor of safety of 1.5 as at present, the ultimate design load factor would have to be raised to 15.75. Using the PW-9 airplane and reference 24 as a basis of calculation, this would necessitate an increase in structural weight of 23 per cent and a corresponding increase in the gross weight of 6 per cent. Reference 31 does not exactly check these figures, for in that investigation a particular airplane was used, and the increase shown is less than would be indicated from the theoretical approach of reference 24. However, the use of reference 24 is believed to be the more accurate, since the actual increase found in reference 31 depends entirely upon the amount of excess strength built into the airplane, either intentionally or inadvertently, and therefore can not be used safely for any abstract consideration.

Using then, the increase of 23 per cent in structure, or 175 pounds, reference 30 shows the high speed at the ground would be decreased 0.24 mile per hour (assuming all other factors would be kept constant, which is unlikely) and the rate of climb reduced 150 feet per minute. This would decrease the ceiling something over 2,000 feet, which, with the decreased rate of climb, form the important items. So it is at once apparent that the service will pay heavily for any increase in strength caused by an increased load factor; and is then faced with the problem, either of swallowing this loss of performance with a good face or resolving to limit the operation of the airplanes so that this increase in load factor will be unnecessary.

It is not beyond the range of possibility that the maximum theoretical load may even be exceeded, at least on the lower part of the theoretical curve, due to the increase in normal force coefficient when the maneuver is accompanied by a positive pitching velocity, as is always the case in a fast pull-up. (Report No. 364, p. 52.)

Some general remarks will be interjected at this point regarding the text of Report No. 364, the first of which concerns the data given on the level flight runs
and the conclusions to be drawn from them. (Pages 13 and 19 of Report No. 364.) The discussion of the results is well presented although the conclusions drawn, that it will be necessary to compute the load curve by the laborious trial and error method (because of the twist of the wings in flight), is not concurred in. It will be noted that the rigged twist, as shown by Figure 20, page 18, of Report No. 364, is 23° as against $\frac{1}{3}$° torsional deflection, so that even though it be assumed a load factor of 4 may be experienced in the early stages of the pull-out from a dive, the twist due to torsional deflection would still be slightly less than half of the total. The factor of rigged twist will always be an unknown quantity, depending upon the specific rigging calculations, to determine the other half of the twist. It is noted, also, that reference 28, which is used as a reference in Report No. 364, points out that the torsional deflections of biplanes will usually be so small it may be neglected. The deflection of the PW-9 was no doubt greater than for most biplane cellules, due to the absence of flying wires in the plane of the rear spars. For these reasons, it is believed the suggestion for calculating the twist in arriving at the true load curve can be dismissed from consideration in practical design.

The phenomenon of the rearward movement of the center of pressure as the tip is approached (p. 49 of Report No. 364) is much more important, as it will have a pronounced effect on the mid-span moment of the front spar; but caution must be exercised in drawing general conclusions, since the tip shape itself plays so important a part in determining the form of load curve over this portion of the wing. It is confidently believed the series of tests now being run on the Douglas mail plane with a number of different tip forms will lead to more definite knowledge on this point.

It will be noted, in examining the time histories of various pull-ups, that in several cases the tail accelerations are less than those of the center of gravity, indicating high accelerations forward of the center of gravity, and consequently higher loads on the forward portion of the fuselage than may have been supposed. However, no concern need be felt on this point, even though the center of gravity accelerations approach the expected load factor because of the general practice of building excess strength into the forward portion of the fuselage (in the case of single-engine tractor types) to take care of vibration and to guard against fatigue failure. In the case of twin-wing-mounted-engine planes, the maneuverability being much less and the acceleration proportionally less, it is believed that this phenomenon should not cause much concern.

In cases where the tail accelerations are higher than those of the center of gravity this apparent overload is probably automatically taken care of in the actual design, since the weight and inertia loads are not the critical design loads, the rear portion of the fuselage generally being designed by either the maximum horizontal surface loads or the 3-point landing condition. For this reason the last two paragraphs on page 49 of Report No. 364, while bringing out important points, need not be followed in actual design.

The data on spins are discussed on pages 62 and 63 of Report No. 364, and, as there stated, are relatively unimportant from the structural point of view. The time histories of the two spins, Figure 79, page 75, and Figure 80, page 76, of Report No. 364, show maximum accelerations of about 2.0 and 2.5 $g$ for the center of gravity. As might be expected, the tail acceleration is higher in both the right and left spin.

The discussion of the slip stream investigations, page 81 of Report No. 364, is complete in itself and needs no comment other than to mention that the slip stream effect is shown to be less than is generally supposed, the net effect being, in general, to translate the power-off curve upward, parallel to itself, for the power-on condition.

The general accuracy of the test data is indicated by comparing the results of run No. 73 and run No. 75, pages 36 and 37 of Report No. 364. Both these runs were mild power-on pull-ups, the air speed upon entering the maneuver being 110 miles per hour in each case. The average centers of pressure were 38.5 and 38.5, per cent, the relative loadings 0.90 and 1.15 upon entering the maneuver, and reached maximum values of 1.2 and 1.4; the maximum accelerations were 2.0 and 2.9, respectively. These rather large differences are probably accounted for by the fact that the tests may have been run on different days, at different temperatures and different altitudes, and under different atmospheric conditions (as regards bumps and gusts). It is understood the tests now under way at the Langley Memorial Laboratory are being conducted with all the recording instruments housed in an insulated compartment which is heated with the hangar current while on the ground until brought up to the desired operating temperature, and maintained at this temperature while in flight by means of current from an engine driven generator. This elimination of temperature errors, combined with other refinements made as a result of the experience gained with the cruder instruments of previous tests, have enabled the engineers to get results comparable in accuracy and consistency with the best wind-tunnel methods.

The fact that the airplane used in the tests is of pronounced individuality of design, embodying as it does a number of complicated design features, from the structural viewpoint, makes it exceedingly difficult and hazardous to draw any general conclusions from the results of the tests. Of course, this same limitation would be placed upon the tests made on any single airplane, since no two airplanes are exactly identical, but not to such a pronounced extent as in the case of the PW-9.

It is believed the PW-9 airplane was selected for the running of the most complete set of pressure distribution tests thus far published, purely as a matter of expediency. Both the Army and the Navy, at the time, were contemplating the procurement of a number of this type; and as it was the cleanest and fastest pursuit plane constructed up to that time, it was desired to know more about the loads to which such an airplane might be subjected in violent maneuvers and the distribution of these loads.
The fundamental fallacy of such reasoning is clearly shown in the history of this research, since, due to the immense amount of work involved and the time consumed in obtaining the data, analyzing them, and writing the report in final form, procurement of this type had ceased and it was practically obsolete before the published report was available to the activity originating the project.

The value of these tests is considerably increased, nevertheless, by the fact that complete model pressure distribution tests were run on the PW-9 cellule before the full flight tests were undertaken. A comparison of the two will show a close check in some instances, and in other cases will indicate that model testing can never wholly replace full flight testing. As an illustration, the average center of pressure location on the upper wing as determined from the pull-ups (p. 19 of Report No. 364) agrees with the value obtained in the wind-tunnel tests (reference 12), but does not check the monoplane tests of the airfoil used. But although the center of pressure location checks closely, the normal force coefficient in flight is considerably greater due to the positive pitching velocity, a fact which, of course, would never have been discovered in the wind tunnel.

The check on the lower wing center of pressure is not as good (p. 19 of Report No. 364), but in this case, exact comparisons are not strictly valid since the wind tunnel lower wing model was extended to the plane of symmetry, and hence has a greater span, proportionally, than the full-scale wing. However, the lower wing normal force coefficient is in good agreement with that determined from the model tests (p. 52 of Report No. 364) and as pointed out in this same part of the text, the comparison between the model and full-scale results discloses the important effect of pitching velocity upon the relative loading at maximum load. Were either one of the tests lacking, this information would not have been disclosed.

Under the discussion of Roll (p. 53 of Report No. 364), mention is made of the delayed stall of the lower wings of a biplane, as indicated by model tests, being checked by the full-flight data on rolls.

From the above brief discussion, it is concluded that some of the most important and valuable points brought out in Report No. 364, would not have come to light, or would not have been possible of such definite assertion, had the model pressure distribution data been lacking.

It is also desired to call attention to the fact that, throughout this report, whenever the relative efficiency obtained in flight is mentioned, the reader should bear in mind that this is the ratio of the load per square foot on the right upper wing to the load per square foot on the left lower wing. Differences in rigged incidence, inbuilt twist, etc., between the right and left wings may give a slightly erroneous impression as to the magnitude of the efficiency. Consequently, the relative efficiencies given herein should serve for the purpose of comparison with each other, rather than as absolute values.

In order to eliminate the complications introduced by the structural features of an airplane similar to the PW-9, and to guard against the danger of attempting to investigate too many variables at one time (as is pointed out in reference 4, p. 30), it is believed some logical full-scale pressure distribution test program covering a period of years, should be thought out and agreed upon by the several services, so that each item will "tie in" with, and bear a definite relation to, every other one.

The following general recommendations are offered, with the warning that previous comments regarding the individuality of design of the airplane must be kept constantly in mind:

1. The specified center of pressure of airfoils, for the high angle of attack and inverted flight conditions, should be arbitrarily moved 3 per cent farther forward. This should be done for a monoplane, the upper wing of a biplane; and to be conservative, for the lower wing of a biplane also.

2. The increase in normal force coefficient due to positive velocity in pitch, should be taken care of by assuming a new relative efficiency for the high angle of attack condition which will be nearer the true value as determined by full flight tests.

3. A different relative efficiency should be used for low angle of attack, from that used for high angle of attack. This will have to be determined on some arbitrary basis similar to the British method (see reference 28) until data are available for a more rational determination.
(4) Accelerometer studies should be made, on various service types, to determine the range of probable accelerations encountered in the maneuvers necessary for the proper performance of the tactical mission of the particular type, and when this "envelope" of load factor values has been determined, use should be made of this as the basis for the design ultimate load factor; and then maneuvers of the airplane should be restricted to those necessary for the performance of its tactical function.

(5) It is recommended that every service flying field be furnished at least one pursuit airplane, equipped with an indicating accelerometer on the instrument board; in order that service pilots may be able to calibrate their physical reactions in terms of the loads being imposed upon the airplane structure. Each pilot should be required to do a certain minimum amount of accelerometer flying each year, just as a certain minimum amount is now required for night and blind flying. The cost of such a program would certainly be less than the cost of equipping an airplane for blind flying; and would perhaps yield results of as great value.

(6) Adopt new roll, or unsymmetrical load requirements, of less severity than the present ones, if further tests corroborate the results obtained on the PW-9.

(7) Adopt the suggested test program, given on page 18 of this report, or one as closely approaching it as practical considerations permit. The question of cost should be a secondary consideration, since data of this nature are of such fundamental importance that they should be obtained, even though they may be expensive.

IX. BIBLIOGRAPHY


Reference 24: Air Service Information Circular No. 318. Effect on Variation in Load Factor on Structural Weight of Wings.


Reference 28: Handbook of Strength Calculations, Air Ministry, Air Publication 970 (His Majesty's Stationery Office), 1930.


X. APPENDIX

TABLE 1.—Characteristics of PW-9 airplane

<table>
<thead>
<tr>
<th>Characteristic</th>
<th>Values</th>
</tr>
</thead>
<tbody>
<tr>
<td>Span of upper wing, 32 feet, 0 inches.</td>
<td></td>
</tr>
<tr>
<td>Span of lower wing, 22 feet, 53 inches.</td>
<td></td>
</tr>
<tr>
<td>Gap, 82 inches.</td>
<td></td>
</tr>
<tr>
<td>Stagger at wing roots, 9 inches.</td>
<td></td>
</tr>
<tr>
<td>Dihedral (upper wing lower surface), 1° 16'.</td>
<td></td>
</tr>
<tr>
<td>Dihedral (lower wing lower surface), 1° 29'.</td>
<td></td>
</tr>
<tr>
<td>Airfoil section, Gott. 436.8.</td>
<td></td>
</tr>
<tr>
<td>C. G. position:</td>
<td></td>
</tr>
<tr>
<td>Above lower surface root section, lower wing, 263 inches.</td>
<td></td>
</tr>
<tr>
<td>Distance from C. G. to center line of elevator hinge, 15 feet, 30 inches.</td>
<td></td>
</tr>
<tr>
<td>Area of upper wing, 263.4 square feet.</td>
<td></td>
</tr>
<tr>
<td>Area of lower wing, 90.8 square feet.</td>
<td></td>
</tr>
<tr>
<td>Total wing area, 241.2 square feet.</td>
<td></td>
</tr>
<tr>
<td>Fuselage area included between lower wings, 12.5 square feet.</td>
<td></td>
</tr>
<tr>
<td>Total wing area including fuselage, 263.7 square feet.</td>
<td></td>
</tr>
<tr>
<td>Area of horizontal tail surfaces, 27.4 square feet.</td>
<td></td>
</tr>
<tr>
<td>Area of vertical tail surfaces, 10.74 square feet.</td>
<td></td>
</tr>
<tr>
<td>Weight of airplane during tests, 2,578 pounds.</td>
<td></td>
</tr>
<tr>
<td>Weight of wing cellule, 363 pounds.</td>
<td></td>
</tr>
<tr>
<td>fuselage area included between lower wings, 12.5 square feet.</td>
<td></td>
</tr>
<tr>
<td>fuselage area included between lower wings, 12.5 square feet.</td>
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<tr>
<td>fuselage area included between lower wings, 12.5 square feet.</td>
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<tr>
<td>fuselage area included between lower wings, 12.5 square feet.</td>
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<tr>
<td>fuselage area included between lower wings, 12.5 square feet.</td>
<td></td>
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<tr>
<td>fuselage area included between lower wings, 12.5 square feet.</td>
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<tr>
<td>fuselage area included between lower wings, 12.5 square feet.</td>
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<tr>
<td>fuselage area included between lower wings, 12.5 square feet.</td>
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APPROXIMATE PERFORMANCE

<table>
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<th>Performance</th>
<th>Values</th>
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<tr>
<td>High speed, 163.5 miles per hour.</td>
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</tr>
<tr>
<td>Rated horsepower of engine at 2,155 revolutions per minute, 431 horsepower.</td>
<td></td>
</tr>
<tr>
<td>Landing speed, 64 miles per hour.</td>
<td></td>
</tr>
<tr>
<td>Climb at sea level, 1,454 feet per minute.</td>
<td></td>
</tr>
<tr>
<td>Absolute ceiling, 28,730 feet.</td>
<td></td>
</tr>
<tr>
<td>Sources of data:</td>
<td></td>
</tr>
<tr>
<td>2. Boeing 3-view drawing of airplane.</td>
<td></td>
</tr>
<tr>
<td>4. Material Division report on weights.</td>
<td></td>
</tr>
<tr>
<td>5. Reference 40.</td>
<td></td>
</tr>
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</table>

TABLE 2.—Location of stations

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<thead>
<tr>
<th>In inches aft of leading edge</th>
<th>B</th>
<th>A</th>
<th>A’</th>
<th>A”</th>
<th>C</th>
<th>D</th>
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<tbody>
<tr>
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<tr>
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<td>Station</td>
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<td>38.0</td>
<td>38.0</td>
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<td>48.0</td>
<td>48.0</td>
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<tr>
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<td>58.8</td>
<td>58.8</td>
<td>54.0</td>
<td>44.8</td>
</tr>
</tbody>
</table>

Station No. 1 is first station aft of leading edge.
Each station represents two orifices, one on the upper surface and one
on the lower surface directly beneath it. The anemometer recorded the
difference in pressure between these stations.
Pressure riba are located on Figure 15.

21
COMPUTATION OF SPECIFIED DESIGN RUNNING LOADS ON SPARS

The loads are computed in accordance with paragraph 11, page 14, reference 5.

Tip length of upper wing=60 inches.

Tip length of lower wing=24 inches.

The area included between the tip of each wing and one-fourth the tip length, will be termed the "noneffective" area of the wing.

The "noneffective" area of the upper panel as planimetered from a one-tenth scale drawing, is 4.33 square feet; that of the lower panel is 0.807 square feet.

Effective area of upper wing=100.4 - 2 (4.33) = 151.7 square feet.

Effective area of lower wing=80.8 - 2 (0.807) = 79.2 square feet.

The mean aerodynamic chords, from which the relative efficiency is determined, are found by the method of paragraph 8, page 54, reference 5.

The upper wing is divided into two sections; the center section, and that outboard of 42 inches from the center line of the airplane.

Area of center section=Ac = 38 square feet

Area of outer panel =A,,= 160.4-38 = 122.4 square feet.

Distance of m. a. c. of center section from center line of airplane=21 inches.

Distance of m. a. c. of outer panel from center line of airplane =110 inches. (Both determined by fig. 11, p. 55, reference 5.)

Taking moments about center line of airplane to find m. a. c.'s of upper wing:

Distance of m. a. c. from center line= e =

\[ e = \frac{(21)(38) + (110)(122.4)}{160.4} = 88.8 \text{ inches} \]

The length of the m. a. c. of the upper wing is determined by the formula:

\[ l = C_p \left( \frac{C_o - C_i}{H - e} \right) \]

where:
- \( C_p \) = m. a. c. of outer panel =60.5 inches.
- \( C_i \) = m. a. c. of inner panel =60.0 inches.
- \( H \) = distance of m. a. c. of outer panel from center line=110 inches.
- \( e \) = distance of m. a. c. from center line=88.8 inches.

then:

\[ l = 60.5 \left( \frac{66.0 - 60.5}{110 - 88.8} \right) = 61.55 \text{ inches}. \]

Because of the irregular shape of the lower wing, the position of its m. a. c. is determined by dividing the wing into 10 equivalent trapezoidal sections, each 12 inches wide (measured along the span), and taking moments of these sections, and the area of the fuselage included between the two panels, about the center line of the airplane. The computations are performed in Table 3.

### Table 3.—Location of m. a. c. of lower wing

<table>
<thead>
<tr>
<th>Section No.</th>
<th>Average of bases</th>
<th>Width of section</th>
<th>Area</th>
<th>Distance from inner base to centroid</th>
<th>Distance to center line of airplane</th>
<th>Moment of area 6X4</th>
</tr>
</thead>
<tbody>
<tr>
<td>(Fuse) 1</td>
<td>60.0</td>
<td>15</td>
<td>560</td>
<td>7.5</td>
<td>7.0</td>
<td>6,750</td>
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<td>2</td>
<td>36.05</td>
<td>12</td>
<td>710</td>
<td>6.0</td>
<td>21.6</td>
<td>15,096</td>
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<td>3</td>
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<td>12</td>
<td>704</td>
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<td>57.45</td>
<td>12</td>
<td>690</td>
<td>5.98</td>
<td>44.98</td>
<td>31,100</td>
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<td>55.70</td>
<td>12</td>
<td>688</td>
<td>5.96</td>
<td>56.96</td>
<td>38,000</td>
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<tr>
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<td>53.50</td>
<td>12</td>
<td>642</td>
<td>5.95</td>
<td>68.95</td>
<td>44,350</td>
</tr>
<tr>
<td>7</td>
<td>56.80</td>
<td>12</td>
<td>619</td>
<td>5.94</td>
<td>80.94</td>
<td>49,400</td>
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<tr>
<td>8</td>
<td>47.05</td>
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<td>595</td>
<td>5.90</td>
<td>92.90</td>
<td>52,500</td>
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<tr>
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<td>42.50</td>
<td>12</td>
<td>566</td>
<td>5.88</td>
<td>104.9</td>
<td>55,100</td>
</tr>
<tr>
<td>10</td>
<td>35.70</td>
<td>12</td>
<td>428</td>
<td>5.77</td>
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<td>11</td>
<td>23.00</td>
<td>12</td>
<td>257</td>
<td>5.33</td>
<td>128.3</td>
<td>36,850</td>
</tr>
</tbody>
</table>

Total: 6,716 600,310

Distance of m. a. c. to center line of airplane=400,310

=59.7 inches.

The length of the lower wing m. a. c. is similarly determined as equal to 55.2 inches.

The following sketch shows the length and positions of the m. a. c.'s:

### Diagram

- **Upper wing root**
- **Lower wing root**

\[ \theta = \text{angle of stagger} = \tan^{-1} \frac{5.48}{52} = 6° \]

Gap = 52

Arithmetical mean chord = \( \frac{1}{2} (61.55 + 55.2) = 59.01 \) inches.

Relative efficiency = \( e = 1.183 \) (from fig. 4, p. 12, reference 5).

For a comparison, the relative efficiency was computed according to reference 3. This class of cellule falls under Case III. The computations follow:

Span, upper wing=32 feet (b1).

Span, lower wing=22.48 feet (b2).

Upper chord (m. a. c.)=61.55 inches (c1).

Lower chord (m. a. c.)=55.2 inches (c2).

Gap=52 inches (G).

Stagger = +6°.

\[ r = \frac{b_2}{b_1} = \frac{22.48}{32} = 0.703. \]

\[ \eta = \frac{r}{\alpha} = \frac{55.2}{61.55} = 0.897. \]

\[ G/c_1 = \frac{52}{61.55} = 0.844. \]
From Figure 6, page 14 (s=0.9); e=1.11.
From Figure 7, page 14 (s=0.8); e=1.10.
Therefore, e for r=1, in low angle of attack = 1.11.
From Figure 14, page 16 (s=0.9); e=1.23.
From Figure 15, page 16 (s=0.8); e=1.21.
Therefore, e for r=1, in high angle of attack= 1.23.
From Figure 1, page 10; C_{a1}^1=1.045.
From Figure 2, page 11; C_{a1}^1=1.10.
Then:
\[ e = r e (\alpha - \theta) + \frac{1 - r}{2} C_{a1}^1 \]
For low angle of attack; e=0.703(1.11) + \frac{1 - 0.703}{2} 1.045
=1.091.
For high angle of attack; e=0.703(1.23) + \frac{1 - 0.703}{2} 1.10
=1.195.
Relative efficiency from Report No. 364, page 49= 1.29.
Relative efficiency from reference 40=1.25.
Both e equal to 1.29 and e equal to 1.25 were originally obtained from the stress analysis which was prepared about 1922 or 1923. The latest design rules, either from reference 39 or reference 5, give values of e which are much lower than either of these. Since this airplane is fundamentally an Army type, all design loads will be based on a value of e equal to 1.183 (computed according to the latest Army method) unless otherwise mentioned. The same relative efficiency will be used in high and low angles of attack.
Gross load on lower wing= \frac{2070}{(1.183) (161.7) + 79.2} = 11.47 lb./sq. ft.
Gross load on upper wing= 11.47 (1.183)= 13.6 lb./sq. ft.
Where the effective areas have been obtained from page 22.

![Diagram](image-url)
### TABLE 4.—Spar locations and center of pressure positions

#### UPPER WING

<table>
<thead>
<tr>
<th>Station</th>
<th>Distance from center line of airplane</th>
<th>Chord length</th>
<th>Front spar location</th>
<th>Rear spar location</th>
<th>Distance between spars</th>
<th>H.A.A. Distance center of pressure to front spar</th>
<th>L.A.A. Distance center of pressure to front spar</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>Inches</td>
<td>Inches</td>
<td>Per cent</td>
<td>Per cent</td>
<td>Per cent</td>
<td>Per cent</td>
<td>Per cent</td>
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<td>74.0</td>
<td>129.2</td>
<td>86.2</td>
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</table>

*NOTE.—Columns 7 and 15 are obtained by subtracting columns 4 and 12, respectively, from 31, the specified H.A.A. center of pressure position. Likewise columns 8 and 16 are obtained by subtracting columns 4 and 12, respectively from 50, the specified L.A.A. center of pressure position. Station 13 on the upper wing and station 9 on the lower wing, are the strut points.

#### LOWER WING

<table>
<thead>
<tr>
<th>Station</th>
<th>Distance from center line of airplane</th>
<th>Chord length</th>
<th>Front spar location</th>
<th>Rear spar location</th>
<th>Distance between spars</th>
<th>H.A.A. Distance center of pressure to front spar</th>
<th>L.A.A. Distance center of pressure to front spar</th>
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<tbody>
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<td>Inches</td>
<td>Per cent</td>
<td>Per cent</td>
<td>Per cent</td>
<td>Per cent</td>
<td>Per cent</td>
</tr>
<tr>
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<td>15</td>
<td>60.0</td>
<td>17.9</td>
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<td>48.35</td>
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</table>

*NOTE.—Columns 7 and 15 are obtained by subtracting columns 4 and 12, respectively, from 31, the specified H.A.A. center of pressure position. Likewise columns 8 and 16 are obtained by subtracting columns 4 and 12, respectively from 50, the specified L.A.A. center of pressure position. Station 13 on the upper wing and station 9 on the lower wing, are the strut points.
### Table 5.—Computation of running loads on spars

#### UPPER WING

<table>
<thead>
<tr>
<th>Station</th>
<th>Load, per square foot</th>
<th>Span load per inch run</th>
<th>Per cent load on rear spar</th>
<th>Per cent load on front spar, lb./in.</th>
<th>Load on rear spar, lb./in. 3X4</th>
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#### LOWER WING

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<th>Load, per square foot</th>
<th>Span load per inch run</th>
<th>Per cent load on rear spar</th>
<th>Per cent load on front spar, lb./in.</th>
<th>Load on rear spar, lb./in. 12X13</th>
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<td>1.43</td>
<td>3.60</td>
<td>73.2</td>
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#### NOTES

The load per square foot is obtained from the sketch on page 23. The span load, per inch run, is obtained by multiplying the corresponding load, per square foot, by the chord in inches and dividing by 144. The per cent load on rear spar is obtained by dividing the "c. p. to front spar distance" by the distance between spars. The results of the table are plotted on Figure 16, page 27.
DISCUSSION OF EFFECT OF CHANGES MADE IN NEW HANDBOOK (REFERENCE 6)

During the time the major portion of the design loads were being computed and compared with flight loads, the latest available design specifications were those contained in the fifth edition of the Army Handbook (reference 5). Toward the close of this report the sixth edition of that volume (reference 6) was issued. Since it is desirable to make all comparisons with the very latest design rules, the sixth edition was carefully perused to note where changes had been made over the previous edition.

Essentially, there were found to be two major changes that would affect the design running loads on the spars. One of these was a slight modification of the effective area used; the other was a change in the specified center of pressure location.

The change in effective area will be discussed first. It will be recalled that the fifth edition termed the "noneffective" area as that area which is outboard of one-fourth of the tip length (p. 15, reference 5). The sixth edition, however, terms the "noneffective" area as one-fourth the area of the tip section (p. 187, reference 6). The new method seems to be an improvement, since the old method did not give the correct total load for curved tips. The gross loads for the new method are computed as follows:

Area of both upper tips = 41.5 square feet (by planimeter).
Area of both lower tips = 10.52 square feet (by planimeter).

Then according to reference 6 —
Effective area of upper wing = 160.4 - 41.5 = 150 square feet.
Effective area of lower wing = 80.8 - 10.52 = 70.28 square feet.

\[ w_{10} = (1.183)(150) + 78.2 = 11.62 \text{ pounds per square foot.} \]
\[ w_{10} = 11.62(1.183) = 13.75 \text{ pounds per square foot.} \]
\[ w_{10} \text{ according to the fifth edition was } 13.6 \text{ pounds per square foot.} \]
\[ \text{Increase} = \frac{13.75 - 13.6}{13.6} \times 100 = 1.1 \text{ per cent.} \]

The specified high angle of attack center of pressure position for the Göttingen 436 airfoil was moved forward 1 to 30 per cent chord, and the low angle of attack specification was moved back 2 to 52 per cent chord. (See reference 5, p. 17; and reference 6, p. 189.)

It will be assumed that the average distance between spars is 80 per cent. Then the effect of center of pressure change will be to increase the high angle of attack front spar design load about 2 per cent, and the low angle of attack rear spar design load about 4 per cent.

The combined effects of effective area change and center of pressure change will tend to counteract each other on the lower spars, but will be additive on the upper spars. The total increase in the upper front spar specified design load is then about 3 per cent; that in the rear spar specified design load is then approximately 5 per cent.

Unless otherwise specified, the design curves given for comparison throughout this report are based on the specifications of reference 5, since the labor involved in changing these loads to conform with the new requirements is not feasible in the time permitted. After all, the differences pointed out are the maximum possible and in themselves are not too great to change the qualitative value of the comparisons. Furthermore, it is felt that the intelligent reader intent on making rigidly accurate quantitative use of the data, can easily perform the necessary transformations with the aid of the above discussion.

COMPARISON OF DESIGN LOAD CURVES

In several places in Report No. 364, there are plotted design load curves for comparison with flight load curves. It was discovered that these design curves differed from the design curves shown in Figure 16 of this report. A careful investigation was conducted in an attempt to ascertain the reason for the discrepancy, but no error could be discovered in the design curves here presented, and there were insufficient data to check the N. A. C. A. design curves. It was not until after a personal conference with Mr. R. V. Rhode, that it was learned that the N. A. C. A. curves were taken bodily from the stress analysis, which was performed eight or nine years ago. Design specifications have changed considerably since then, and naturally the later design rules are more valid for comparison with flight loads. Both sets of curves are plotted in Figures 17 and 18 for comparison. Broadly speaking, the curves are somewhat alike but differ considerably at the tips.

In addition, it will be noted that Figure 92, Report No. 364, is a comparison of low angle of attack flight loads and design loads. In the text of Report No. 364, page 63, it is stated that the design curves are based on a load factor of 6.5/2 or 3.25. This statement is apparently in error, since a check of the curve makes it appear as though it is based on a load factor of 6.5/1.5 or 4.33 instead; or a load factor very close to this. The difference is graphically portrayed in Figure 18, where curve "a" is the "design" curve for the upper rear spar (fig. 92, Report No. 364) divided by 3.25—the supposed load factor—in order to reduce it to a factor of 1.0. Then the same curve was divided by 4.33—the apparent load factor—and resulted in curve "b" Figure 18, which is much closer to the design curve given in this report. The comparison is shown only for the upper rear spars since the difference is at once obvious. Incidentally, the flight loads shown in Figure 92, have been plotted against the authors' design loads in Figure 9 of this report.
TABLE A.—Spar locations and pressure rib positions

(NOTE.—In several subsequent tables where the spar loads in flight are computed, it is necessary to know the spar locations with respect to the chord. These are given below and will therefore be omitted from the subsequent tables.)

### UPPPER WING

<table>
<thead>
<tr>
<th>Rib</th>
<th>Distance from center line of airplane</th>
<th>Chord length</th>
<th>Front spar location</th>
<th>Rear spar location</th>
<th>Distance between spars</th>
</tr>
</thead>
<tbody>
<tr>
<td>A</td>
<td>Inches</td>
<td>Inches</td>
<td>Per cent</td>
<td>Per cent</td>
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</tr>
<tr>
<td>B</td>
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</tr>
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### LOWER WING

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<th>Front spar location</th>
<th>Rear spar location</th>
<th>Distance between spars</th>
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### Table 6.—Spar loads in flight

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<th>Rib</th>
<th>Center of pressure position</th>
<th>Distance from front spar, per cent</th>
<th>Per cent load on rear spar, per foot run</th>
<th>Span load, per inch run, lb. ft.</th>
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### Notes

The loads are those obtained 0.92 second after the start of the maneuver.

Column 2 is obtained from Figure 25, page 26, Report No. 364.

Column 3 is obtained by subtracting column 4 of Table A from column 2.

Column 4 is obtained by dividing column 3 by column 6 of Table A.

Column 5 is obtained from Figure 25, page 26, Report No. 364.

The results are plotted on Figure 1 of this report.
Table 7.—Spar loads in flight
[Run No. 196; abrupt power-off pull up at 181 m. p. h. (H. A. A.)]

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<th>8</th>
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<td>Center of pressure position</td>
<td>Distance center of pressure to front spar, per cent</td>
<td>Per cent load on front spar</td>
<td>Span load, per inch</td>
<td>Load on front spar, lb. fin.</td>
<td>Load on front spar, lb. fin. 6-7</td>
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Notes
The loads are those obtained 0.85 second after the start of the maneuver.

Columns 2 and 3 are obtained from Table 13 of this report.
Column 4 is obtained by subtracting column 4 of Table A from column 2.
Column 4 is obtained by dividing column 3 by column 6 of Table A.
The results are plotted on Figure 2 of this report.

Table 8.—Spar loads in flight
[Run No. 136; abrupt power-on pull-up at 172 m. p. h. (H. A. A.)]

<table>
<thead>
<tr>
<th>Upper Wing</th>
<th>1</th>
<th>2</th>
<th>3</th>
<th>4</th>
<th>5</th>
<th>6</th>
<th>7</th>
<th>8</th>
</tr>
</thead>
<tbody>
<tr>
<td>Rib</td>
<td>Center of pressure position</td>
<td>Distance center of pressure to front spar, per cent</td>
<td>Per cent load on front spar</td>
<td>Span load, per inch</td>
<td>Load on front spar, lb. fin.</td>
<td>Load on front spar, lb. fin. 6-7</td>
<td></td>
<td></td>
</tr>
<tr>
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<td>10.57</td>
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<td>13.6</td>
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<td></td>
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</table>

Notes
The loads are those obtained 0.85 second after the start of the maneuver.

Columns 2 and 3 are obtained from Table 13 of this report.
Column 3 is obtained by subtracting column 4 of Table A from column 2.
Column 4 is obtained by dividing column 3 by column 6 of Table A.
The results are plotted on Figure 3 of this report.
### TABLE 11.—Spar loads in flight

[Run No. 222; right barrel roll at 167 m. p. h. (H. A. A.)]

<table>
<thead>
<tr>
<th>Rib</th>
<th>Center of pressure position</th>
<th>Distance of pressure to front spar, per cent</th>
<th>Per cent load on rear spar</th>
<th>Span load, per foot run</th>
<th>Span load, per inch run</th>
<th>Load on rear spar</th>
<th>Load on front spar</th>
</tr>
</thead>
<tbody>
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<td>7.6</td>
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<td>7.77</td>
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<td>9.6</td>
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<td>39.8</td>
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### LOWER WING

<table>
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<tr>
<th>Rib</th>
<th>Center of pressure position</th>
<th>Distance of pressure to front spar, per cent</th>
<th>Per cent load on rear spar</th>
<th>Span load, per foot run</th>
<th>Span load, per inch run</th>
<th>Load on rear spar</th>
<th>Load on front spar</th>
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</table>

### NOTES

The loads are those obtained 0.88 second after the start of the maneuver.
Column 2 is obtained from fig. 66f, p. 55, Report No. 364.
Column 3 is obtained by subtracting column 4 of Table A from column 2.
Column 4 is obtained by dividing column 3 by column 6 of Table A.
Column 5 is obtained from fig. 68, p. 59, Report No. 364.
The results are plotted on fig. 3 of this report.

---

### COMPUTATION OF NEW H. A. A. DESIGN LOADS ON SPARS WITH ARBITRARY 3 PER CENT FORWARD MOVEMENT OF SPECIFIED CENTER OF PRESSURE, AND ARBITRARY INCREASE OF RELATIVE LOADING TO 1.50

**Gross load on lower wing** = \((1.5) (151.7) + 79.2\)
\[= 2970\text{ lb./sq. ft.}\]

**Gross load on upper wing** = \((9.69) (1.50) = 14.52\text{ lb./sq. ft.}\)

The spar loads are computed in Table 12.
### Table 12.—Running loads on spars (H. A. A.) with center of pressure at 28 per cent and relative efficiency equal to 1.50

#### UPPER WING

<table>
<thead>
<tr>
<th>Station</th>
<th>Load per square foot</th>
<th>Span load per inch run</th>
<th>Distance between spars, per cent</th>
<th>Distance center of pressure to front spar, per cent</th>
<th>Per cent load on rear spar</th>
<th>Load on rear spar lb/in.</th>
<th>Load on front spar lb/in.</th>
</tr>
</thead>
<tbody>
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#### LOWER WING

<table>
<thead>
<tr>
<th>Station</th>
<th>Load per square foot</th>
<th>Span load per inch run</th>
<th>Distance between spars, per cent</th>
<th>Distance center of pressure to front spar, per cent</th>
<th>Per cent load on rear spar</th>
<th>Load on rear spar lb/in.</th>
<th>Load on front spar lb/in.</th>
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<td>1.17</td>
<td>.86</td>
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</table>

#### NOTES

The load per square foot is obtained from the sketch on page 30.
The span load per inch run is obtained by multiplying the corresponding load per square foot by the chord in inches and dividing by 144.
The per cent load on rear spar is obtained by dividing the "center of pressure to front spar distance" by the distance between spars.
Columns 4 and 12 are obtained from columns 6 and 14 of Table 4 of this report.
Columns 5 and 13 are obtained by subtracting 3.0 from columns 7 and 15, respectively, of Table 4 of this report.
The results are plotted on Figure 6 of this report.
The inertia moment of the wings must be subtracted from the unbalanced air-load moment in order to obtain the net moment imposed on the structure in flight.

The resisting inertia moment is computed as follows: Assume that the weight per square foot of the upper and lower wings is constant.

Weight of entire wing cellule = 393 pounds.

Weight of upper wing = 160.4 pounds.

Weight of lower wing = 210.0 pounds.

The curve of acceleration plotted against span is shown as the upper curve of Figure 19 in this report. This curve has been reproduced from Figure 72, page 62, of Report No. 364.

The resisting inertia moment is computed in Table 15.

**TABLE 15.—Computation of resisting inertia moment in a roll**

**UPPER WING**

<table>
<thead>
<tr>
<th>Station</th>
<th>Distance from center line of airplane, inches</th>
<th>Chord length, inches</th>
<th>Dead-weight per inch run</th>
<th>y</th>
<th>Inertia load per inch run 4x5</th>
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</table>

**LOWER WING**

<table>
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<tr>
<th>Station</th>
<th>Distance from center line of airplane, inches</th>
<th>Chord length, inches</th>
<th>Dead-weight per inch run</th>
<th>y</th>
<th>Inertia load per inch run 10x11</th>
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<td>0.183</td>
<td>0.623</td>
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<tr>
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<td>33.8</td>
<td>390.0</td>
<td>0.159</td>
<td>0.553</td>
</tr>
<tr>
<td>11</td>
<td>135</td>
<td>29.6</td>
<td>320.0</td>
<td>0.126</td>
<td>0.483</td>
</tr>
</tbody>
</table>

Notes:
- The dead weight per inch run is equal to 1.63 times the chord in inches divided by 100. The dead weight per inch run is plotted as the center curves of Figure 19.
- Column 5 and 11 are obtained from the upper curve of Figure 19.
- The inertia load per inch run is plotted as the lower curves of Figure 19.
The inertia loads and moments are obtained by mechanically integrating the lower curves of Figure 19.

Area under upper wing curve = 146.23 pounds.
Lateral center of gravity of curve is 10.05 feet from center line of airplane.
Area under lower wing curve = 55.08 pounds.
Lateral center of gravity of curve is 6.98 feet from center line of airplane.

Resisting inertia moment = 2 \times (10.05) \times (146.23) + 2 \times (6.98) \times (55.08) = 2940 + 768 = 3,708 foot-pounds.

Net air moment = 9,620 - 3,708 = 5,912 foot-pounds.

Then the actual net moment is \( \frac{5,912 \times 100}{20075} \) or 29.5 per cent of the design net moment.

This is equivalent to having a design unsymmetrical moment of (0.295) \( \times 30 \) or 8.83 per cent instead of the present specified unsymmetrical moment of 30 per cent.

**COMPUTATION OF DIVE DESIGN SPAR LOADS**

*First Method*

This is by the method of the fifth edition of the Army Handbook (reference 5, p. 17). Briefly, this consists of applying the inverted flight load on the front spar, and a load necessary for equilibrium on the rear spar.

The inverted flight load is obtained by integrating the area under the H. A. A. front spar load curves (same as inverted flight), given on Figure 16 of this report, and multiplying this basic load by the expected load factor of \( \frac{4.0}{1.5} \) or 2.67.

Area under upper front spar curve times 2.67 = 1,900 pounds.
Area under lower front spar curve times 2.67 = 850 pounds.

Distance from upper front spar to tail post = 196.9 inches.
Distance from upper rear spar to tail post = 167.9 inches.
Distance from lower front spar to tail post = 191.1 inches.
Distance from lower rear spar to tail post = 164.4 inches.

Taking moments about the tail post:

Upper rear spar load = \( \frac{(1,900 \times 196.9)}{167.9} \) = 2,230 pounds.
Lower rear spar load = \( \frac{(850 \times 191.1)}{164.4} \) = 990 pounds.

The design dive loads by this method are plotted on Figures 12a and 12b of this report. The front spar loads are the basic high angle of attack front spar loads multiplied by 2.67, and acting down, of course. The rear spar loads are obtained by multiplying the upper front spar ordinates by \( \frac{2230}{1900} \) and the lower front spar ordinates by \( \frac{990}{850} \). The rear spar loads act up.
Second method

This is by the method of A. D. M. 1061 (reference 8), where the maximum normal tail load in a dive is given and the wing spar loads are those necessary to maintain equilibrium.

The maximum normal tail load (p. 25, A. D. M. 1061) is given as 1,480 pounds. It will be assumed that the loads taken by the upper and lower wings will be proportional to their respective areas.

Then the proportion of load affecting the upper wing will be:

\[ 1480 \times \frac{160.4}{241.2} \times \frac{1}{2} = 492 \text{ pounds.} \]

(The factor of \( \frac{1}{2} \) is used because it is desired to get the final load acting on the semispan, while 1,480 pounds is the complete tail load.)

The proportion of load affecting the lower wing will be:

\[ 1480 \times \frac{80.8}{241.2} \times \frac{1}{2} = 248 \text{ pounds.} \]

Then taking moments about upper rear spar:

\[ \text{Load on upper rear spar} = \frac{(167.9)(492)}{29} = 2,842 \text{ pounds.} \]

And taking moments about lower rear spar:

\[ \text{Load on lower rear spar} = \frac{(164.4)(248)}{29} = 1,530 \text{ pounds.} \]

As mentioned on the preceding pages, the area under the inverted flight upper front spar design curve is 1,900 pounds, the area under the corresponding curve for the lower front spar is 850 pounds. Then the design curves by this method are obtained in the following manner:

For the upper front spar, the corresponding ordinates of the former curve are multiplied by \( \frac{2842}{1900} \) for the upper rear spar, they are multiplied by \( \frac{3334}{1900} \) and are then plotted acting in the proper directions.

For the lower front spar, the corresponding ordinates of the latter curve are multiplied by \( \frac{1530}{850} \) for the lower rear spar, they are multiplied by \( \frac{1778}{850} \) and are then plotted acting in the proper directions.

The resulting design curves are also plotted on Figures 12a and 12b of this report.

Third Method

This is the method specified in the latest Army Handbook (reference 6, pp. 174–179).

The resultant forces, components and moments are obtained graphically with the aid of a vector sheaf similar to the one given on page 118 of the Handbook. The results are first computed on the basis of a monoplane wing having the length and position of the mean geometric chord; and the loads are then divided between the upper and lower wings.

Mean geometric chord = M. G. C. = \( C_a A_u + C_b A_l \)

(From p. 65 of the Handbook. The relative efficiency is assumed equal to unity.)

Where:

\[ C_a = \text{upper M. G. C.} = 61.55 \text{ inches.} \]
\[ C_b = \text{lower M. G. C.} = 55.2 \text{ inches.} \]
\[ A_u = \text{area upper wing} = 160.4 \text{ square feet.} \]
\[ A_l = \text{area lower wing plus fuselage} = 93.3 \text{ square feet.} \]

\[ M. G. C. = \frac{(61.55)(160.4) + (55.2)(93.3)}{253.7} = 59.4 \text{ inches.} \]

Front spar location on M. G. C., obtained graphically = 10

\[ \frac{59.4}{59.4} = 16.85 \text{ per cent of chord.} \]

Rear spar location = 38.5

\[ \frac{89.4}{59.4} = 64.8 \text{ per cent of chord.} \]

Distance between spars = 47.05 per cent.

Center of gravity aft leading edge of M. G. C. = 20.25 inches = 34.2 per cent.

Center of gravity below M. G. C. = 9.15 inches = 15.4 per cent.

Distance from center of gravity to center of pressure of tail plane = \( d_{trz} = 178 \text{ inches} \)

\[ \frac{59.4}{253.7} = 300 \text{ per cent.} \]

The loads are computed in Table 16, and the following nomenclature is used:

\[ K_{w} = \text{value of resultant airfoil cross coefficient.} \]
\[ K_{v} = \text{value of airfoil lift coefficient.} \]
\[ K_{b} = \text{value of airfoil beam coefficient.} \]
\[ K_{c} = \text{value of airfoil chord coefficient.} \]
\[ K_{d} = \text{value of resultant airfoil vector to the center of gravity of the airplane.} \]
\[ d_{trz} = \text{distance in per cent of M. G. C. from the airfoil vector to the center line of the front spar.} \]
\[ K_{p} = \text{parasite drag coefficient=0.000072 from page 79 of Handbook.} \]
\[ K_{f} = \text{tail force coefficient=} \frac{K_{w} \times d_{trz}}{d_{trp}} \]
\[ K_{ra} = \text{value of resultant air force coefficient for complete airplane=} \frac{W \times \frac{1}{A}}{K_{ra}} \]

\[ = \frac{2970 \times 1}{253.7 \times \frac{1}{K_{ra}}} \]

\[ A/V^2 = 253.7 \text{ times } V^2. \]

Gross rear beam load = \( K_{ra} d_{g} A \)

Gross front beam load = Gross rear beam load + gross beam load.

1 These coefficients and distances are obtained graphically from the vector sheaf.
The Handbook is not very specific on the exact method to be used in transferring the spar loads from the M. G. C. to the upper and lower spars. It must be borne in mind that at the critical angle of attack (near zero lift), the couple is the predominant and critical force system, rather than the resultant force; that is, the moment is more important than the difference between up-and-down loads. If the distance between spars on the upper or lower wing is different from the distance between spars on the M. G. C., this fact must be taken into account. For example, if the distance between spars on the upper wing is less than the distance between spars on the M. G. C., the loads on the upper wing spars should be more than that indicated by using the factor $P$, and conversely for the lower wing.

It is suggested that this be taken account of by employing two additional factors, $C$ and $C'$, in the following manner:

- Gross upper rear beam load = (gross rear beam load) $(P) (C)$.
- Gross lower rear beam load = (gross rear beam load) $(1-P) (C')$.

where:

- $C$ is the distance between spars on the M. G. C. divided by the distance between spars on the upper wing M. G. C.
- $C' = \frac{C}{29.5} = 0.983.$

and $C'$ is the distance between spars on the M. G. C. divided by the distance between spars on the lower wing M. G. C.

$C' = 26.7 = 1.068.$

Gross upper front beam load = $(P)$ (gross beam load) - (gross upper rear beam load).

Gross lower front beam load = $(1-P)$ (gross beam load) - (gross lower rear beam load).

From the curves of Figure 13 it is seen that the critical gross rear beam load occurs at $-4.7^\circ$ and has a magnitude of 10,150 pounds. Therefore:

- Gross upper rear beam load = $10,150 \times 0.633 \times 0.983 = 6,310$ pounds.
- Gross lower rear beam load = $10,150 \times 0.367 \times 1.068 = 3,970$ pounds.
- Gross upper front beam load = $(0.633 \times 1.064) - 6,310 = 5,295$ pounds.
- Gross lower front beam load = $(0.367 \times 1.064) - 3,970 = 3,331$ pounds.

These gross beam loads are each divided by two since they are plotted over the semispan only. Then these loads are distributed over the spars in a manner similar to that described for the first two methods; that is, the load distribution is based on the H. A. A. front spar design curve.

One of the essential differences between the second and third methods of determining the design tail loads is that the wing area employed in the former is that of the wing panels alone (241.2 square feet), while in the latter method the wing area includes that portion of the fuselage between the lower wings. In the third method the lower wing area is accordingly, 12.5 square feet more than is used in the second method.

Since the third method of computation covers a considerable range of diving speeds, it was thought
interesting to compare these theoretical loads with the 260 miles per hour dive listed as run No. 226 in Report No. 364. The small curve in the corner of Figure 13 of this report is merely an interpolation curve from which the tail load at 260 miles per hour is determined as 1,000 pounds. In Table VI, page 103 of Report No. 364, the load for half the tail is listed for this run as 458 pounds; making a total load of 916 pounds. A better comparison would be that of tail moment rather than tail load, since then the error of assumption of center of pressure position would be nullified. From Table VI of Report No. 364, it can also be seen that the down load of 458 pounds has a positive moment of 324 foot pounds about the elevator hinge.

Then if moments are taken about the center of gravity of the airplane, the center of pressure is either 324 or 0.71 feet beyond the hinge. The distance of the hinge from the center of gravity is 14.97 feet, so the total distance to the center of pressure is 15.68 feet.

Moment about center of gravity in flight = 2(458) (15.68) = 14,350 foot pounds.

Moment calculated by third method = 1,000 X 178 = 1,000 X 14.83 = 14,830 foot pounds.

Difference = 14830 - 14350 = 3.34 per cent.

Sometime after the above computations were made, an inspection of the plotted pressures over the horizontal tail surfaces, given on Figure 85, page 77, of Report No. 364, led the authors to believe that the center of pressure of the horizontal surfaces should be ahead of the hinge line rather than behind it as indicated in Table VI. Time did not permit an actual determination of the center of pressure position, but nevertheless for the purpose of comparison, it was assumed that the moment about the hinge, instead of being positive, should have been listed as -324 foot pounds.

Then the center of pressure would have been 14.97 - 0.71 or 14.26 feet from the center of gravity. Then:

Assumed moment about center of gravity in flight = -2(458) (14.26) = 13,070 foot pounds.

Difference = 14830 - 13070 = 3.5 per cent.

### Table 17.—Summary of quantity and type of data presented in Report No. 364

<table>
<thead>
<tr>
<th>Run No.</th>
<th>Type</th>
<th>Speed, miles per hour</th>
<th>Re­corded pressures</th>
<th>Plotted pressures</th>
<th>Time history</th>
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</tr>
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<td>6</td>
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</tr>
</tbody>
</table>

1 These runs are not of much use from a structural standpoint, but have been run to determine the effect of the slip stream on the distribution. These runs give fabric pressures over a few orifices.

Plotted pressures with an asterisk (*) indicates that center of pressure positions are also given on the diagram.

Note:—In addition to the data given above, center of pressure positions, and rib loads for runs Nos. 138, 195, and 196 were received through the courtesy of Mr.E. V. Rhode. These additional data were used in Table 13 of this report.