[redacted]

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WARTIME REPORT

ORIGINALLY ISSUED
February 1946 as
Confidential Bulletin L5H27

EFFECTS OF EXTERNAL FUEL TANKS AND BOMBS ON CRITICAL SPEEDS OF AIRCRAFT

By S. Katzoff and Robert S. Finn

Langley Memorial Aeronautical Laboratory
Langley Field, Va.

NACA

WASHINGTON

NACA CB No. L5H27

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

CONFIDENTIAL BULLETIN

EFFECTS OF EXTERNAL FUEL TANKS AND BOMBS ON CRITICAL SPEEDS OF AIRCRAFT

By S. Katzoff and Robert S. Finn

SUMMARY

The danger of shocks and separation in high-speed flight due to the presence of externally carried fuel tanks and bombs is pointed out, and a rough analysis of the phenomena involved is attempted. A discussion is given of the application and limitations of the superposition methods for calculating local velocities in the region between the wing and the external stores, and for predicting critical Mach numbers. Some suggestions for improved designs are also presented.

INTRODUCTION

The typical external fuel tank is a streamline body of revolution between one and two times as long as the wing chord and is attached under the fuselage or wing by a system of brackets with or without a fairing. The externally carried bomb, although smaller, may be equally heavy and require similar large brackets and fairings. Such installations, although hardly of highest aerodynamic efficiency, have heretofore served satisfactorily. Difficulties have arisen, however, with the recent development of tactics requiring fighters to engage in combat without dropping these external stores. In high-speed dives in particular, compression shocks and separation occur in the region between the wing and the tank or bomb, with corresponding reduction of airplane speed, large and erratic yawing moments, and buffeting so severe that damage to the airplane may result.

The purpose of the present report is to point out the existence of these difficulties to designers of such installations, to give methods of roughly predicting the speeds at which local shocks will occur, and to indicate what methods have been tried or might be tried to raise the critical speed. The standard methods are outlined for predicting local velocities on airfoils or on bodies of revolution in low-speed flight, and some discussion is included of the superposition of these velocities in the region between the wing and the external stores and of the prediction of critical Mach numbers. The methods outlined for predicting interference or for improving the design, however, are based mainly on very meager information, and it cannot be expected that efficient designs will be achieved without considerable experimental work.

LOCAL VELOCITIES ON AIRFOILS AND ON BODIES OF REVOLUTION AT LOW SPEED

The local velocity on the surface of any of the NACA airfoils can be readily determined, at least at low angles of attack, by means of the procedure given in reference 1, pages 23 to 31. The airfoil is considered to be formed from one of the basic symmetrical airfoils by curving its center line according to one of the standard mean lines, and the velocity distribution is then given as the sum of the following components:

(1) The velocity distribution on the symmetrical airfoil at 0° angle of attack. The chordwise distribution of the local velocity v relative to the free-stream velocity V is given in supplement I of reference 1 for the NACA thickness distributions.

(2) A velocity increment Δv corresponding to the lift distribution along the mean line at its design lift coefficient, that is, the lift coefficient for which the forward stagnation point is precisely at the leading edge. The chordwise distribution of $\Delta v/V$ is given in supplement II of reference 1 for the NACA mean lines. It is generally positive on the upper surface and negative on the lower surface.

(3) A lift correction $\Delta c_l\left(\frac{\Delta v_a}{V}\right)$ proportional to the difference between the actual section lift coefficient and the design section lift coefficient Δc_l. The chordwise distribution of $\frac{\Delta v_a}{V}$ is characteristic of the thickness distribution and is tabulated in supplement I of reference 1.

The net velocity increment on the airfoil is thus

$$\left(\frac{v}{V} - 1\right) + \frac{\Delta v}{V} + \Delta c_l \frac{\Delta v_a}{V}$$

Near zero lift, for symmetrical or only slightly cambered airfoils, the thickness term $\frac{v}{V} - 1$ is the most important of these increments. For an airfoil of given thickness ratio, such as might be required to streamline a given arrangement of brackets, the velocity increment $\frac{v}{V} - 1$ at the thickest part is about equal to the thickness ratio if the thickness distribution is approximately elliptical, as in the NACA 16-series or 65-series low-drag airfoils. For the NACA 00-series airfoils, however, of which the maximum thickness occurs farther forward, the increment is of the order of 1.5 times the thickness ratio and occurs 10 to 20 percent of the chord back from the nose.

The decrease in the velocity increment with distance from the airfoil surface may be estimated from figure 7 of reference 2 and from the velocity contours shown in reference 3. The decrease is relatively gradual, except near the nose of the airfoil, and is of little interest inasmuch as the critical region is normally at or very close to the wing surface itself at the juncture with the support.

For streamline bodies of revolution, there exists no systematization of forms or of methods for computing velocities, such as exists for airfoils. For any particular form the methods of references 4 or 5 can be used to compute velocity distributions, or the velocities can be estimated by comparison with velocities already derived for similar shapes. In figure 1 a number of shapes are shown together with their velocity distributions. For an ellipsoid of revolution having a thickness ratio of 0.2, the velocity increment at the center is about 0.06V; and the increment varies approximately as the 1.5 power of the thickness ratio. Moving the section of maximum thickness farther forward moves the position of peak velocity forward and also increases the value of the peak velocity with, however, a rapid reduction of velocity farther back. The velocity increment near such a body of revolution drops rapidly with increasing distance from the body; 1 diameter from the surface the velocity increment is about one-third of that at the surface of the body for a thickness ratio of about 0.2 (reference 2).

For a very blunt shape consisting of a cylinder surmounted with a hemisphere, very high velocity increments, of the order of 0.25V, exist at the base of the hemisphere; the effect is local, however, and at a point a short distance back and $\frac{1}{3}$ diameter from the surface (about where the support would be attached to the wing) the increment is only 0.03V to 0.04V.

SUPERPOSITION AND INTERFERENCE

In studies of the flow near arrangements of aerodynamic bodies, it has been found that the velocity increment at any point in the field can, to a first approximation, be taken as the sum of the velocity increments that the isolated bodies would contribute at the same point (references 2, 6, and 7). The adequacy of such a generalization, or, in fact, its interpretation, may be very uncertain in many cases, however; and the following remarks have been included to help broaden the concept and to aid in its application in cases of interest for the present problem:

(1) For a long tank below the wing (fig. 2(a)), the net velocity increment at point A - about where the support might be placed - is slightly greater than that given by the combined increments of the wing and the tank. A rough estimate of this interference, derived by disposing along the surfaces of each body sources and sinks of sufficient density to neutralize the normal components of the flow induced by the other body, indicated an increment at point A of about 0.03V for the arrangement shown and an increment of about 0.05V for an arrangement in which the distance between the wing surface and the tank is about one-half of that shown. A corresponding decrement in velocity occurs in the region under the leading edge, near point B.

(2) A shorter object, such as a bomb (fig. 2(b)), may be considered as reflected in the lower surface of the wing; hence the velocity increment at point A due to adding the bomb below the wing is about twice that which the isolated bomb would induce at the same point.

(3) The velocity increment in the juncture of a wing tip and a tank (fig. 2(c)) is considerably more than the sum of the velocity increments of the wing tip and of the tank, because the tank surface tends to act as a reflection plate and accordingly to make the wing contribution equal to that of a two-dimensional wing (twice that of the wing tip). For the case shown in this figure, the total velocity increment in the juncture is the sum of that due to the tank and about three-fourths of that corresponding to the two-dimensional wing. Where the wing passes through the body (fig. 2(d)), as for a midwing airplane, the effect is greater; however, the net velocity increment in the juncture generally remains slightly less than that given by combining the increments of the fuselage and of the two-dimensional wing.

(4) The velocity increment on the bottom of the fuselage for a typical midwing arrangement (fig. 2(d)) is the sum of the increment for the fuselage alone and about 0.3 to 0.4 of the increment that the two-dimensional wing would produce at the same point. For the case shown in this figure the fuselage increment at point A is 0.06V and the wing increment is about 0.03V (reference 8).

(5) When a small-chord airfoil is placed in the space between the wing and the tank, it behaves as a two-dimensional airfoil in the given field. A large faired strut of the type commonly used between the wing and the tank, however, will probably show only about three-fourths of the velocity increment (calculated relative to the local field) that would correspond to a two-dimensional flow.

Example.- Pressure distributions measured on models of four different arrangements of external tanks are reproduced from reference 9 in figure 3. As a rough check of the concepts just discussed, the following analysis has been made of the first case (fig. 3(a)) for comparison with the test results:

The critical region is expected to be near the nose of the fairing, about at point A. The wing has an NACA 2216 section and is operating at a lift coefficient of 0.1; in the region indicated, which is about 0.20 chord from the leading edge, the wing velocity increment should be 0.19V. The forward part of the tank seems almost exactly an ellipsoid of revolution of thickness ratio 0.3; from figure 1 the velocity increment on the tank surface closest to point A is found to be about 0.09V, but this increment will be reduced to about 0.06V at point A. The additional interference, estimated by comparison with values given in the preceding section, is about 0.04V, which makes a total increment, with the fairing absent, of $(0.19 + 0.06 + 0.04)V = 0.29V$. The fairing is 14 percent thick; its nose seems somewhat more blunt than that of an ellipse but considerably less blunt than that of the NACA 00-series airfoils; thus the two-dimensional velocity increment is estimated to be 0.16 times the free-stream velocity. If only three-fourths of this increment is considered effective, the net velocity at point A is estimated to be

$$1.29(1 + 0.75 \times 0.16)V = 1.45V$$

The corresponding low-speed pressure coefficient, defined as $\left(\dfrac{\text{Local velocity}}{\text{Free-stream velocity}}\right)^2$, is $1.45^2 = 2.1$. Figure 3(a) shows a pressure coefficient of 2.0 at point A. Ahead of this region the pressure coefficient drops rapidly along the juncture but rises sharply to about 2.2 for the median section. The check seems fairly satisfactory.

A procedure that has been suggested for estimating the velocity increment for such installations is to consider the fairing as a two-dimensional airfoil and add the increments. Thus the net velocity in this case would be calculated as

$$(1 + 0.19 + 0.06 + 0.16)V = 1.41V$$

which is nearly the same result.

ESTIMATION OF CRITICAL MACH NUMBER

For two-dimensional flow, the curve given in supplement IV of reference 1 may be used to predict, from the maximum low-speed pressure coefficient, the flight Mach number at which local sonic speed will occur (the critical Mach number). For a slender body of revolution, however, it appears that the critical Mach number is about 0.05 above that indicated by this curve, so that the curve is considered to underestimate slightly the critical Mach number when applied to a flow field, part of which is contributed by a tank or a fuselage. Figure 4, the curve of which is about 0.02 above that of reference 1, is, therefore, suggested as more applicable to the present problem. This increment may be slightly too large for a typical wing-tank arrangement and slightly too small for a typical fuselage-tank arrangement; however, further refinement of the analysis is considered unwarranted in view of the existing inaccuracy both in the methods of computing compressibility effects and in the methods of computing interference effects. In the preceding example, for which the low-speed pressure coefficient in the critical region was estimated as 2.1, figure 4 predicts a critical Mach number of 0.58.

In conservative designs, wherever possible, low-speed pressure coefficients are kept so low that the critical Mach number will not be reached in normal operation; actually, however, serious difficulties such as large increases in drag and violently separated flow frequently do not occur until the Mach number slightly exceeds this critical value.

SUGGESTIONS FOR HIGH-SPEED DESIGNS

The use of a tank in the form of a blister or low nacelle, by eliminating the exposed strut fairing, will greatly increase the critical Mach number. A tank having more nearly universal application might be made with a somewhat recessed region on the upper surface, near the front part of the strut fairing, to help reduce the velocities in this region. Similarly reshaping a bomb is, of course, out of the question; nevertheless, consideration might well be given to the use of a blister fairing between the upper part of the bomb and the wing surface. Reducing the thickness of the support and providing its maximum thickness far back along the chord is an obvious improvement, subject, however, to considerations of structural strength and rigidity. A staggered