THE AREA RULE
FIRST SPECIFIC LECTURE
INTRODUCTION

This series of lectures will be concerned with the physical basis, the development, the application, and limitations of the area rule. In a broad sense, the area rule is a means for relating the broad shock waves produced by configuration at transonic and supersonic speeds and the resulting wave drag to the cross-sectional area of the airplane. In the original conception, the area rule was relatively simple. However, throughout the past several years, a number of extensions of the rule have been proposed which while greatly enhancing the effectiveness and reducing wave drag have at the same time made the application of the concept considerably more complex. The area rule has been utilized to significantly reduce the drag of a number of American airplanes and has been extremely helpful in predicting the wave drag of various airplane configurations. Because of its basic limitations, it is probable that the area rule cannot be used to improve and predict the drag characteristics of high performance airplanes and missiles flying at Mach numbers greater than roughly a value of 2.0. Transport, logistic, and ground-support type airplanes will probably continue to be designed to fly
the area rule will probably be utilized to significantly reduce and improve the drag characteristics of such types of airplane configurations.

The first of the two lectures will consider the transonic and the supersonic area rules as interpreted from experimental measurements and theoretical calculations. In the second lecture, some of the considerations involved in applying the area rule in the design of airplane fuselages for reduced drag at transonic and supersonic speeds and recent applications of the area rule in the design of configurations with delayed drag rise at high subsonic speeds will be discussed.

Note: The remainder of the first general lecture is taken from the first of the more detailed lectures, the parts to be taken are bracketed in red.
SECOND GENERAL LECTURE

Note: The second lecture is taken from the fourth and fifth of the detailed lectures. The parts to be taken are bracketed in red.
plane. By assigning different values to \( X \) while keeping \( \theta \) constant, we obtain a series of parallel planes at the same angle \( \theta \) around the x axis. By assigning different values to \( \theta \) while keeping \( X \) a constant, we obtain a set of planes enveloping that Mach cone whose apex lies at the point \( X = x \).

Selecting a value of \( \theta \), we cut through the wing-body system with a series of planes corresponding to different values of \( X \). The total intercepted area in each plane is then equated to the area intercepted by this plane passing through the equivalent body of revolution. If we denote the area intercepted obliquely by \( s(X, \theta) \), then the area \( S(X, \theta) \) is defined by

\[
S = s \sin \gamma
\]

where \( \gamma \) is the Mach angle (i.e., \( \sin \gamma = 1/M \)). Thus, \( S \) is the area intercepted by normal planes passing through the equivalent body of revolution on the assumption that this body is slender. Again, we write

\[
S'(X, \theta) = \frac{S(X, \theta)}{X} = A \sin \gamma
\]

with

\[
\cos \gamma = \frac{X}{X_0}
\]

Here, however, both the length \( 2X_0 \) and the shape of the equivalent body vary with the angle \( \theta \).
those for below this plane. The wing areas above or below the chord plane are considered with the corresponding fuselage areas. The favorable effects on drag that may be obtained through the use of fuselage contours designed on the basis of such divided areas for a cambered wing are illustrated in figure 13. The cambered 45° sweptback wing of reference 5 was tested in combination with two contoured bodies. The wing has an aspect ratio of 1.0, a taper ratio of 0.15, an NACA 64A206, a = 0 section at the root, and an NACA 64A203, a = 0.8 (modified) section from the 50-percent semispan to the tip. The wing is placed symmetrically on the bodies. The total cross-sectional areas for the two bodies were essentially the same. One body was shaped symmetrically to obtain favorable total area distributions by using complete wing cross-sectional areas; the other body was shaped asymmetrically to obtain favorable area developments above and below the wing plane by utilizing divided wing cross-sectional areas. Since the area for the cambered wing above the chord plane is greater than that below, the indentation of the fuselage above the wing is deeper than that below. The design Mach number was 1.4. The results for a Mach number M of 1.43 presented in figure 14 indicate that the asymmetrical indentation
of the area rule would probably be realized by utilizing equally weighted developments obtained with cuts for values of $\theta$ of 15°, 45°, and 75°. Such cuts define the approximate mean developments for 30° segments of a quadrant. However, sufficiently accurate approximations of the results obtained using such cuts are arrived at by utilizing the cuts for $\theta = 0°, 45°$, and 90°. These latter cuts are usually
The mechanism by which indenting or contouring the body reduces the drag can be somewhat better understood if the local flow field in the vicinity of the wing and body is examined. Figure 5 gives a little insight into the mechanism by which body indentation manages to reduce the drag. Plotted in this figure are contours obtained from pressure-distribution measurements made in the Langley 8-foot transonic tunnel of the increment in local pressure coefficient due to indenting the fuselage. The incremental pressure coefficients shown were obtained by subtracting the pressure coefficients of the wing in the presence of the basic body from those of the wing in the presence of a body indented for a design Mach number of 1.2. The data are for an angle of attack of 0° and a Mach number of 1.13. The wing geometry is described in reference 4.

It can be seen that indenting the fuselage caused a reduction in pressure over the forward half of the wing and an increase in pressure over the rearward half of the wing which obviously reduces the pressure drag of the wing. The reduction in pressure forward is the result of expansion waves from the beginning of the indentation where the fuselage narrows down; the increase in pressure at the rear is due to compression waves from the portion
of the indentation where the diameter increases. The expansion field dissipated rapidly as it travels across the span. The compression field, however, maintains its strength for a considerable distance across the span. Although the wing was cambered, the pressure contours over the lower surface of the wing were similar to those over the upper surface. It appears that the pressure fields cross the wing at approximately the Mach angle of the free-stream flow. Also, since the zero-pressure line and the line of maximum thickness of the wing should coincide for minimum drag, the design Mach number should influence the position of the maximum thickness line.

Presented in figure 6 is the spanwise distribution of pressure drag of the wing-body combination just discussed at an angle of attack of 0° and a Mach number of 1.13. It can be seen that reductions in drag are realized up to approximately 60 percent of the semispan of the wing with smaller reductions extending out to the wing tips. The reduction in wing pressure drag due to body indentation (represented by the region between the two curves) is approximately 80 percent of the total reduction obtained for the wing-body configuration. The remaining 20 percent is due to the favorable effect of the distribution of pressures over the indented fuselage.
Fuselage shaping may also be used to reduce the drag of canopies.

Figure 9 shows the results of such a symmetrical body modification on the pressure drag of canopies having flat and vee windshields. The symmetrical indentations used were designed to cancel the exposed canopy cross-sectional areas normal to the body axis. The indentations reduced the fuselage volume by approximately 3 percent.

The normal area indentation produced substantial reductions in the total pressure drag of both the flat and vee windshields (fig. 9) at transonic and supersonic speeds. The test results for the flat windshield are compared with the theoretical pressure drags for both the indented and original configurations in this figure. The theory indicates a large reduction in pressure drag due to indentation and shows that the effectiveness of the transonic indentation diminishes with increasing Mach number. The actual reduction in drag is slightly less than one-half of that predicted; nevertheless, the actual reduction is an appreciable part of the canopy drag.

These tests and others show that $M = 1.0$ indentations may be expected to give from 25 percent to 50 percent reduction in canopy drag at low
supersonic speeds. Greater reductions may be possible from supersonic indentations or unsymmetrical indentations.

The results just described are applicable, more or less, to airplanes having a smooth total normal area distribution for the body, wings, and other components. For a more practical case, where the airplane area diagram has a bump due to the wing, the optimum canopy size and location may depend, to a large extent, on designing the canopy to make the total normal area distribution smooth, as is shown in figure 10. The configuration is a fighter airplane, with a canopy modification that was recently tested in the Langley 8-foot transonic tunnel. The original model had a small canopy and a poor area distribution in the region of the wing and small canopy. The canopy volume was almost doubled and its fineness ratio was increased to make the total airplane area distribution smooth. As a result, the total drag coefficient (based on wing plan-form area) was reduced about 4 percent and the pressure drag by approximately 7 percent at \( M = 1.13 \). The reductions at transonic speeds were less, with no reduction being noted below a Mach number of 0.9.
sweep of 45°. The solid lines indicate the body contour obtained through an axisymmetrical application of the transonic-area-rule principle. The dashed lines indicate the wing-body juncture of the second configuration, which was made to conform to the calculated streamline shape. This streamline shape was calculated with the use of experimental two-dimensional velocity distributions. These data were measured at a Mach number corresponding to the velocity component normal to the swept-wing leading edge. The body cross section was then adjusted at the top and bottom so that the longitudinal area development was identical with that of the axisymmetric area-rule configuration. The fairing behind the wing trailing edge was arbitrary.

Tests of the two configurations were made in the Langley transonic blowdown tunnel at a Reynolds number of about $2.5 \times 10^6$ based on the wing mean aerodynamic chord and at angles of attack up to about 10°. Transition was fixed to minimize change in viscous effects.

Some of the results of the investigation are presented in figure 9. Plotted are drag coefficients based on total wing area as a function of Mach number for two lift coefficients, 0 and 0.4. The solid lines refer