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RESEARCH MEMORANDUM

INVESTIGATION OF FLOW CONDITIONS AND THE NATURE OF THE WALL-CONSTRICTION EFFECT NEAR AND AT CHOKING BY MEANS OF THE HYDRAULIC ANALOGY

By

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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INVESTIGATION OF FLOW CONDITIONS AND THE NATURE OF THE WALL-CONSTRICTION EFFECT NEAR AND AT CHOKING BY MEANS OF THE HYDRAULIC ANALOGY

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SUMMARY

The closed wind-tunnel phenomenon of choking and the wall-constriction effects in the subsonic Mach number range where supersonic Mach numbers appear was investigated by means of the hydraulic analogy. For this investigation, Mach number fields were obtained about several symmetrical airfoils at zero lift in a water channel. In the course of the analysis, consideration was given to several factors affecting the applicability of the results to wind-tunnel operation. With the approach of choking, the flow was found to approach the one-dimensional form. Boundary-layer thinning in the region of the model appreciably increased the choking Mach numbers, but critical speed, position of maximum thickness, and ratio of maximum thickness to chord had little effect. The constriction effect of the walls was of the same nature in the flows investigated as in completely subsonic flow. Approximate correction for the constriction effect appeared possible with Mach numbers up to the first attainment of choking. Possibilities of correction were discussed.

INTRODUCTION

Although a number of investigations have shown that serious wall-constriction effects occur when models are tested in closed wind tunnels at Mach numbers less than, but near, unity (references 1 and 2), additional information is needed concerning the mechanism of this phenomenon, the progression of its severity with increasing Mach number, and the extent to which simple corrections can be applied. Because the theory available is not valid for stream Mach numbers greater than the critical (that is, greater than the stream Mach number for which a speed equal to the speed of sound is first attained in the field (reference 3)) and experimental measurements of the flow fields in a wind tunnel are tedious, the influence of the walls on the flow about a model as demonstrated by the flow fields has not previously been extensively studied and analyzed. The purpose of this investigation is to study several compressible-flow fields observed by means of the hydraulic analogy in order to obtain qualitative information concerning choking and the nature of the wall interference in the Mach number range near choking, that is, near the maximum stream Mach number attainable with a model in place.

The hydraulic analogy was used because of its convenience and economy and because field surveys can be made without interference by the testing equipment. Since its basic theory is valid (reference 4) and previous experimental work with the analogy (reference 5) has shown effects of compressibility on a circular cylinder comparable with those obtained in two-dimensional wind tunnels, it was thought that the qualitative nature of wind-tunnel-wall interference for stream Mach numbers below and near unity could be reliably studied. This Mach number range will be referred to as the transonic range.

The desired information was obtained by surveying the flow fields about models having various shapes which would show the effects of chord length, critical speed, and position of maximum thickness on the development of the tunnel-choking condition. A more detailed investigation of wall-interference effects was made on one model in two channels of different width.

In several respects, such as a very low Reynolds number, a very thick boundary layer, an effective value of 2.0 for the ratio of specific heats, and the effects of vertical accelerations (see reference 5), the test conditions in the water channel depart from those usually encountered in wind-tunnel testing. Because of these effects, special care must therefore be exercised in the interpretation of the results.

SYMBOLS

h	depth of water
h_l	depth of water at any point in the field; that is, local depth
h_o	depth of water at zero velocity; that is, total depth
h_s	depth of water at great distance ahead of the airfoil; that is, stream depth
h_{cr}	depth of water at position of Mach number unity
c	chord of airfoil
g	acceleration of gravity
M	Mach number $\left(\sqrt{\frac{2(h_o - h)}{h}}\right)$

M_l	local Mach number
M_s	stream Mach number
M_{cr}	critical Mach number
M_{ch}	choking stream Mach number
R	Reynolds number of model, based on chord length
V_l	velocity at any point in the field
V_s	indicated stream velocity
Q	mass rate of flow in channel, cubic inches per second
w	width of channel
x	coordinate along tunnel or channel axis
y	coordinate perpendicular to tunnel or channel axis
δ_a^* or δ_b^*	wall boundary-layer-displacement thicknesses in two-dimensional wind tunnel
δ_h^*	floor boundary-layer-displacement thickness
δ_w^*	side-wall boundary-layer-displacement thickness
μ	viscosity of water
ρ	density
m	strength of source used to represent the wake
γ	ratio of specific heats
S	stream-tube area
ΔS^*	change in stream-tube area due to changes in boundary-layer thickness
a	breadth of wind tunnel
b	height of wind tunnel

APPARATUS AND METHODS

The tests were conducted in the 20-inch-wide vertical-return water channel of the Langley 8-foot high-speed tunnel. (See reference 5.) A sketch of the channel is shown in figure 1. The motor-driven propeller forces the water into a divergent section, then through an 80-mesh antiturbulent screen into a large quieting section. The water is then partially accelerated by flowing through a section having converging walls and a horizontal floor. It is accelerated to test-section velocity by flowing over an elliptical floor section just ahead of the test section (fig. 2).

The test-section floor (fig. 2), which is a modification to the original channel, was used to compensate for gradients caused by the thickening of the floor boundary layer, and had the additional effect of thinning the boundary layer in the test section and of reducing the secondary flow in the corners. The new floor was raised 1 inch from the original floor. It was curved to compensate for the boundary layer so that all gradients were removed for all stream Mach numbers up to 1.0.

The tunnel speed was automatically controlled to within 0.1 of 1 percent. The airfoil surface pressures were measured with a slant-tube manometer which could be read to 1.5 percent of free-stream dynamic pressure at the choking condition. The field water depths were measured with a platinum wire probe mounted on a two-way cross-rail survey system which permitted placing the probe over any point in the field. The probe was connected to a dial gage such that direct readings of the water depth were obtained. A sensitive electronic relay indicated the exact instant of contact between the probe and water surface. The accuracy of the field measurements was about 0.1 of a percent.

The models, which were tested at zero lift, were 24-inch-chord NACA 0012 and NACA 16-012 airfoils, a 2.88-inch-diameter circular cylinder, a 2.88-inch-wide flat plate, a 24-inch-chord, 2.88-inch-wide airfoil having parallel sides and circular ends (fig. 3), and a 5-inch-chord, 10-percent-thick, biconvex, circular-arc airfoil. The NACA 0012 and NACA 16-012 airfoils were tested in the reversed position, that is, with the flow from tail to nose, as well as in the conventional position. A special channel 7.5 inches wide for use with the circular-arc airfoil was constructed in the center of the regular channel by placing two strips of sheet aluminum 7.5 inches apart in the channel. These strips extended from the leading edge of the elliptical section of the floor through the test section.

The floor boundary layer and wake surveys were made with a total-head tube of 0.061-inch outside diameter placed in the stream. It was mounted on the survey probe and hence could be placed at any point in the stream. Another probe was mounted directly over the tube for reading the local pressure at that point.

The methods of measurement and conversion of measured to desired quantities were essentially the same as those used in reference 5.

The field surveys were taken in the following manner: The airfoil model was set on the channel center line at zero angle of attack. The static water level was adjusted to 1.000-inch depth in the test section.

The propeller-drive motor was started and its speed set for the desired stream Mach number. Water-depth readings were taken over one-half the field at a sufficient number of points to determine the depth distribution about the airfoil. The total depth was measured at a point over the settling chamber. The test condition was held constant by checking the total depth and the depth at the center of the test-section entrance and then making any minor adjustments necessary to keep these values within 0.1 of a percent of their original values. The usual adjustments were changing the motor speed and addition of water, the first to correct for the collection of dirt on the antiturbulent screen, the second to correct for evaporation and leakage.

The stream Mach number was determined by averaging the local Mach numbers across the upstream end of the test section. For large-chord airfoils, the region immediately ahead of the model was neglected in the averaging process. From an analysis of the measurements obtained, it is believed that the Mach numbers obtained are within 1 percent of the correct indicated stream Mach numbers.

PRESENTATION OF RESULTS

Choking tests.- Although the measurement of the stream Mach number is within 1 percent, the stream Mach numbers of the choking fields which are to be presented show several apparent inconsistencies. These inconsistencies may be due to variations in floor boundary-layer conditions or in the separation phenomena about the bodies tested.

Choking.- Mach number fields for several model chord-to-channel width ratios are presented in figure 4 at choked Mach numbers obtained at maximum power input (later called full choking) to show the approach to one-dimensional flow at the choking condition. The Mach number fields about the 2.88-inch-thick models in the 20-inch channel are presented in figures 5 to 11. The flow conditions vary from subcritical to choking and show the progressive development of compressibility effects on large models. Since the surveys were taken at zero angle of attack, the fields are symmetrical about the center line, and only one-half the survey in each plot is therefore necessary. The field lines represent constant Mach numbers. The solid lines are Mach numbers less than 1.0, the dotted lines are Mach numbers of 1.0 and greater. In these figures, after the choking Mach number has once been reached, that is, after the $M = 1.0$ line has reached the channel wall, the succeeding field configurations are obtained by progressively increasing the drive motor speed so that the total head is increased and the depth downstream of the airfoil decreased.

Wall-interference tests.- The results of the wall-interference tests are presented in figures 12 to 15. The figures show field surveys and surface Mach number distributions about the 5-inch-chord, 10-percent-thick, biconvex, circular-arc airfoil at zero lift in a 7.5-inch-wide channel and in a 20-inch-wide channel. The same airfoil was used in two channels of different width rather than different size airfoils in the same channel so that Reynolds number and the effect of size on vertical acceleration would be about the same leaving only wall interference to cause changes in the flow patterns. The surface Mach numbers were determined by taking the pressures from orifices located in the walls of the model. Since it was necessary to compare fields at definite stream Mach numbers, the data were cross plotted and faired so that the field surveys and surface distributions could be plotted for any desired stream Mach number.

The field surveys presented in figure 13 show the effects of different width channels on the flow field about a model by comparing the fields in the 7.5-inch and the 20-inch channels at the same stream Mach number. The field surveys presented in figure 14 show the results of making a simple correction for wall effects by comparing fields which have the same maximum local Mach number on the model. The surface Mach number distributions which correspond to the field surveys of figures 13 and 14 are plotted in figure 15. The compared surveys (figs. 13 and 14) are plotted in the reflected position; half of each field is shown. The upper half of the figure shows the airfoil in the 7.5-inch channel; the lower half shows the airfoil in the 20-inch channel with only 4 inches of the field showing.

Theoretical Mach number fields.- Several theoretical Mach number fields of a circular-arc airfoil in an infinite field (labeled free field) and in a 7.5-inch channel were computed for comparison with the experimental fields of figures 13 and 14. These theoretical fields are presented in figures 16 and 17. A free or infinitely extended theoretical field is substituted for the experimental field in the 20-inch channel. The theoretical fields are presented with and without the wake in order to show the effects of wake distortion.

The free potential-flow field was calculated for a stream Mach number of 0.750 by use of the small perturbation compressibility theory in the manner employed in reference 6 in conjunction with the theory for the incompressible potential flow over a circular mound as given in reference 7. (See figs. 16(a) and 16(c).) The effects of solid blockage were computed by applying this same method of taking account of the effect of compressibility to the incompressible interference field obtained by the method of reference 8. The methods of reference 3 were used in computing the effects of the wake except that a source, with strength linearly distributed along the chord between the center and trailing edge, was used in order to minimize the effects of local induced velocities which would result from the concentrated source of reference 3. The total source strength m was made equal to the wake volume displacement found from a survey behind the airfoil with a total-head tube. The gradients due to wake blockage were computed by first assuming that the reflected source images would act as concentrated sources. The effects of the images were computed by considering all the source images as incompressible and calculating their total velocity increments by the method given in reference 8. The velocity increments due to the first two sets of incompressible images were subtracted from the total velocity increments, thus giving the increments due to the remaining outer images. These were then corrected for compressibility by multiplication with the factor $\left(1 - M_s{}^2\right)^{-\frac{3}{2}}$ and added to the compressible increments due to the first two source images computed directly from formula (A-11) of reference 3.

The interference velocity increment at the airfoil due to the wall-constriction effect on the wake was calculated from the formula

$$\frac{\Delta V}{V_s} = \frac{m}{2V_s w} \frac{1}{1 - M_s{}^2}$$

(See reference 3.) The fields were then combined and the final velocity ratios converted into Mach number by the formula

$$\frac{M_l}{M_s} = \frac{V_l/V_s}{\sqrt{1 - \frac{\gamma - 1}{2} M_s{}^2 \left[\left(\frac{V_l}{V_s}\right)^2 - 1\right]}}$$

with $\gamma = 2$. The resultant field with solid blockage but without the wake is presented for a Mach number of 0.750 in figures 16(b) and 16(d), and is compared with the free field at the indicated stream Mach number (fig. 16(a)) and at the effective stream Mach number (fig. 16(c)). In figure 17, the effect of the wake has been included. The two channel fields are compared with the free fields at M_s equal to 0.750 and with the free fields at stream Mach numbers corresponding to equal maximum Mach numbers on the airfoil. The theoretical results for the compressible flow in the channel are to be considered an extrapolation as well as an approximation, because the process by which they were obtained is no longer strictly valid when Mach numbers greater than unity occur in the field, but past experience indicates that the error should not be great so long as Mach number 1.0 is only slightly exceeded.

Several comparable incompressible fields were also computed using directly the formula from reference 7 and the wall corrections from reference 8. These are presented in figures 18 and 19, and are used in conjunction with figures 16 and 17 to study the effect of compressibility on the wall interference.

Boundary-layer surveys.- Surveys of the floor boundary layer in the fields of the 24-inch-chord NACA 16-012 airfoil and of the 2.88-inch-wide flat plate normal to the flow were obtained by use of a total-head probe in much the same manner usually employed for boundary-layer surveys in air. With regard to the floor boundary layer, the flow may, of course, be considered incompressible. The displacement thicknesses referred to the upstream end of the test section are presented in figure 20. The negative values are explained by the fact that the test-section floor drops 0.06 inch in 24 inches (fig. 2). These boundary-layer surveys correspond to full choking conditions in the channel with the channel drive motor operated at the highest speed; the comparable Mach number fields are shown in figures 6(j) and 11(i), respectively. The boundary-layer-displacement thicknesses of the empty channel for several choking conditions are presented in figure 21. These curves show the effects of increasing the power beyond that required barely to choke the channel.

APPLICABILITY OF THE RESULTS TO FLOW IN AIR

Several factors affecting the operation of the water channel and the application of results obtained in the hydraulic analogy to air are discussed in reference 5. One of these factors is the ratio of specific heats γ which in the analogy has a value of 2.0 as compared with 1.4 for air. This divergence of values is shown in reference 5 to have small effect in most cases of subsonic flow. If supersonic regions exist, however, the effect may no longer be negligible and may in fact be considerable.

With supercritical flows the hydraulic analogy also suffers from the fact that the energy dissipated in the shock waves (hydraulic jumps in the analogy) does not re-enter into the determination of the flow quantities as is the case for a gas.

The effects of the vertical accelerations (see reference 5) were minimized in the present tests by using relatively shallow water (1 in.), by employing large models in the case of the choking tests, and, for the wall-interference tests, by using a thin, sharp-nose model. Furthermore, any effect of size on vertical accelerations was largely eliminated from the comparisons involved in the wall-interference studies by using the same model for all wall-interference tests.

Another factor affecting the applicability of test results is the Reynolds number R. If the Reynolds number is assumed to be the same as if the test airfoils were submerged in an infinite water flow with the velocity of the flow in the water channel,

$$R = \rho\frac{Vc}{\mu}$$

where ρ is the density of the water and μ is its viscosity. Under the conditions of operation of the water channel with a 5-inch-chord model and a speed near Mach number 1.0 of about 18 inches per second, this Reynolds number is very small, about 80,000. This value corresponds approximately to that which would be obtained in air near Mach number 1.0 on a model of 0.1-inch chord. On the same scale the wind tunnel would be 0.4 inch high and approximately 0.04 inch wide. Although the low Reynolds number is a serious limitation, the influence of compressibility is believed to be even more important than that of viscosity in the transonic Mach number range. This fact is shown in figure 22 where, once the critical Mach number has been exceeded, the separation point is seen to move in the same manner in the wind-tunnel tests as in the water channel, though because of the much larger wall interference in the water channel and possibly for other reasons, including the differences in Reynolds number and values of ratio of specific heats, the separation points are not the same at any given indicated stream Mach number. The essential characteristic of both the air and water flows for this comparison at low angles of attack is that the large wake which occurs at Mach numbers slightly beyond critical is reduced as the shock moves toward the trailing edge with increasing Mach number. This behavior has been visually observed in both the air and water flows. The two flows, though characterized by vastly different Reynolds numbers, are therefore in a general manner comparable in the region near and at choking, particularly as regards the phenomena of choking and other wall-constriction effects here investigated.

With low fluid velocities, relatively thick boundary layers may be expected on the channel walls and floor. Experimental measurements have shown that the boundary-layer-displacement thickness is approximately 0.05 inch. This thickness is relatively insignificant on the side walls as it is a small portion of the channel width; however, it becomes extremely important on the floor as it is such an appreciable portion of the water depth that it will affect the mass flow, and hence the local Mach number and other quantities. The question therefore arises as to what effect this boundary layer has on the determination of the flow quantities in the water channel and on the analogy between the water and gas flows.

Inasmuch as the velocity outside the boundary layer is a function only of the difference between total head and the height of the free surface, it is correctly determined without consideration of the floor boundary layer. The equation used in calculating the wave velocity, which is also necessary in obtaining the Mach number, assumes constant flow conditions between the water surface and the floor. Since the flow velocity in the floor boundary layer is different from that at the surface, it will affect the wave velocity and hence will require that the depth used to calculate the wave velocity be corrected because of the boundary layer. Experiments were made to determine the corrections needed. Although inconclusive, the tests suggested that the depth used to calculate the wave velocity should be equal to the water depth minus the displacement thickness. Since no appreciable axial velocity gradients existed in the channel, it was assumed that the displacement-thickness surface would be at the same height as the floor at the test-section entrance; hence, the floor at the entrance to the test section was used as the reference for the depths measured in the channel.

The effects on velocity of changes in boundary-layer thickness will now be shown to be the same in the water flow as in the corresponding gas flow. Consider a gas flowing in a rectangular wind tunnel of width a and height b. Let the corresponding boundary-layer-displacement thicknesses be $\delta_a{*}$ and $\delta_b{*}$ and let these values receive the increments $d\delta_a{*}$ and $d\delta_b{*}$, respectively.

The equation of continuity for one-dimensional flow is

$$Q = \rho V(b - 2\delta_b{*})(a - 2\delta_a{*})$$

which, when differentiated, becomes

$$0 = \frac{d\rho}{\rho} + \frac{dV}{V} - \frac{2d\delta_b{*}}{b - 2\delta_b{*}} - \frac{2d\delta_a{*}}{a - 2\delta_a{*}} \tag{1}$$

From a consideration of compressible-flow relations, it may be shown that

$$\frac{d\rho}{\rho} = -M^2\frac{dV}{V}$$

which, when substituted in equation (1), yields

$$(1 - M^2)\frac{dV}{V} = 2\left[\frac{d(\delta_b{*})}{b - 2\delta_b{*}} + \frac{d(\delta_a{*})}{a - 2\delta_a{*}}\right] \tag{2}$$

Equation (2) represents the relation between changes in boundary-layer thickness and changes in wind-tunnel velocity.

A similar equation may be developed for the water channel. Assume a channel of width w with water depth h. Let the boundary-layer-displacement thicknesses be $\delta_w{*}$ and $\delta_h{*}$, respectively.

The equation of continuity for one-dimensional water-channel flow is

$$Q = \rho V(w - 2\delta_w{*})(h - \delta_h{*})$$

which, when differentiated, becomes (ρ is constant)

$$\frac{dV}{V} - \frac{2d\delta_w^*}{w - 2\delta_w^*} + \frac{dh}{h - \delta_h^*} - \frac{d\delta_h^*}{h - \delta_h^*} = 0 \tag{3}$$

From reference 5

$$V^2 = 2g(h_o - h)$$

which, upon differentiation, becomes

$$V\,dV = -g\,dh \tag{4}$$

Divide both sides by $g(h - \delta_h^*)$ or

$$\frac{V^2\,dV}{g(h - \delta_h^*)V} = -\frac{dh}{h - \delta_h^*} \tag{5}$$

In references 4 and 5, it is shown that

$$M^2 = \frac{V^2}{gh}$$

However, the experimental evidence in this report shows that h should be corrected for the floor boundary layer δ_h^* or

$$M^2 = \frac{V^2}{g(h - \delta_h^*)} \tag{6}$$

Substituting equation (6) in equation (5) gives

$$M^2\,\frac{dV}{V} = -\frac{dh}{h - \delta_h^*} \tag{7}$$

If equation (7) is substituted in equation (3), the result is

$$\frac{dV}{V}\left(1 - M^2\right) = \frac{2d\delta_w^*}{w - 2\delta_w^*} + \frac{d\delta_h^*}{h - \delta_h^*} \tag{8}$$

If the substitutions

$$w = kb$$

$$h = \frac{ka}{2}$$

$$\delta_h^* = k\delta_a^*$$

$$\delta_w^* = k\delta_b^*$$

are made in equation (8), it can be shown to be identical to equation (2). Any change in boundary-layer-displacement thickness therefore affects the flow in the water channel in the same way as the same change affects the flow in a wind tunnel with height equal to the width of the channel and with breadth equal to twice the water depth.

The effects of value of γ, vertical acceleration, differences in model Reynolds number and boundary-layer changes show that direct results obtained from the water channel have only qualitative application to air flows. Any conclusions drawn from water-channel results must be carefully considered in the light of the various factors affecting the flow; quantitative interpretations are possible only insofar as the differences in flow conditions can be taken into account.

ANALYSIS AND DISCUSSION

Subsonic Choking Phenomena

According to one-dimensional theory, the maximum rate of mass flow with constant total head through a stream tube of given cross section occurs when the velocity is equal to the speed of sound; and the maximum mass-flow rate in a wind tunnel, for which the flow must be essentially one dimensional, is therefore determined by the condition that sonic velocity exists at the throat or minimum cross section of the tunnel. When a model is placed in the test section, which usually corresponds to the throat, the minimum cross section through which the flow must pass and, therefore, also the maximum possible mass-flow rate and the corresponding indicated stream Mach number are reduced. With this maximum rate of mass flow, the tunnel is said to be "choked" and the corresponding indicated stream Mach number is called the "choking Mach number."

If one-dimensional theory be employed to estimate choking Mach numbers, the estimated values will, in most cases, be found to agree reasonably well with the experimental values (references 1, 2, and 9);

however, because the flow is not strictly one dimensional, the choking Mach number would, if other factors were negligible, be less than the value predicted from one-dimensional theory. This fact can be understood from a consideration of the normal components of the velocities at points along a line drawn from the thickest portion of the model perpendicular to the wall. With two-dimensional flow, the velocities at most points along this line must be different from sonic velocity, and the mass flow must be less than if the flow were one dimensional and the velocities at the line therefore sonic. A considerable departure from one-dimensional flow is required, however, to cause an appreciable change in choking Mach number, because near a Mach number of 1.0 the rate of mass flow is relatively insensitive to small changes in Mach number, as may easily be seen from simple one-dimensional compressible-flow theory.

The one-dimensional nature of the experimental choking fields is shown in figures 5 to 11. As the stream Mach number is increased and approaches the choking value, the flow actually does approach the one-dimensional form, at least in the region between the thickest portion of the airfoil and the wall. Lines of constant Mach number, which at lower Mach numbers curve and return to the model, tend at higher Mach numbers to run straight from the thickest part of the model to the wall.

The objection might here be made that the NACA 0012 and NACA 16-012 airfoils tested were so long relative to the channel width that even incompressible flow would be approximately one dimensional. The phenomena that occur as choking is approached would, however, not have been essentially different if smaller airfoils had been employed. A preliminary Mach number field survey about a 12-inch-chord NACA 0012 airfoil in a 24-inch channel with the power considerably increased over that necessary for choking showed exactly the same characteristics (see fig. 4) except that the two-dimensional characteristics in the region near the airfoil were somewhat more pronounced. The same tendency of the flow to approach the one-dimensional form as choking is approached is seen also in the case of the 5-inch-chord, biconvex, circular-arc airfoil in the 7.5-inch channel, figure 4.

The effect of the change from one-dimensional to two-dimensional character of the flow in reducing the choking Mach number is seen from a comparison of the choking Mach numbers of the circular cylinder (fig. 9) and of the parallel-side airfoil (fig. 10) with those of the airfoils with the same thickness (figs. 5 to 8). The choking Mach number is reduced from 0.64 or 0.65 for the airfoils to 0.61 or 0.62 for the circular cylinder and parallel-side airfoil with circular ends. A considerable part of this distortion of the one-dimensional-type choking field is believed to be due to the thinning of the floor boundary layer in the region near the wall. The influence of the boundary-layer thinning is relatively much greater near the wall than near the model, where the model itself largely controls the flow. In the subsonic region, the thinning of the boundary layer tends to decrease the Mach number. At the wall, therefore, the lines of constant Mach number, including the Mach number unity line, are slanted and shifted downstream. In a wind tunnel with a relatively large total rate of mass flow in comparison with that in the boundary layer the flow would be even more nearly one dimensional than appears from the hydraulic analogy. The flat plate (fig. 11) shows an even greater reduction in choking Mach number than was found for the circular cylinder, but in this case the obtained choking Mach numbers are not consistent, an effect that is perhaps due to the influence of the wake.

Except for this small indication in the case of the flat plate, no evidence of wake choking (see reference 3) was found in any of the tests. In all other cases, choking was determined by the thickness of the airfoil and occurred between the airfoil and the wall. Even in the case of the flat plate, the line of Mach number unity approached very near to the edge of the plate. The occurrence of wake choking in any practical airfoil tests appears very unlikely, though the wake may contribute to choking due to solid constriction downstream from the airfoil.

Another factor that might be expected to exert an important influence on choking is the critical Mach number. A comparison of the Mach numbers at first attainment of choking and critical Mach numbers for the 2.88-inch-thick airfoils and circular cylinder listed in the following table shows no such effect.

	M_{ch}	M_{cr}
NACA 0012 airfoil	0.646	0.637
NACA 0012 airfoil, reversed	0.631	0.623
NACA 16-012 airfoil	0.642	0.600
NACA 16-012 airfoil, reversed	0.637	0.630
Parallel-side airfoil with circular ends	0.608	0.525
Circular cylinder	0.618	0.520

The critical Mach numbers here tabulated are the values obtained from measurements in the water channel and, because of the low Reynolds number and of the blockage effects of the walls, need not agree with those obtained in wind tunnels. The choking Mach numbers for the NACA 0012 and NACA 16-012 airfoils are about the same, and although those for the parallel-side airfoil and for the circular cylinder are somewhat reduced over those for the NACA 0012 and NACA 16-012 airfoils, the differences are much less than the differences of the critical Mach numbers. These differences have already been explained as due to the two-dimensional characteristics, which of course cannot be separated from the critical Mach numbers.

The reason for this consistency of choking Mach number may be seen from a comparison of figures 5 and 9, showing the Mach number fields for the NACA 0012 airfoil of 2.88-inch thickness and for the 2.88-inch-diameter circular cylinder, respectively. In figure 9(b) at a stream Mach number of 0.547 the line of Mach number unity has already appeared, but in most of the surrounding region between the cylinder and the wall a moderate

Mach number of 0.65 or 0.70 prevails. With increase in stream Mach number the sonic line is extended, but because of the moderate Mach number in the surrounding region a considerable increase in mass-flow rate and, therefore, in stream Mach number can occur before the line of Mach number unity reaches the wall and choking occurs. With the high-critical-speed airfoil, on the other hand, a Mach number of 1.0 does not appear in the field until a stream Mach number of 0.637 has been reached, and the Mach number in the surrounding field between the airfoil and the wall is already 0.90 or 0.95. Only a small increase in mass-flow rate and of stream Mach number is therefore possible before the sonic line reaches the wall and choking occurs. If the flow were truly one dimensional in both cases, the choking Mach number would be the same for the circular cylinder as for the airfoil.

Two other airfoil characteristics, position of maximum thickness and chord length, were also found to have little effect on the choking Mach number. The effect of position of maximum thickness is seen from comparison of figures 5 through 8; the effect of chord length, from comparison of figures 9 and 10.

The Mach number fields obtained in the present investigation provide a basis for discussion of the choking phenomena theoretically derived in reference 9, in which choking is found to occur when the supersonic region near the model has spread so far that a further rise in speed through it causes such a reduction in mass flow through the supersonic region as to neutralize the increased mass flow in the regions nearer the walls. Development of this idea leads to the conclusion that, after choking has once been reached, the application of additional power with establishment of the sonic line entirely across the channel causes a reduction in mass flow and therefore also of stream Mach number. If any such effect had occurred in any of the present tests, the maximum stream Mach number M_s might have been expected to correspond to some Mach number field in which the supersonic region covered only part of the space between the airfoil and the wall. Examination of figures 5 through 11 and 13 shows that in no case, with the exception of the flat plate, which is not a proper shape on which to base conclusions regarding the flow about a solid, does the stream Mach number fail to increase monotonically until the supersonic flow has been established entirely across the channel. The argument of reference 9 cannot for that reason be considered disproved, however, because other factors, notably the floor boundary layer, strongly affect the flow. Nevertheless, brief consideration will show that the effect, if it exists at all, will be much less pronounced than indicated in reference 9.

First, the fields calculated in reference 9 are free fields and do not include the effects of the tunnel-wall interference. The inclusion of the wall interference would result in a more uniform field between the model and the wall and thus leave less room for variations in the mass-flow rate per unit area at different positions between the wall and the model. In other words, the theory neglects the tendency of the flow to approach the one-dimensional form as choking is approached. Second, the theory of reference 9 apparently assumes that the sonic region develops symmetrically, whereas actually, as may be seen from these tests (figs. 5 through 13 and 23), it develops mainly downstream and, in the narrowest section between the airfoil and the wall, the Mach numbers near the airfoil remain so near to unity that the mass-flow rate per unit area is nowhere much different from the maximum. From these two considerations the conclusion of the preceding paragraph easily follows. Much more important effects are due to the floor boundary layer.

The boundary-layer thinning previously illustrated in figure 20 seriously affects the choking Mach number. One-dimensional theory predicts a choking Mach number of 0.602 for a 2.88-inch-wide model in a 20-inch channel, but the experimental data for the airfoils, figures 5 through 8, show a maximum indicated stream Mach number of about 0.650 for the choking condition. This value is possible if the boundary layers thin between the point at which the stream Mach number is determined and the point of minimum tunnel area. The effect of boundary-layer thinning on choking Mach numbers, both for the water channel and for wind tunnels is shown in figure 24. These curves were calculated by assuming one-dimensional flow in the channel with elemental changes in the boundary-layer thickness between the point of stream Mach number determination and the point of minimum tunnel section. They are presented in terms of percentage change of tunnel area S for the wind tunnels and percentage change of total water depth h_o for the water channels. If the change in displacement thickness $\Delta\delta_h{}^*$ in the water channel had been expressed in terms of the water depth $h_{cr}\left(= \frac{2}{3}h_o\right)$ at the position of Mach number unity instead of on h_o, the two sets of curves would have appeared very much alike except for a slight difference due to the difference in ratio of specific heats γ. The curves show that slight changes in the boundary-layer thickness considerably change the choking Mach numbers, especially for small models.

A quantitative application of figure 24(b) may be made by using the floor boundary-layer increments observed in figure 20(a). The average floor boundary layer at the point of stream Mach number determination was 0.03 inch, that across the throat is -0.01 inch, a change of -0.04 inch. This value divided by the total head, which was 1.2 inches, gives a change of -3.3 percent. When this correction is made, the choking Mach number of the 2.88-inch-thick airfoils in the 20-inch-wide channel becomes 0.657 which in consideration of the probable two-dimensional effects is considered in good agreement with the experimental value of 0.650.

Another related effect of the boundary-layer thinning is progressive choking, that is, a further increase (with increasing pressure difference between sections upstream and downstream from the test section) in the indicated stream Mach number after the line of sonic velocity has reached the wall. Such an effect would be possible if, as the pressure (or water depth) downstream is reduced, the boundary layer were "swept out" and thinned under the sonic line, since the mass flow and thereby also the indicated stream Mach number would by this thinning be increased. The usual simplified boundary-layer theory does not, however, indicate the possibility of any such behavior, because in this theory the boundary layer at any given section is assumed not to depend on any downstream conditions. In order to investigate this effect, boundary-layer measurements were made in the empty channel with varying power to the drive motor. The results are shown in figure 21, where "barely choking" indicates an amount of power just sufficient to bring the sonic line to the wall, "medium choking" indicates somewhat greater power, and "full choking" indicates the maximum power condition. The "subsonic" condition corresponds to a stream Mach number of about 0.95. In these tests the total head in the tank upstream from the test section was maintained constant, so that increasing power corresponds to decreasing water depth in the downstream region.

A progressive thinning of the boundary layer occurs as the power is increased. (See fig. 21(b).) This "sweeping out" of the boundary layer increases the slope of the effective floor boundary-layer surface (fig. 21(a)) and causes an increase in the supersonic Mach numbers downstream from the throat (fig. 21(c)). A comparison of figures 21(b) and 21(c) shows that the boundary layer at the sonic line becomes thinner with increasing power so that the mass flow in the channel must be increased. A cause for the progressive choking has thus been determined.

A possible explanation can be given of the process by which the boundary layer is swept out. Suppose that the "barely choking" condition exists. Let the water depth at a given section be reduced by some means as by a movement of the normal shock downstream. This reduction in depth cannot be transmitted upstream because the velocity of propagation of the surface wave is less than the velocity of the upstream flow. The velocity of transmission of pressure (as at the channel floor) within the fluid is sufficiently great, however, as to be practically instantaneous, and a pressure gradient therefore exists in the boundary layer under a region of depth reduction. The subsonic boundary layer in the next upstream section can thus be accelerated and thinned. This thinning in turn could reduce the water depth and by this process the effect could be progressively transmitted upstream.

The tests with the large airfoils, figures 5 through 8, show only a small effect of progressive choking. In the case of the 10-percent-thick, 5-inch-chord, biconvex, circular-arc airfoil in the 7.5-inch-wide channel (figs. 13(i) and 13(k)), however, the boundary-layer displacement is comparable to the solid displacement of the model and the choking Mach number increases from 0.795 to 0.830 as the downstream depth is decreased. With this same model in the 20-inch-wide channel (fig. 12) as the depth is decreased downstream the boundary-layer-displacement thinning exceeds the model displacement; the effective throat of the channel is moved upstream, and the model is left in a supersonic stream.

Because of the relatively large part of the total mass flow involved in the boundary layer, the boundary-layer effects here discussed are much more severe in the water channel than would be the case in normal wind-tunnel operation.

Effects of the Walls

The influence of the walls on the flow about an airfoil may be seen from figure 13, which gives a comparison at the same indicated stream Mach numbers M_s of the Mach number fields about the 10-percent-thick, 5-inch-chord, biconvex, circular-arc airfoil at zero lift in 7.5-inch and 20-inch-wide channels. The immediately noticeable effect of moving the walls closer to the airfoil is the increase in the Mach numbers in the immediate vicinity of the airfoil; and this effect is the same over the entire stream Mach number range from subcritical to choking. This behavior refutes the argument sometimes advanced that, because a reduction in stream-tube cross section corresponds to a reduction in velocity with supersonic flow as contrasted to an increase in velocity with subsonic flow, the effect of the constriction of the flow by the channel walls would be quite different in transonic flow from its effect in entirely subsonic flow.

The true situation may be clearly understood from consideration of figure 13. The supersonic region near an airfoil is produced by the surrounding subsonic pressure field; but the subsonic part of the flow field comes into contact with the channel walls and is strongly influenced by them. The supersonic region, on the other hand, is influenced by the walls only through their effect on the subsonic flow; and this effect is to increase the pressure differences, which, in turn, cause the intensification and growth of the supersonic regions. In this manner, the wall constriction has the effect of increasing the Mach number values in the field about an airfoil with transonic flow just as with entirely subsonic flow.

The Mach number fields of figures 13(c) and 13(d) may be compared with the theoretical fields at a slightly different stream Mach number in figures 17(a) and 17(b). Although the experimental fields are distorted with respect to the theoretical fields, the qualitative effects of the wall constriction are the same. The distortion in the experimental fields can be explained as largely due to the changes in thickness of the floor boundary layer in regions of accelerating and decelerating flow, which can be shown to shift the lines of constant Mach number toward the regions of higher velocity, particularly in the region near the wall where the effect of the model is least compared to that of the boundary layer. The tendency of the boundary layer to thicken toward the rear of the model produces an asymmetry of the flow field of the same nature as that produced by the wake. Also the floor boundary-layer thinning in the vicinity of the airfoil tends to relieve to some extent the constriction of the walls. This effect explains the fact that at some distance from the airfoil the Mach numbers in the theoretical field slightly exceed those in the experimental field, although the stream Mach numbers are somewhat less. In the 20-inch channel the effect of the boundary-layer thinning is evidently greater than that of the wall constriction, so that the flow behaves somewhat as if the model were in a free jet.

The effect of the wake is seen from a comparison of the theoretical fields in figures 16 and 17. The Mach numbers in the downstream portion of the field are increased and the fields thereby rendered asymmetrical.

Comparison of the theoretical compressible fields of figures 16 and 17 with the incompressible fields of figures 18 and 19 shows that the tunnel-constriction effects in the compressible flow are of the same nature as those which occur in the incompressible flow.

Another comparison with theory was obtained from several unpublished theoretical Mach number fields computed by Dr. Howard W. Emmons of Harvard University under the sponsorship of the National Advisory Committee for Aeronautics. The fields were those about a 5-inch-chord NACA 0012 airfoil in an 18-inch-wide air channel and in a free air stream. The computation for compressible flow at an assumed stream Mach number of 0.750 and with zero lift was effected by means of the relaxation process. No wake was assumed, but otherwise the nature of the wall interference was the same as that observed in figure 13.

The question now arises whether, as for subcritical flow, the effect of the channel walls in supercritical flow can be represented as that of an increment in the free-stream Mach number, provided the model is reasonably small relative to the channel width. In order to obtain a qualitative answer to this question, Mach number fields about the 5-inch-chord, 10-percent-thick, circular-arc airfoil in the 7.5-inch and 20-inch-wide channels are shown in figure 14 for indicated stream Mach numbers corresponding to the same maximum Mach numbers in the two fields. If the 20-inch channel is considered to give essentially free-stream results, the Mach number increment representing the effect of the walls is the difference between the indicated stream Mach number in the 20-inch channel and that in the 7.5-inch channel.

With this method of comparison, the Mach number fields in the vicinity of the airfoil do indeed appear very much the same, though a certain amount of distortion exists. At high stream Mach numbers, the field Mach numbers near the leading and trailing edges of the airfoil in the 7.5-inch channel fall off relative to those in the 20-inch channel, an effect which appears necessary because of the smaller stream Mach numbers in the 7.5-inch channel. Figure 14 shows that the lines of constant Mach number greater than the stream Mach number which loop back to the airfoil extend farther into the field in the 20-inch channel than in the 7.5-inch channel. This effect is opposite to that shown in the theoretical fields (figs. 16(c) and 16(d), 17(c) and 17(d), 18(c) and 18(d), and 19(a) and 19(b)), and is believed to be an error caused by the boundary-layer behavior. For any two fields compared, the Mach numbers in the vicinity of the model are not much different. Since the stream Mach number is less in the 7.5-inch channel than in the 20-inch channel, the boundary-layer growth between a stream Mach number point and any point with given greater field Mach number must be less for the 7.5-inch than for the 20-inch channel. Moreover, the effect of the boundary-layer changes on the subsonic Mach numbers is relatively greater near the wall than near the model, where the model shape is most important. The lesser growth in the case of the 7.5-inch channel, therefore, tends to counteract the constriction effect and causes the high subsonic regions to extend a lesser distance out from the airfoil than they do in the 20-inch channel. These same boundary-layer effects also produce the result that the choking Mach number in the 7.5-inch channel corresponds to a stream Mach number greater than unity in the 20-inch channel (figs. 14(m) and 14(n)), which result is contrary to a conclusion contained in reference 10. Actually, theoretical considerations without boundary layer indicate that with this method of comparison the distortion due to the walls enlarges both the high-velocity regions near the center of the airfoil and the low-velocity regions near the ends and that the choking Mach number must correspond to a free-stream Mach number less than unity. Once choking has been attained, although the flow patterns with additional power, that is, a greater pressure drop across the test section (figs. 14(o) to 14(r)), can be roughly matched to those in the 20-inch channel, correction to free-stream Mach number appears impossible because further changes in indicated stream Mach number depend almost entirely on boundary-layer changes and cannot be definitely related to the Mach number distribution in the supersonic part of the flow. Up to the choking Mach number, the distortion at the surface of the airfoil is, surprisingly, less than in any other part of the field.

The surface Mach number distributions, shown in figure 15, agree very well for subcritical stream Mach numbers if compared on the basis of the same maximum Mach numbers. Almost the entire effect of the walls is represented simply by shifting the indicated stream Mach numbers. At Mach numbers greater than the critical, a considerable distortion appears in the sense of a rotation of the Mach number diagram for the 7.5-inch channel. Such a rotation is caused by the Mach number gradient due to wake blockage. By use of the measured wake displacements in conjunction with formula (21) of reference 3, corrections for the Mach number gradient were therefore applied. Comparisons of the surface Mach numbers for indicated stream Mach numbers of 0.712 and 0.747 are shown in figure 25. Although the agreement is considerably improved, particularly over the rear of the model, the correction is insufficient to bring the Mach number distributions into coincidence. The remaining discrepancy is perhaps due to the behavior of the floor boundary layer by which in the 20-inch channel on account of the greater value of stream Mach number the boundary-layer thickness near the surface of the airfoil is greater than in the

7.5-inch channel at the comparable Mach number. Another factor tending to distort the flow is the distortion of the interference velocities which, as shown in reference 11 for subsonic flow, fall off relatively ever more rapidly toward the leading and trailing edges of the airfoil as the Mach number is increased, a compressibility effect similar to increasing the size of the model relative to the channel. (See, for instance, reference 12.)

The agreement of the matched curves suggests that it may be possible up to a Mach number near choking to correct for the constriction of the channel walls with a simple shift of the stream Mach number, though such correction may not be practical. (Compare reference 10.) If the effective free-stream Mach numbers corresponding to the indicated stream Mach numbers M_s in the 7.5-inch channel are taken to be equal to the indicated stream Mach numbers in the 20-inch channel for the comparable Mach number fields of figure 14, the effective stream Mach numbers can be obtained as a function of the indicated stream Mach numbers. In figure 26, the effective free-stream Mach numbers so obtained are compared with a curve derived from the theory of reference 3. With Mach numbers greater than the critical, this curve is to be considered an extrapolation, inasmuch as the theory is strictly applicable only to entirely subsonic flow. Above the critical Mach number, the experimental values are seen to depart ever more strongly from the theoretical curve, thus indicating a more powerful tunnel-wall interference, as the Mach number is increased. This effect, which has also been observed in wind tunnels (see, for instance, references 10 and 13), is not surprising in consideration of the fact that the subsonic region upon which the wall interference is primarily effective becomes increasingly narrow as the supersonic region approaches the walls. The subsonic theory should more accurately express the correction if the model were made smaller so that the effective Mach number would depart less from the indicated stream Mach number. Since the model chord to tunnel height ratio used in these experiments is much larger than that used in current high-speed-tunnel tests, it may be expected that the extrapolation of the subsonic theory will generally be more accurate than is indicated by these tests. A transonic theory of the tunnel-wall-constriction effect, which would take account of the presence of the supersonic regions, is needed; but, in any case, the crowding of the effective stream Mach numbers in the range near choking is such that accurate correction very near choking appears impractical. Both for this reason and because of the difficulty in correcting for distortion along the chord of the model, which is accentuated by the increasing departure from potential flow as the Mach number is increased beyond the critical, adequate corrections can be applied only if the model dimensions relative to the tunnel dimensions are continuously decreased as the indicated stream Mach number is increased beyond the critical value. What is needed if models of reasonable size are to be tested at Mach numbers approaching unity is some method by which the tunnel-wall-constriction effect and therewith, also, the choking phenomena are automatically eliminated.

CONCLUSIONS

Investigation by means of the hydraulic analogy of transonic-flow fields about various airfoils in water channels of 7.5-inch and 20-inch width, with due consideration for the various factors affecting the results led to the following conclusions:

1. As choking was approached, the flow tended to approach the one-dimensional form. This tendency, in conjunction with the fact that near a local Mach number of unity the change in mass-flow rate approaches zero, explained the success of the one-dimensional theory in predicting choking Mach numbers.

2. Only in the extreme case of the flat plate normal to the flow was any evidence found of wake choking, and in that case the effect was confined to the region near the edges of the plate.

3. The thinning of the floor boundary layer between the airfoil and wall was sufficient to account for the excess of the experimental choking Mach number values over those predicted by the one-dimensional theory. It also caused considerable progressive choking when the boundary-layer displacement was comparable to the solid displacement of the model.

4. The channel-wall-constriction effect was of the same nature in subsonic streams with supersonic regions as in entirely subsonic flow, and this fact explained the ability of the subsonic theory of tunnel-wall interference to yield approximately correct results when extrapolated a little way into the transonic range.

5. Approximate correction for the constriction effects of the walls on symmetrical flows appeared possible with stream Mach numbers up to the first attainment of choking, and this correction could be largely effected simply by adding a Mach number increment to the indicated stream Mach number.

6. Because of the increasing distortion of the flow and because of the fact that the corrections became very large as choking was approached, accurate corrections in the supercritical range could be applied only if the size of the model relative to the tunnel size were greatly reduced with increasing Mach number. For the application of such corrections, a theory of tunnel-wall interference in the supercritical range is needed.

7. If models of moderate size are to be tested near a Mach number of unity, some method is needed by which the constriction itself and therewith, also, the choking phenomena can be eliminated.

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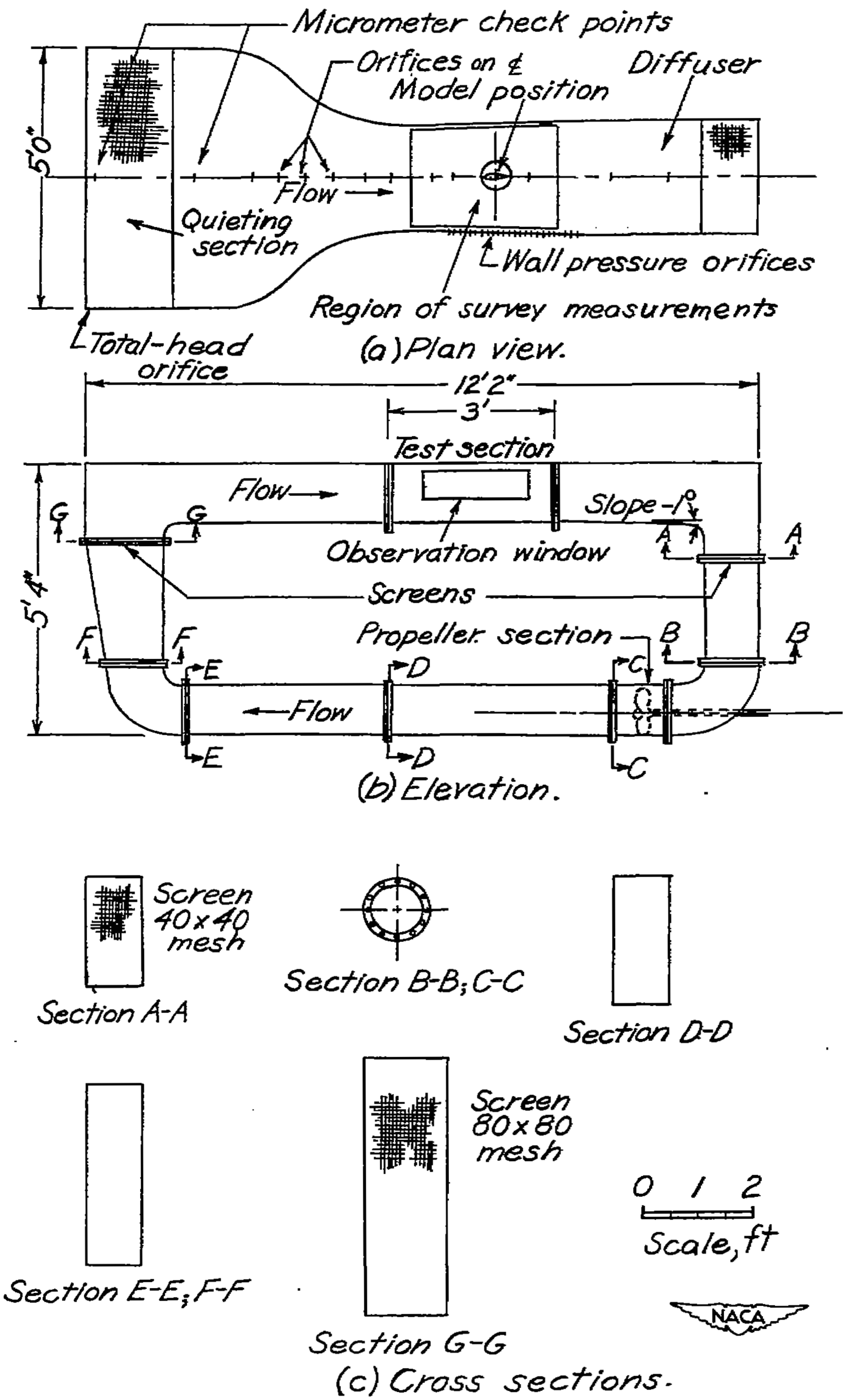


Figure 1.- Schematic view of water channel with sections.

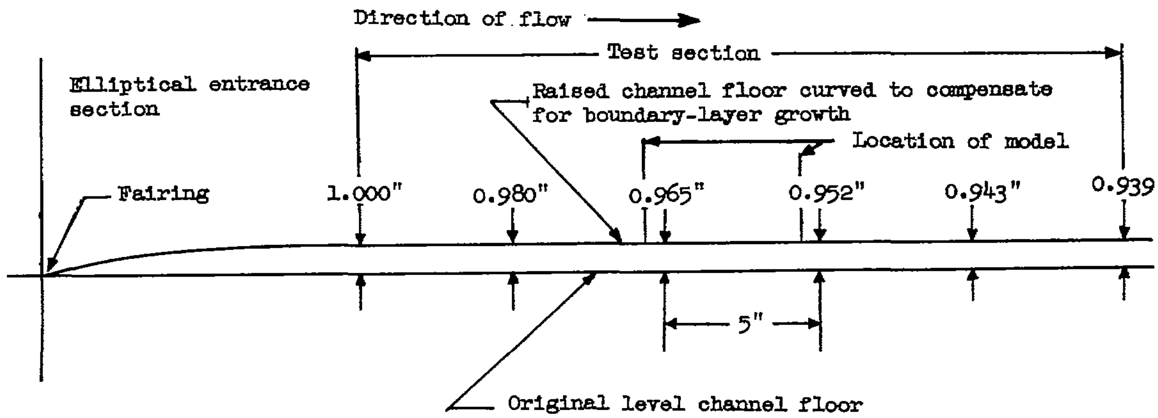


Figure 2.- Sketch of modified test section floor.

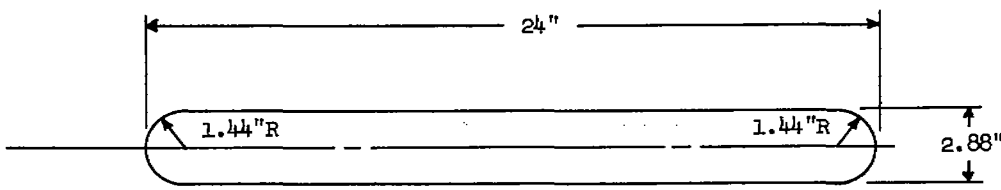
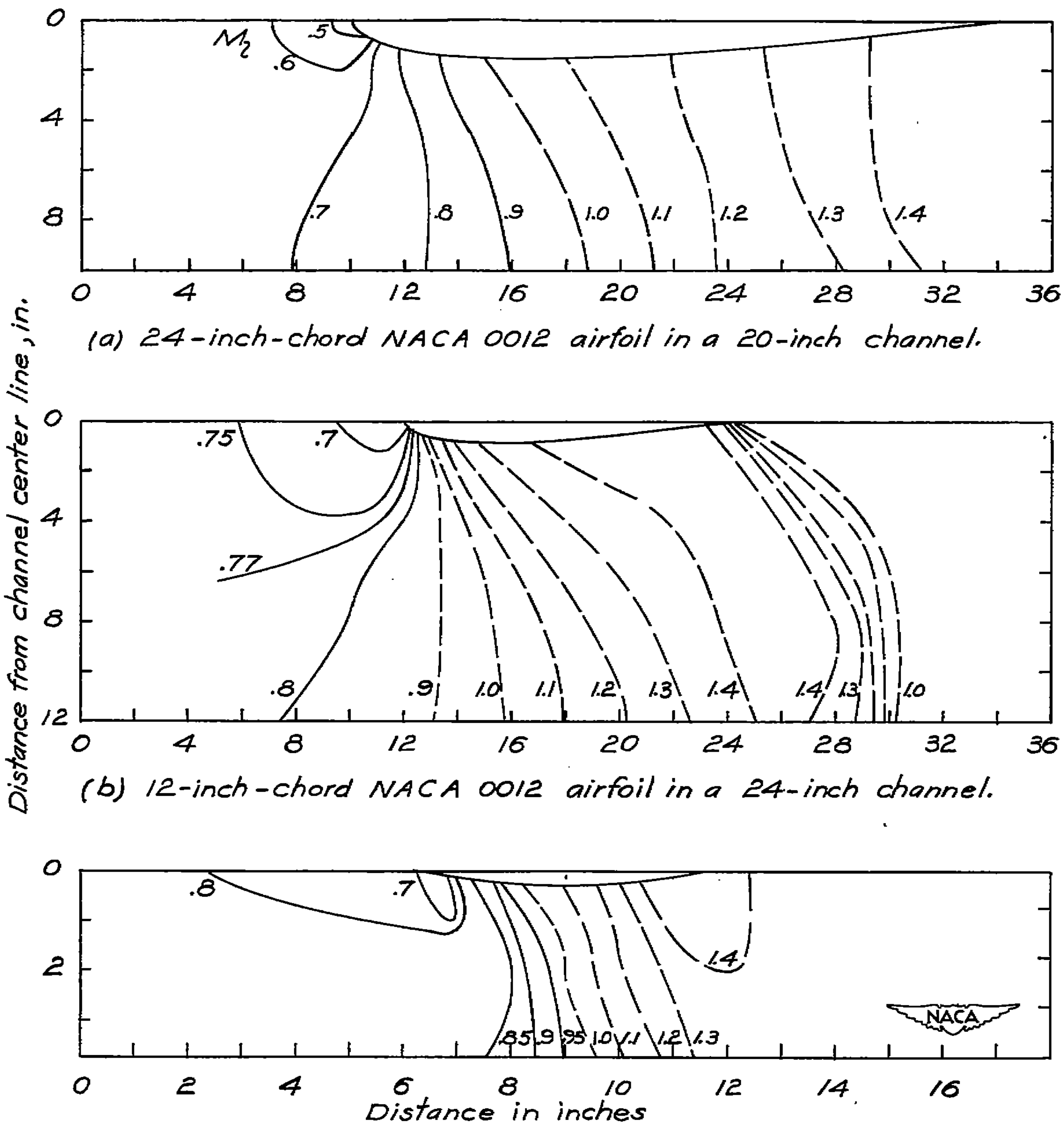


Figure 3.- Sketch of 24-inch-chord airfoil with parallel sides and cylindrical ends.



(c) 10-percent-thick, 5-inch-chord, biconvex, circular-arc airfoil in a 7.5-inch channel.

Figure 4.- Choking Mach number fields at various ratios of chord length to channel width showing approximately one-dimensional flow.

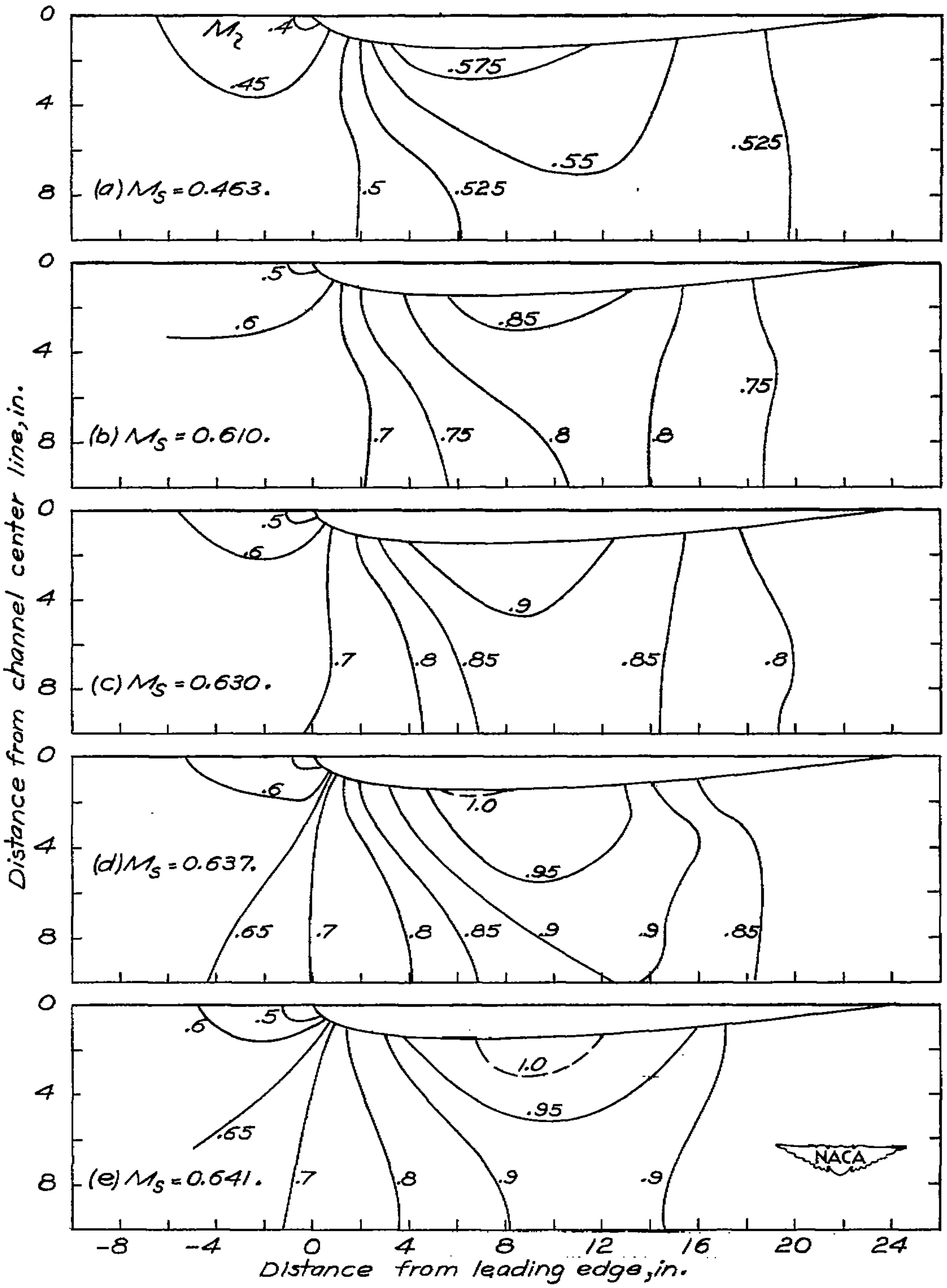


Figure 5.- Mach number fields about a 24-inch-chord NACA 0012 airfoil in a 20-inch channel.

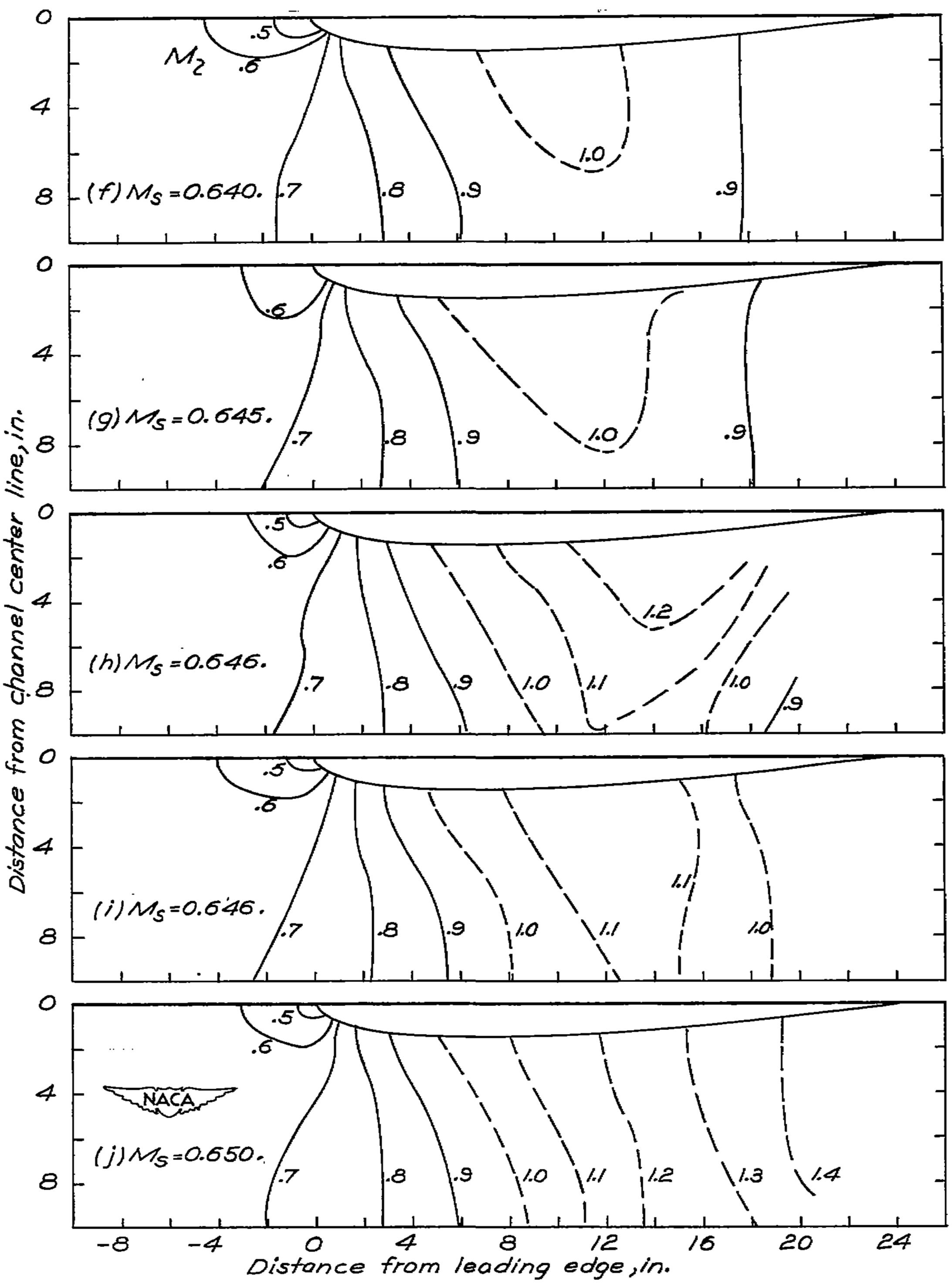


Figure 5.- Concluded.

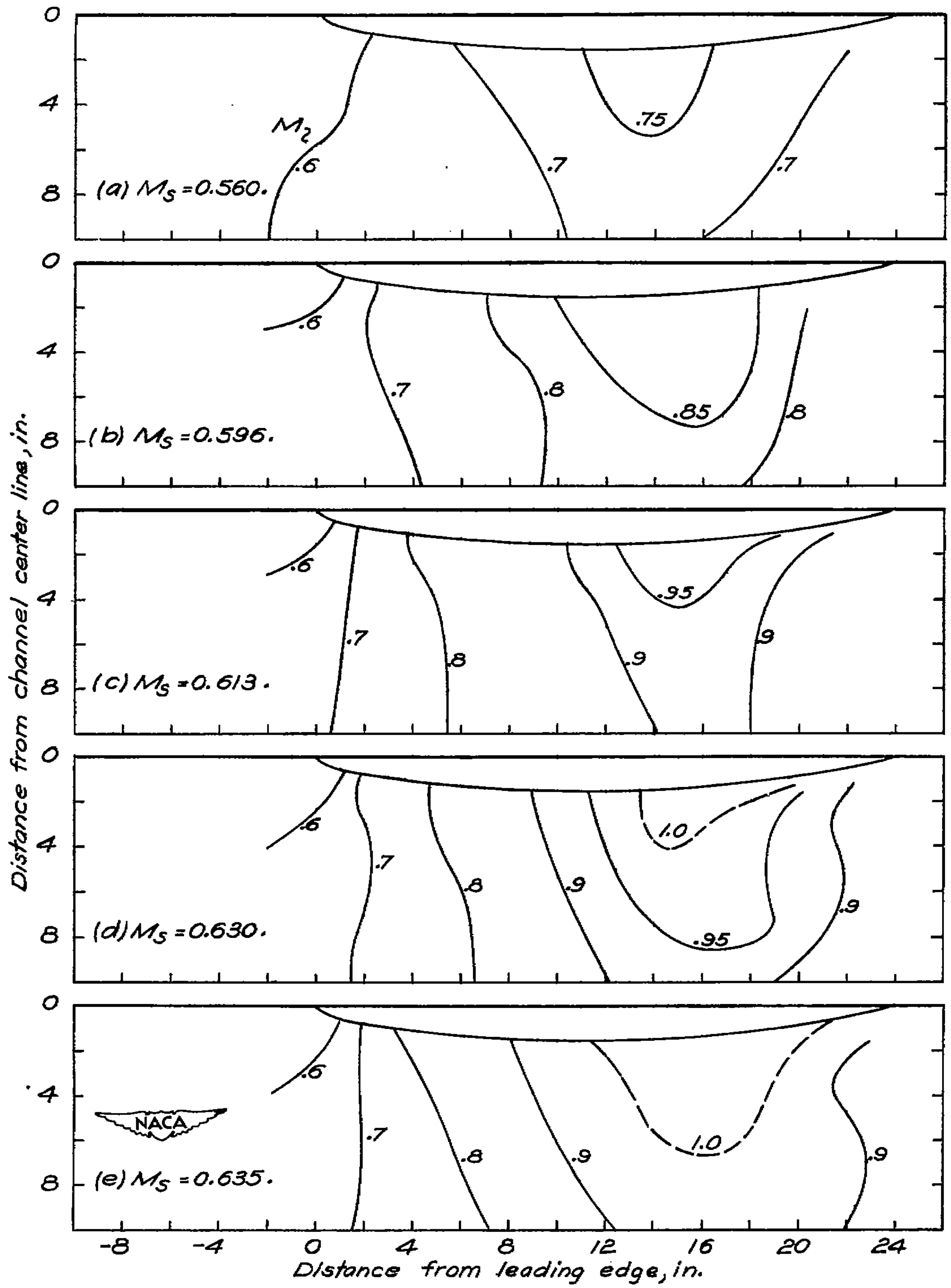
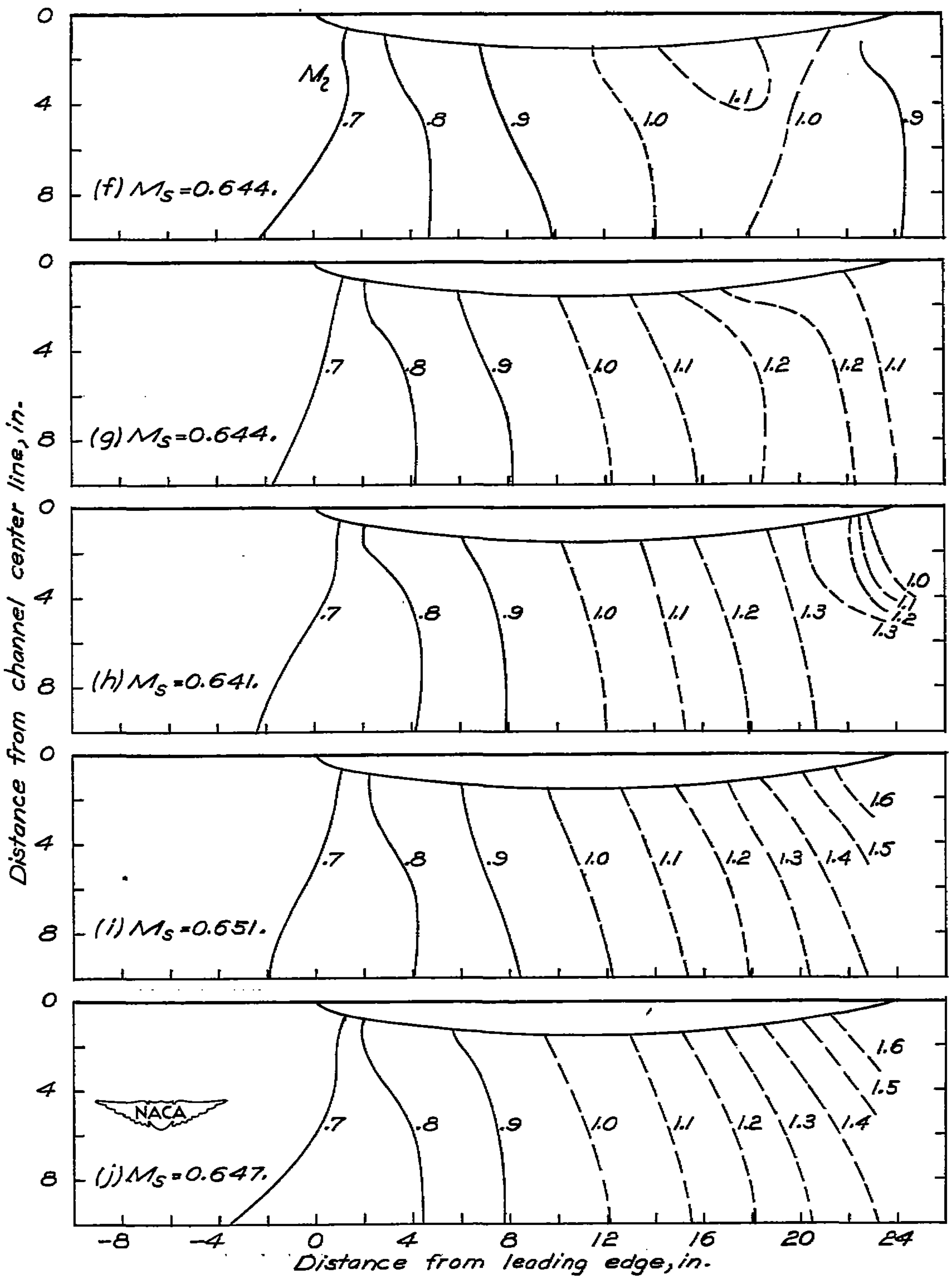


Figure 6.- Mach number fields about a 24-inch-chord NACA 16-012 airfoil in a 20-inch channel.



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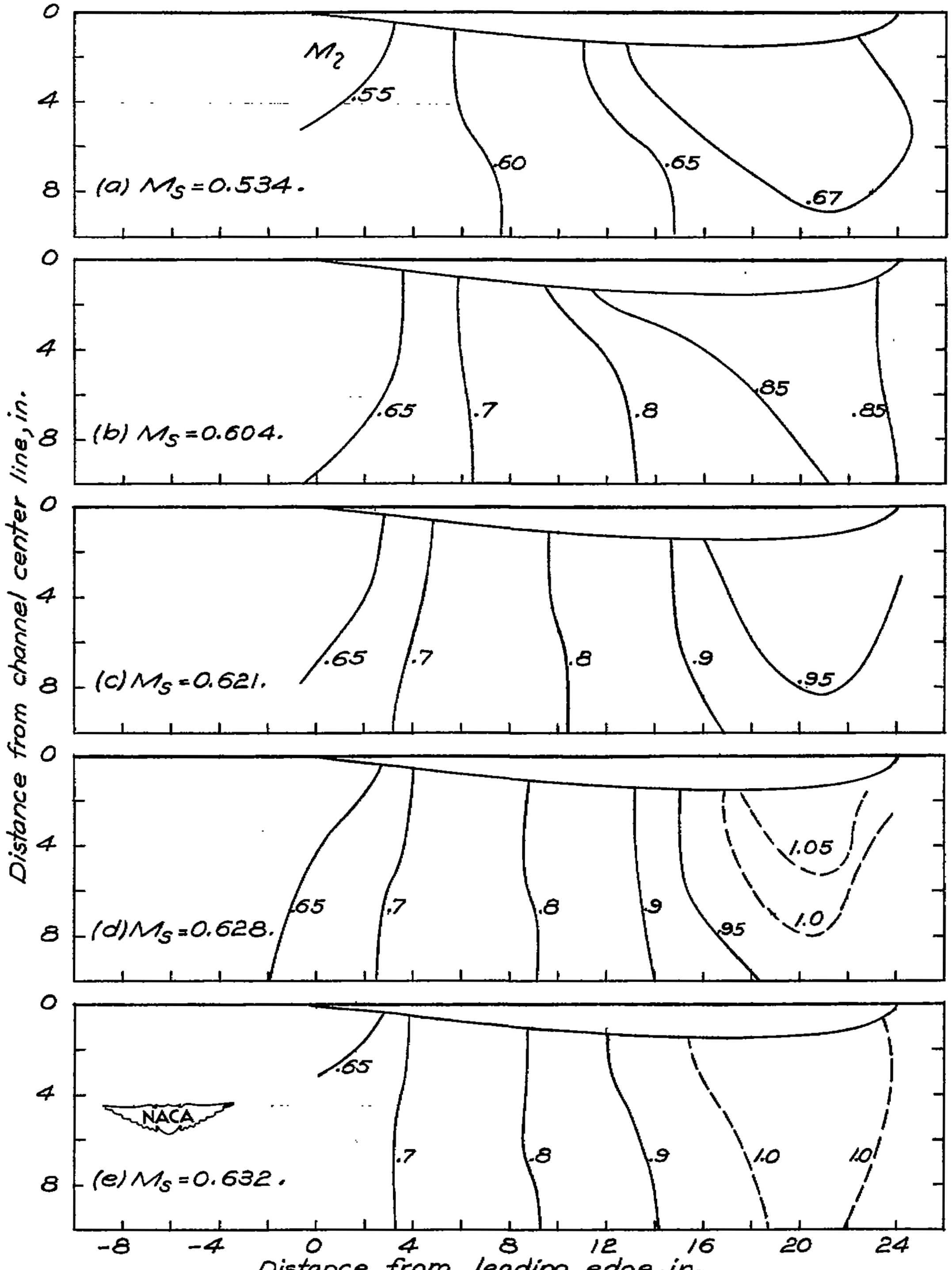


Figure 7.- Mach number fields about a 24-inch-chord NACA 0012 reversed airfoil in a 20-inch channel.

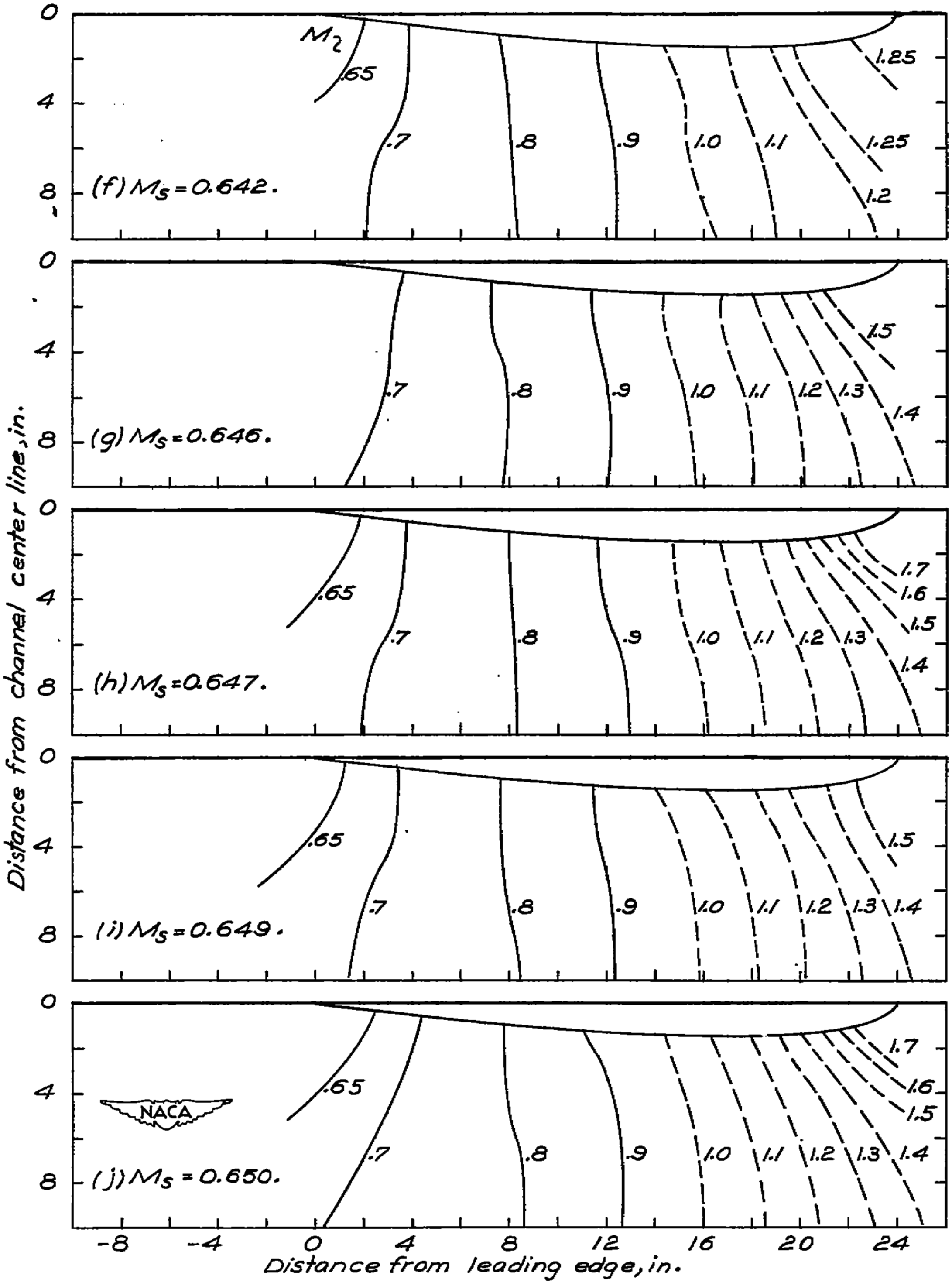


Figure 7.- Concluded.

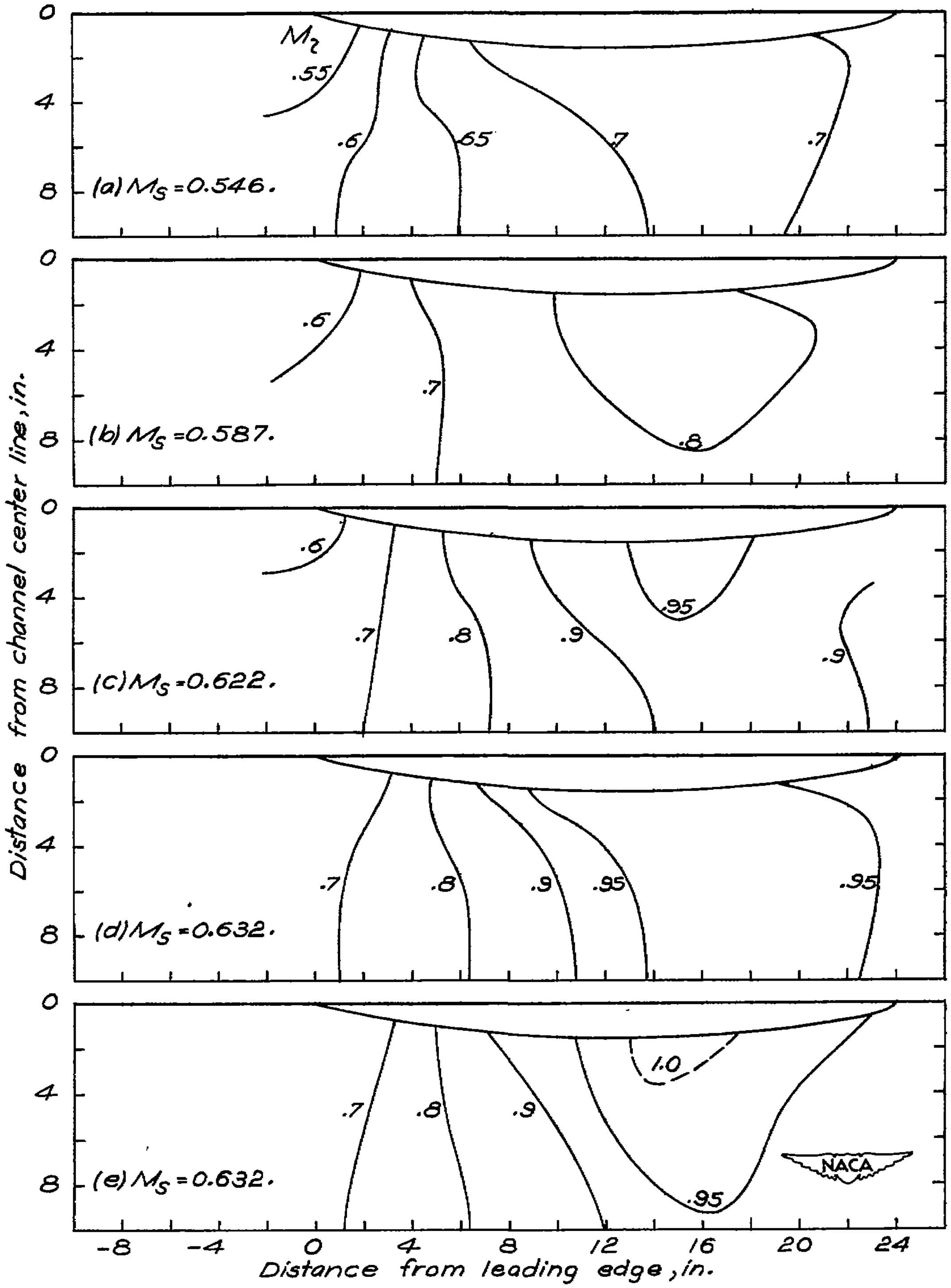


Figure 8.- Mach number fields about a 24-inch-chord NACA 16-012 reversed airfoil in a 20-inch channel.

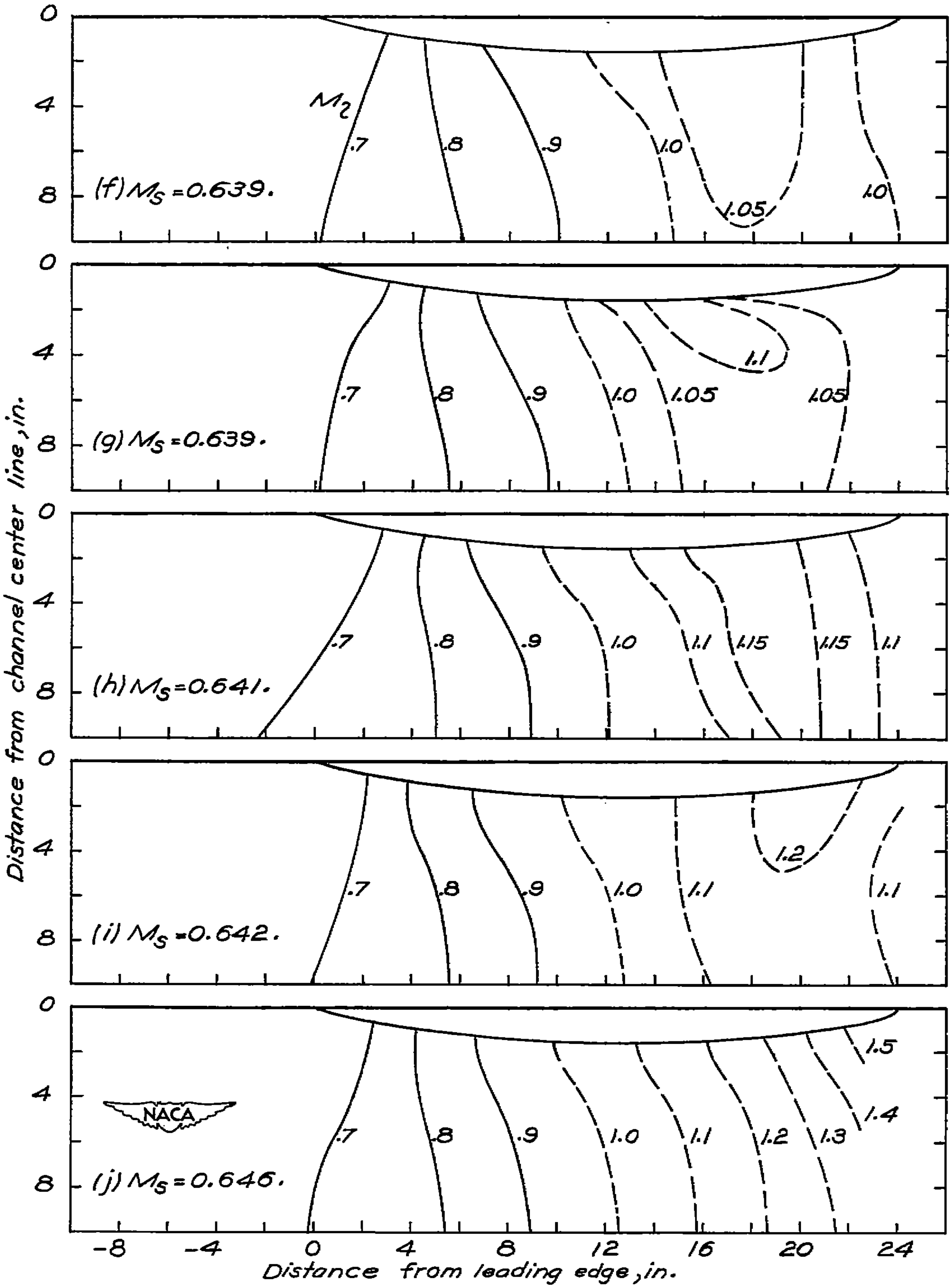


Figure 8.- Concluded.

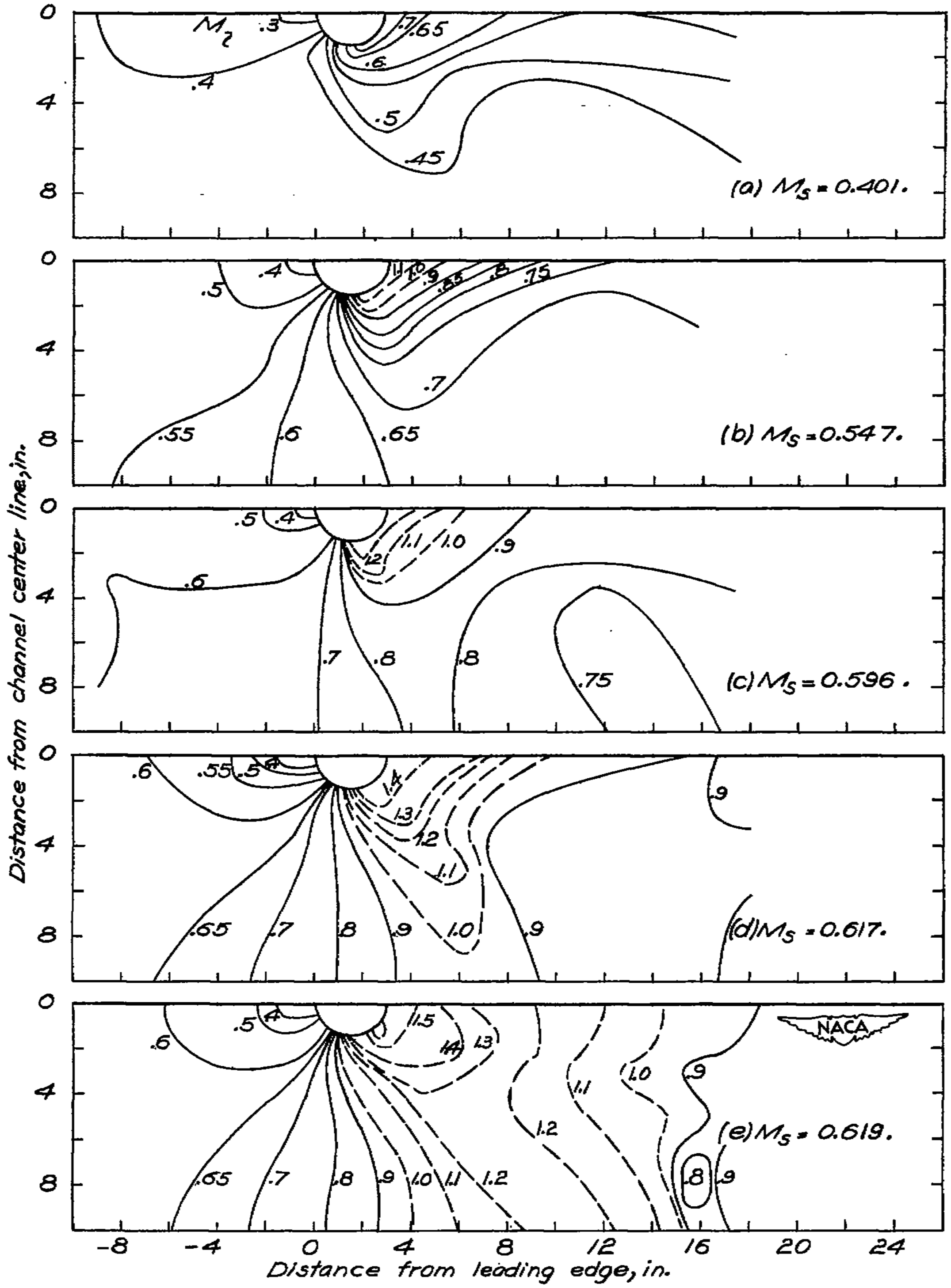


Figure 9.- Mach number fields about a 2.88-inch-diameter cylinder in a 20-inch channel.

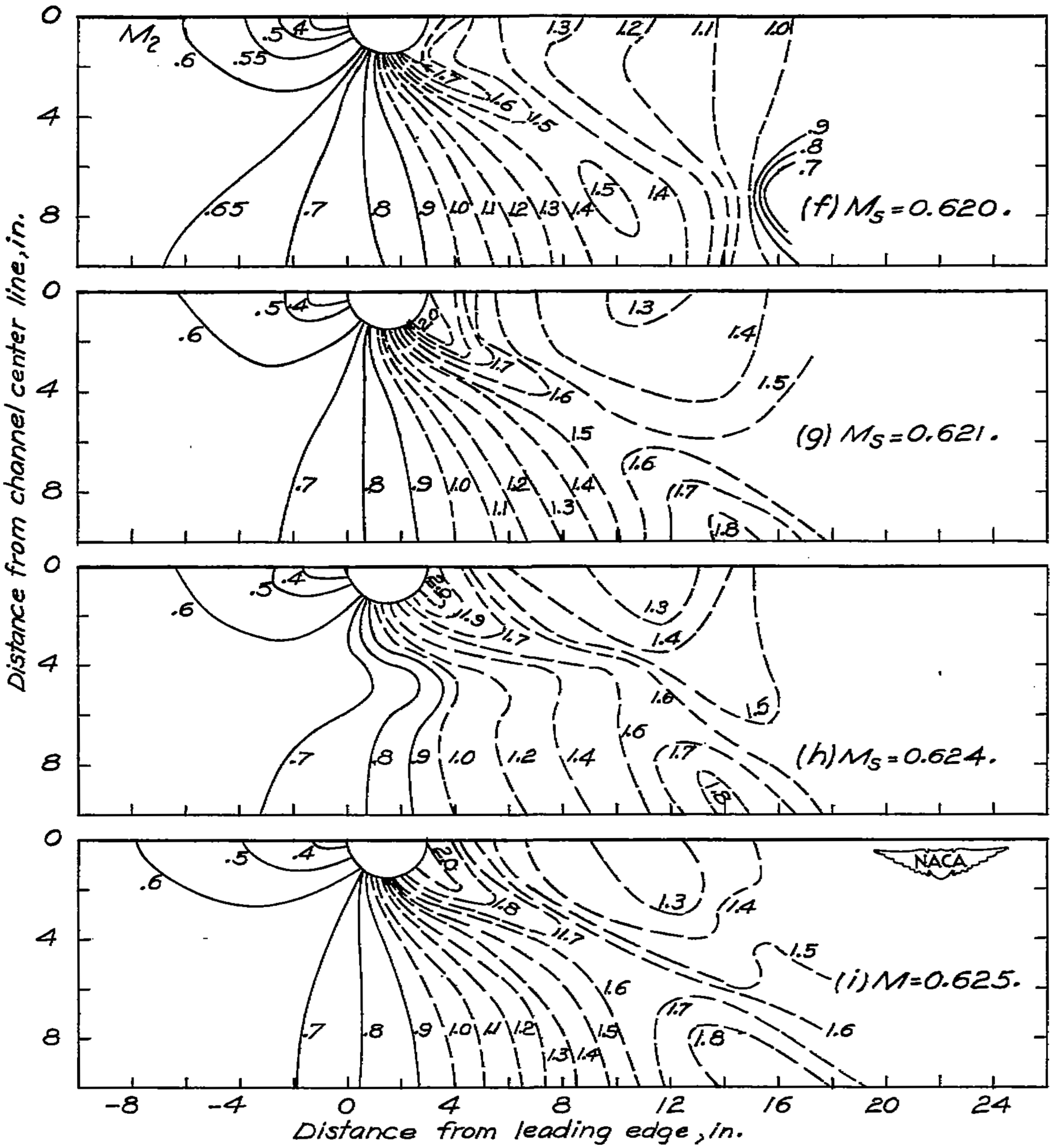


Figure 9.- Concluded.

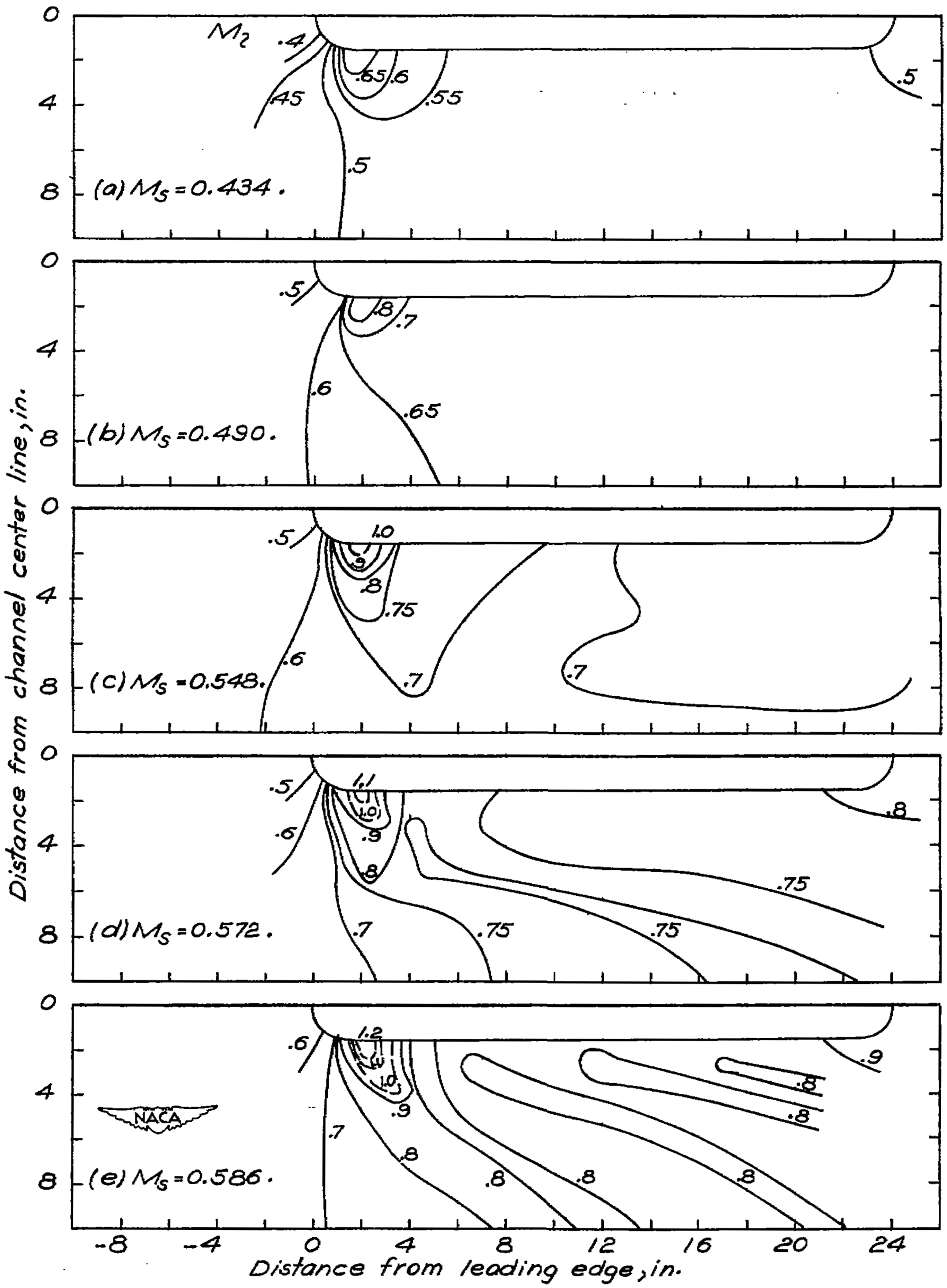
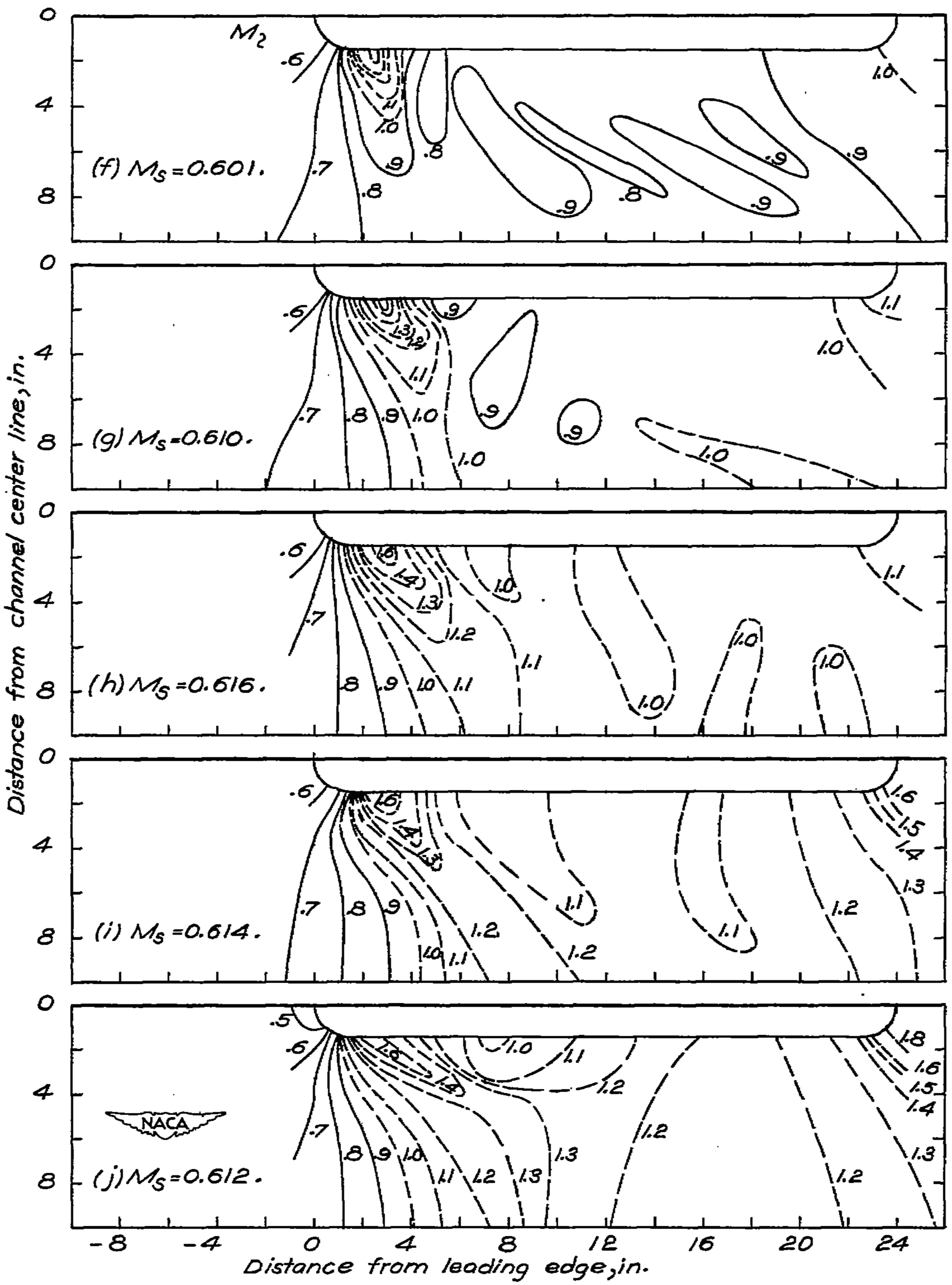


Figure 10.- Mach number fields about a 12-percent-thick, 24-inch-chord, parallel-side airfoil with circular ends in a 20-inch channel.



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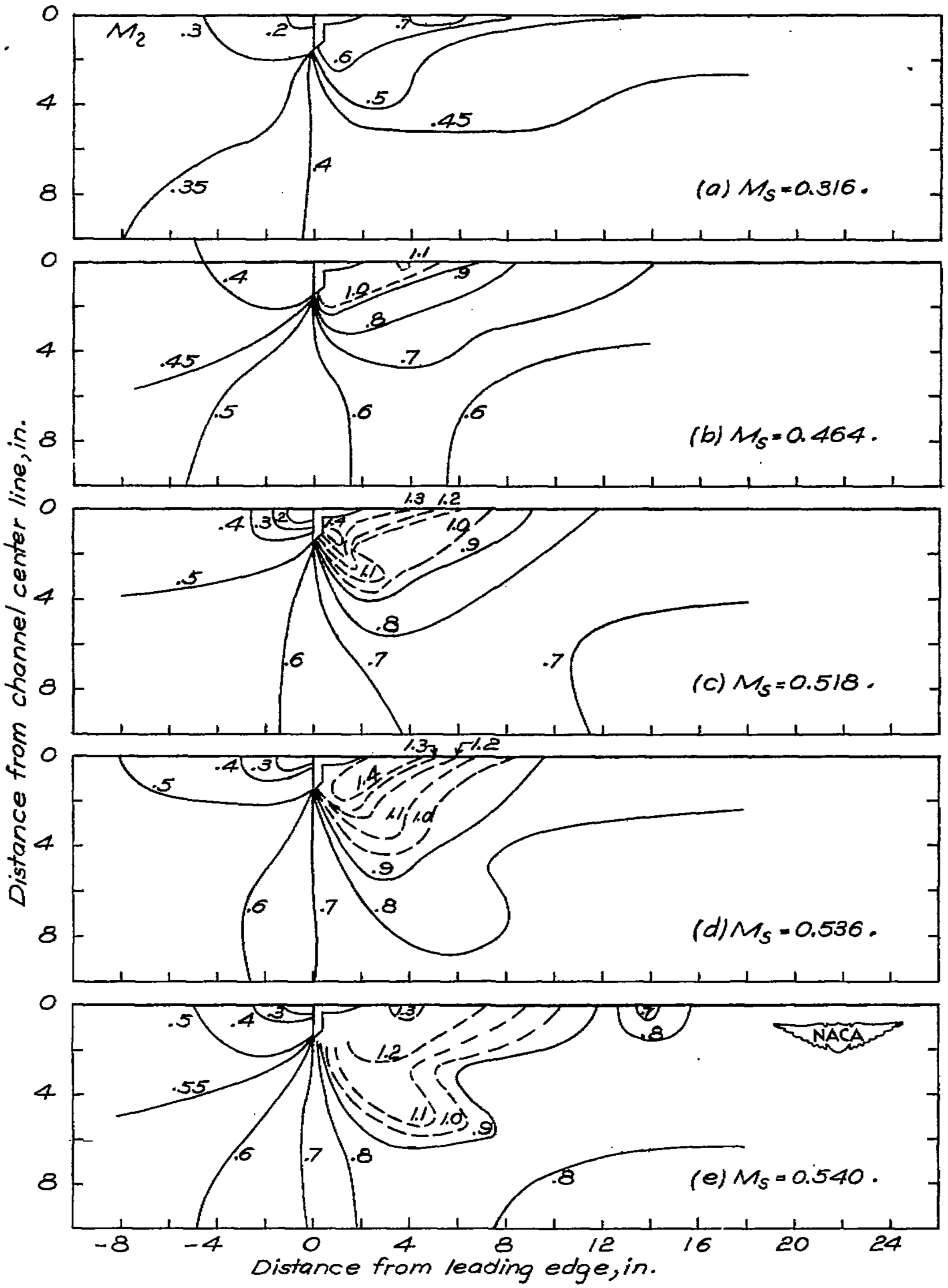


Figure 11.- Mach number fields about a 2.88-inch-wide flat plate in a 20-inch channel.

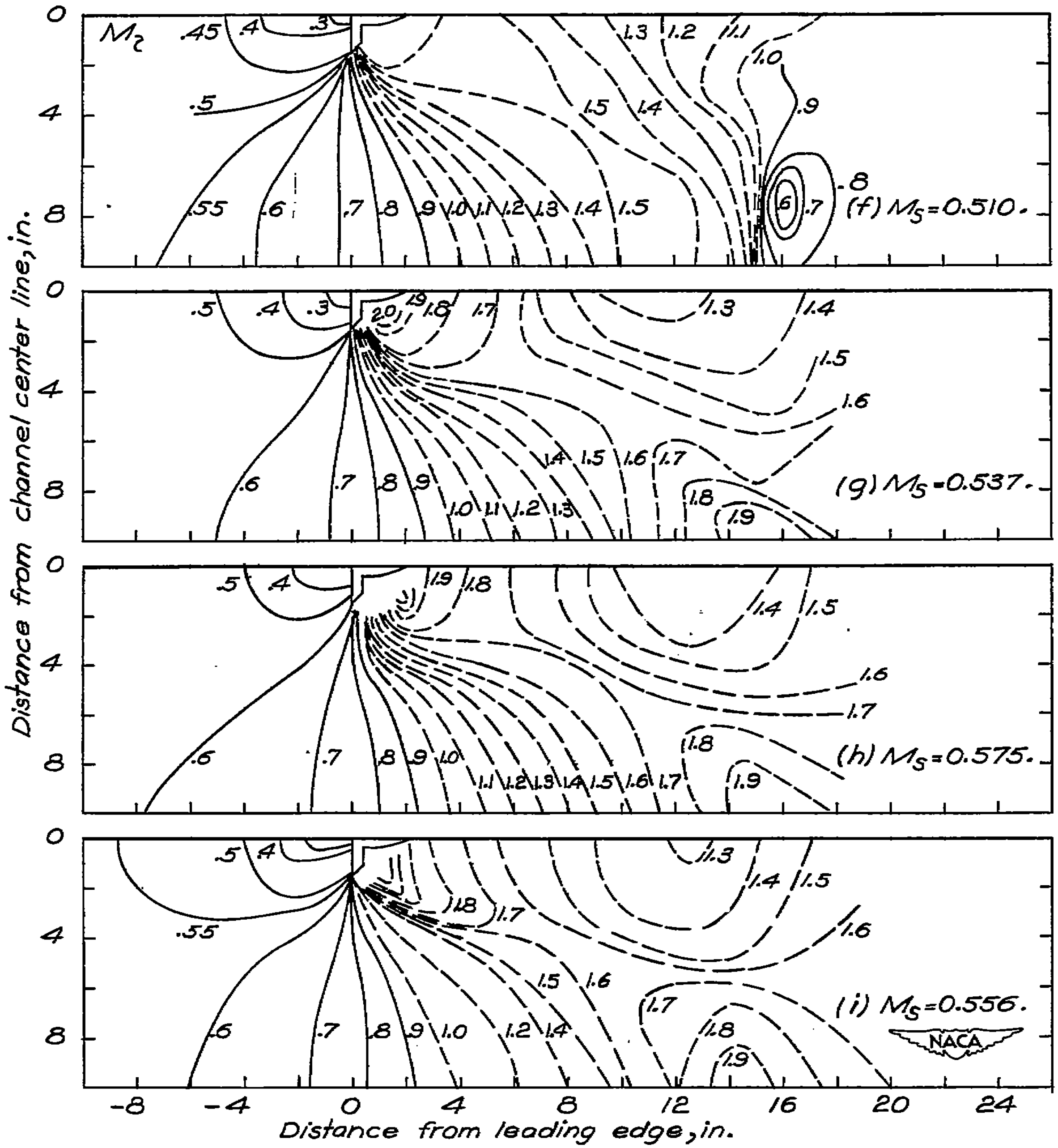


Figure 11.- Concluded.

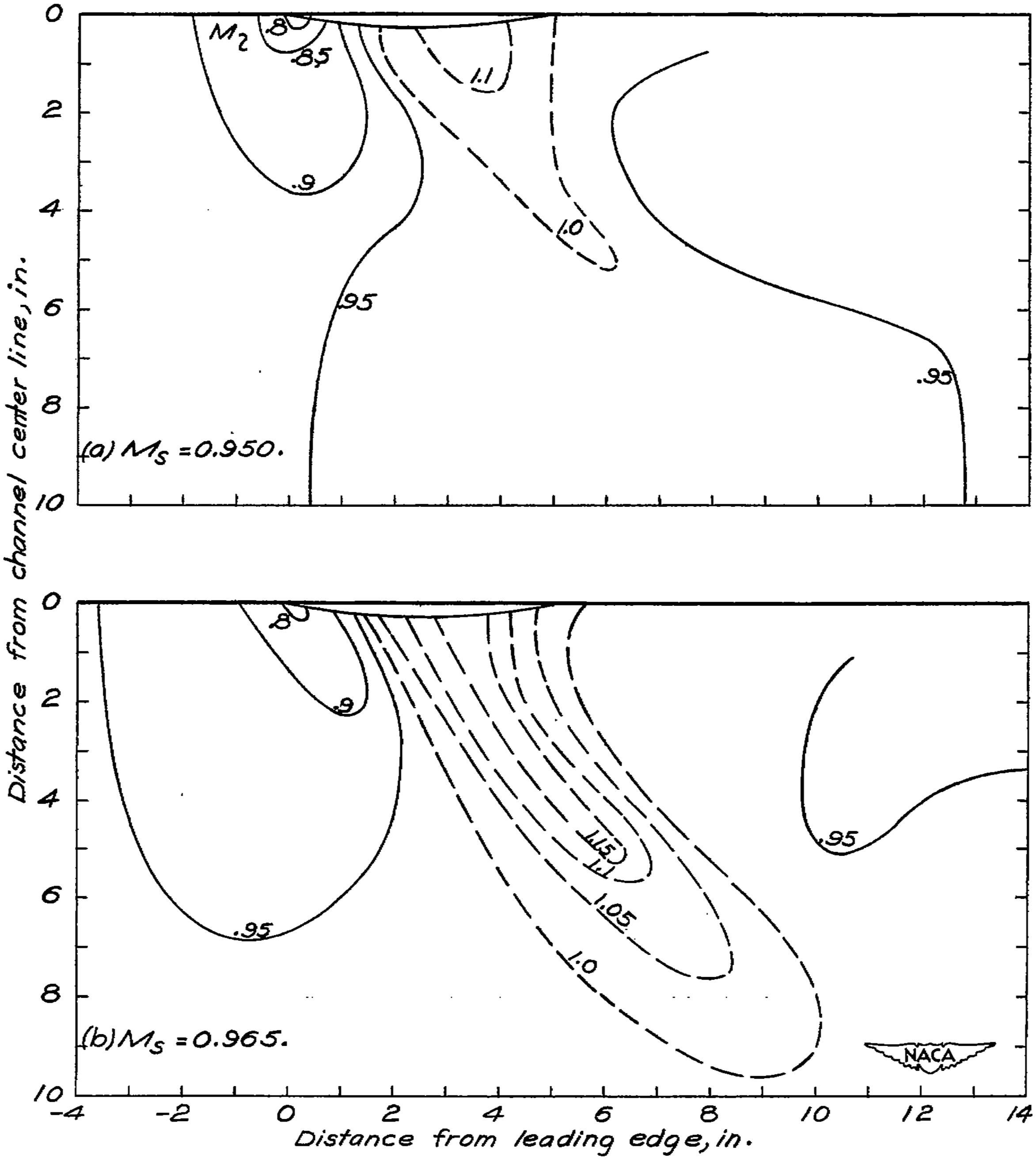


Figure 12.- Mach number fields about a 10-percent-thick, 5-inch-chord, biconvex, circular-arc airfoil in a 20-inch channel.

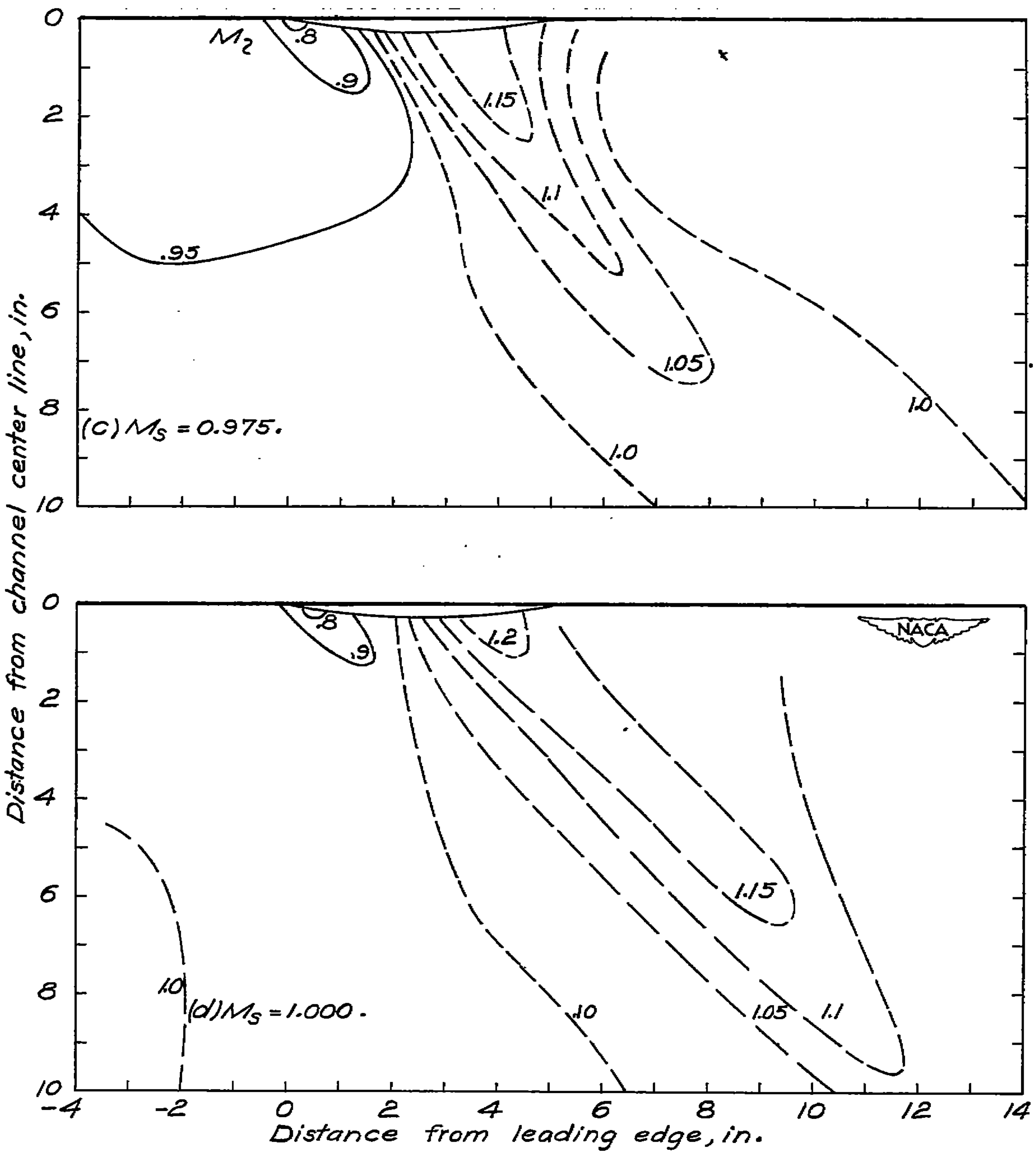


Figure 12.- Continued.

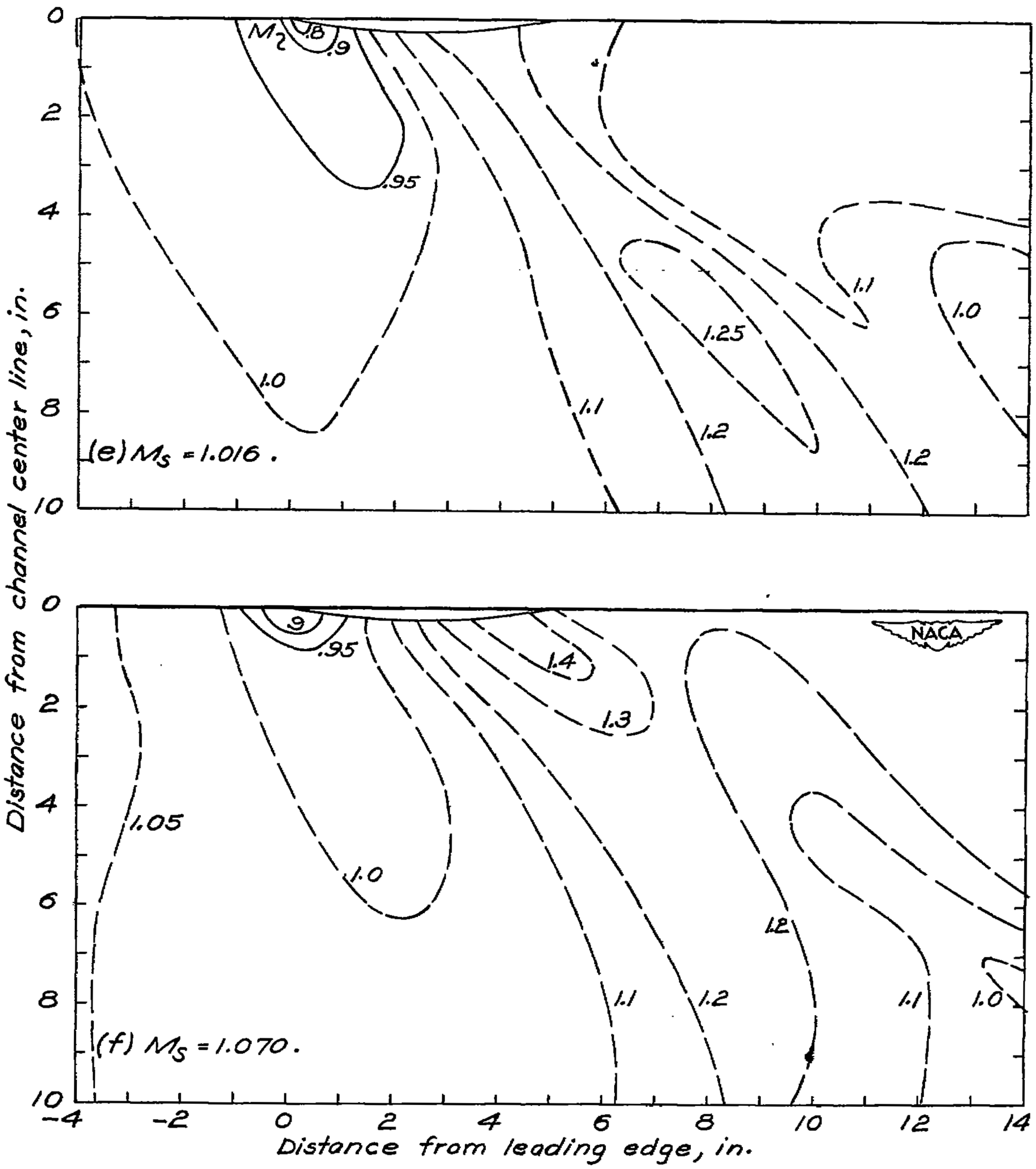


Figure 12.- Concluded.

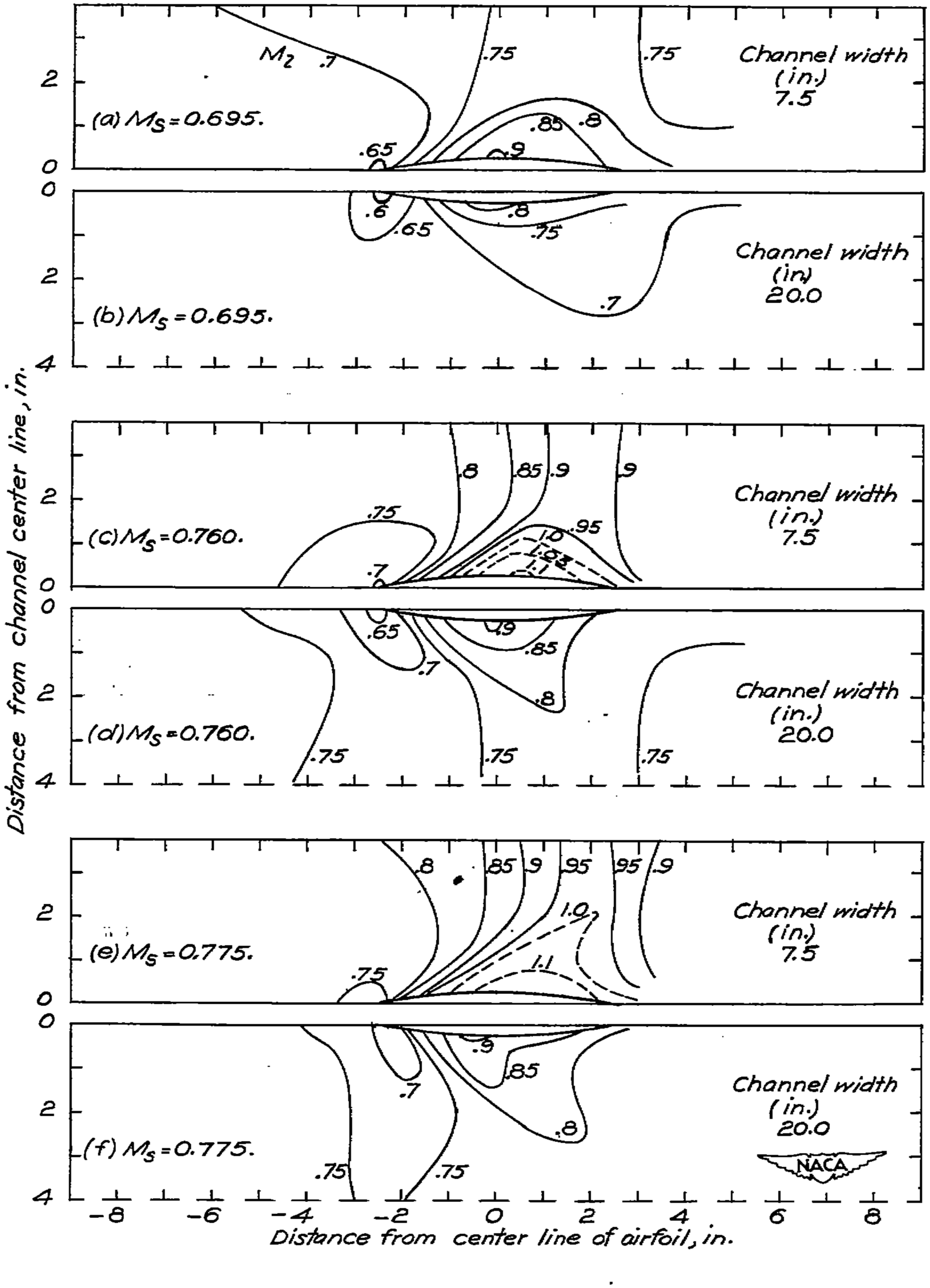


Figure 13.- Comparison of flow fields about a 10-percent-thick, 5-inch-chord, biconvex, circular-arc airfoil in a 7.5-inch and a 20-inch channel with the same indicated Mach number.

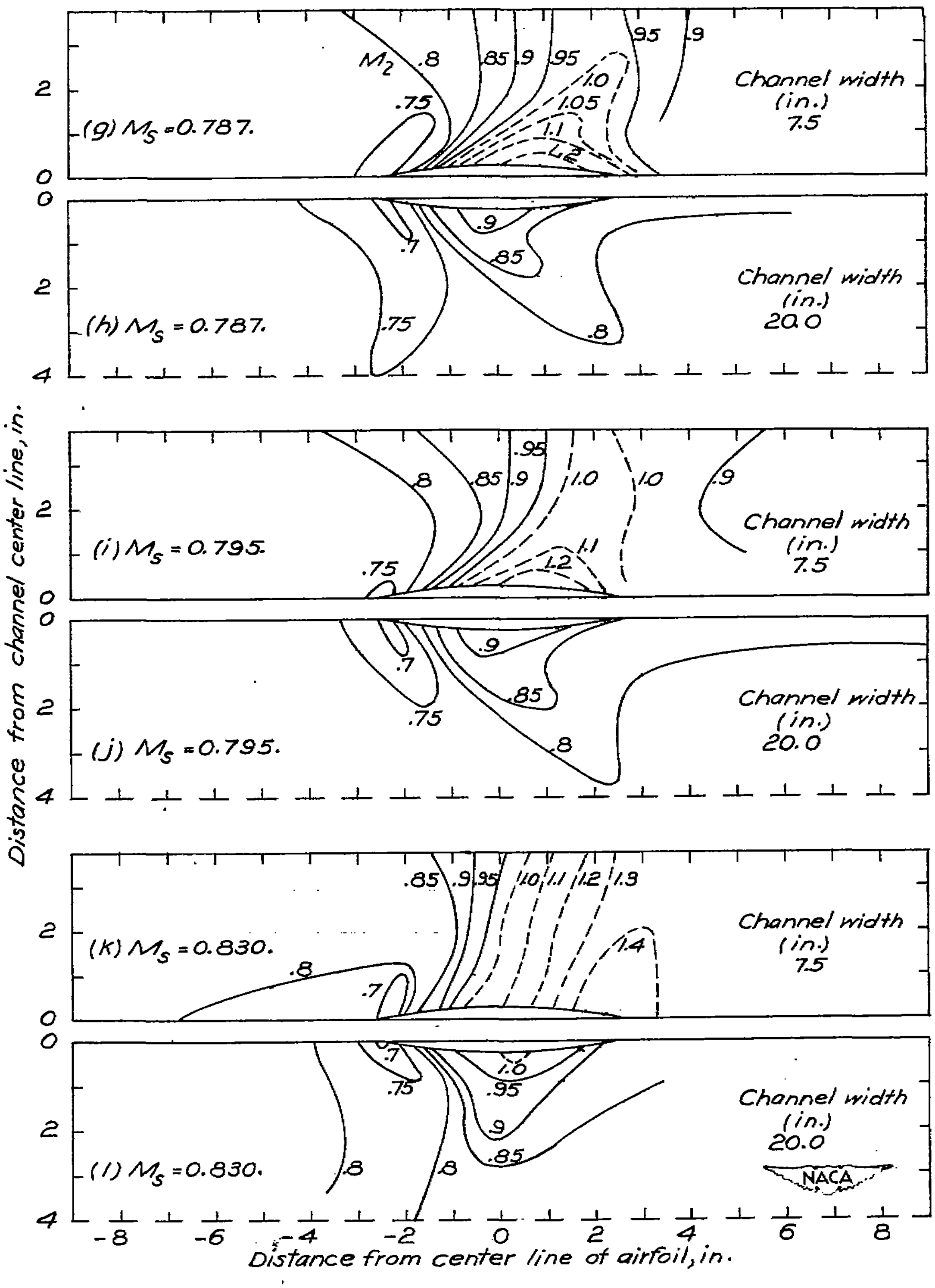


Figure 13.- Concluded.

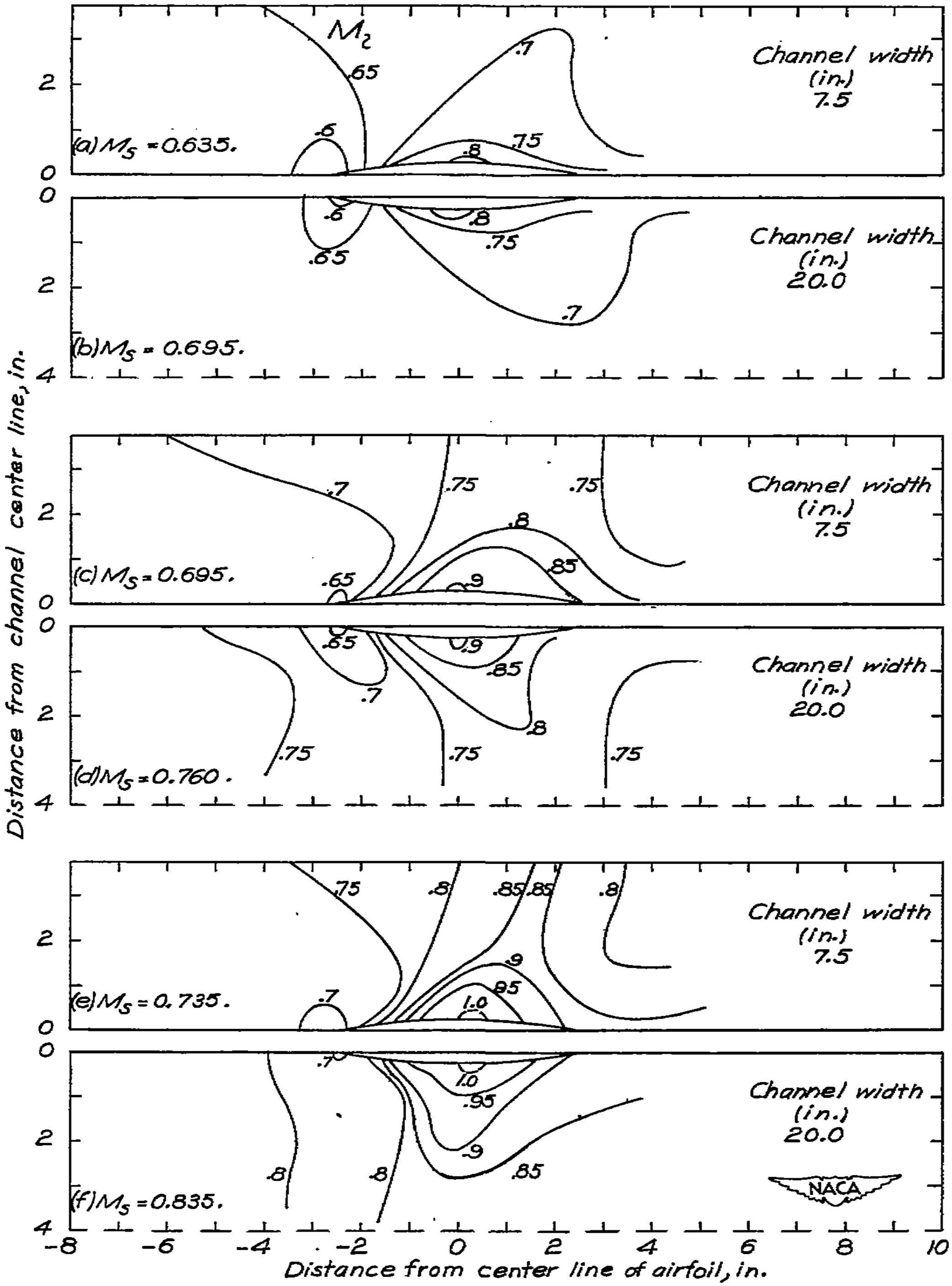


Figure 14.- Comparison with equal maximum Mach numbers of the fields about a 10-percent-thick, 5-inch-chord, biconvex, circular-arc airfoil in a 7.5-inch and a 20-inch channel.

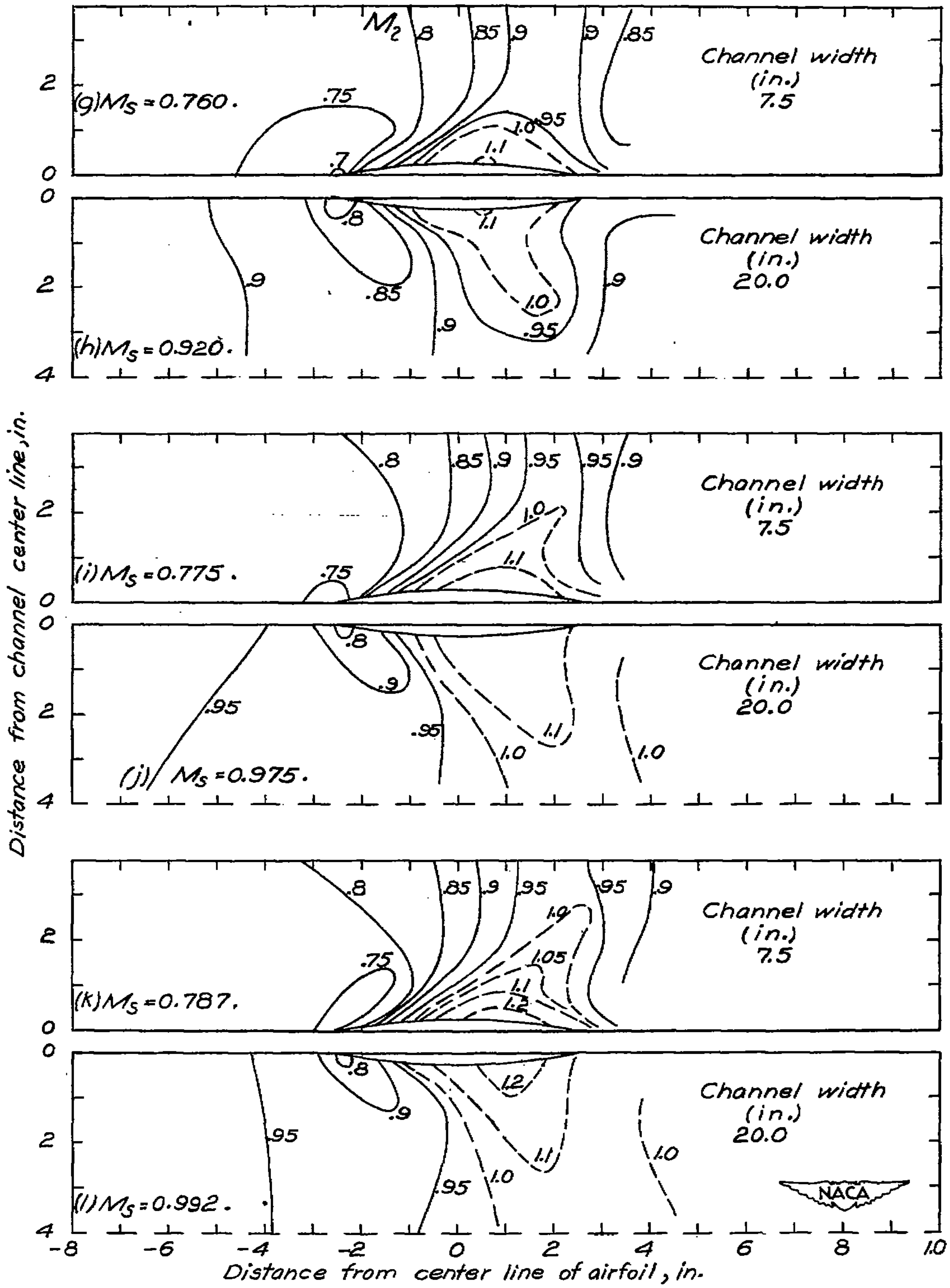


Figure 14.- Continued.

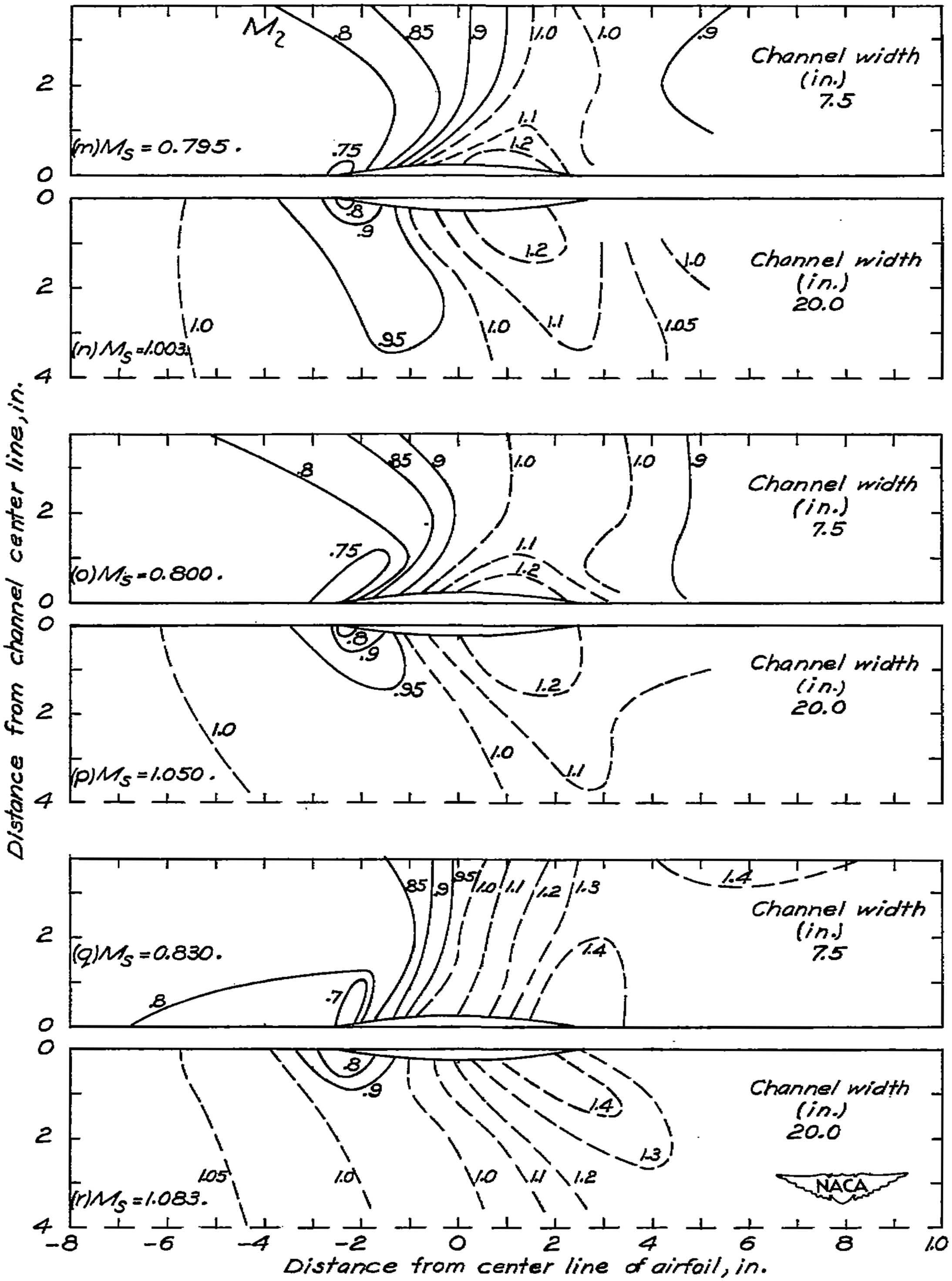


Figure 14.- Concluded.

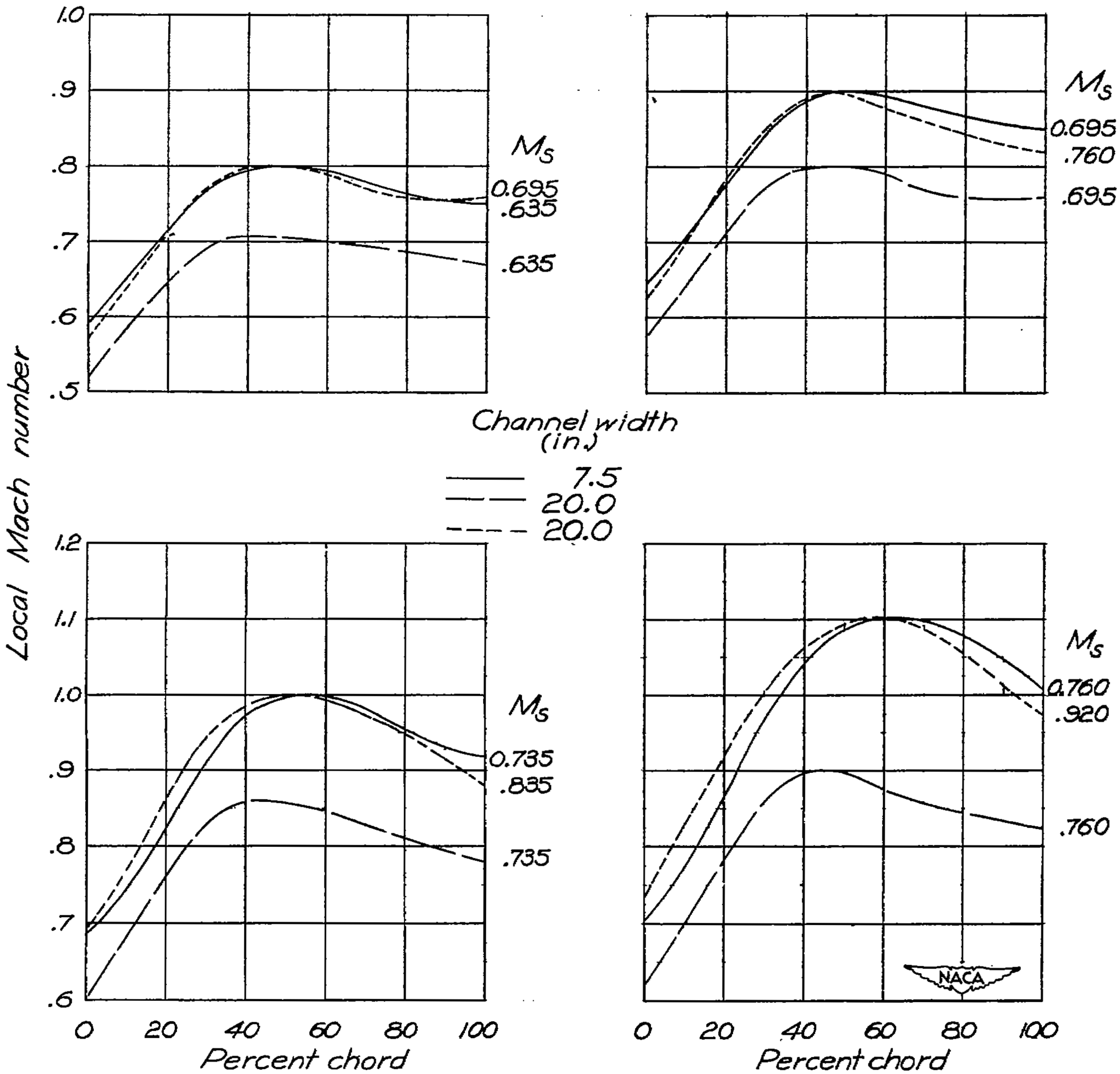


Figure 15.- Comparison of Mach number distributions about a 10-percent-thick, 5-inch-chord, biconvex, circular-arc airfoil in a 7.5-inch and a 20-inch channel.

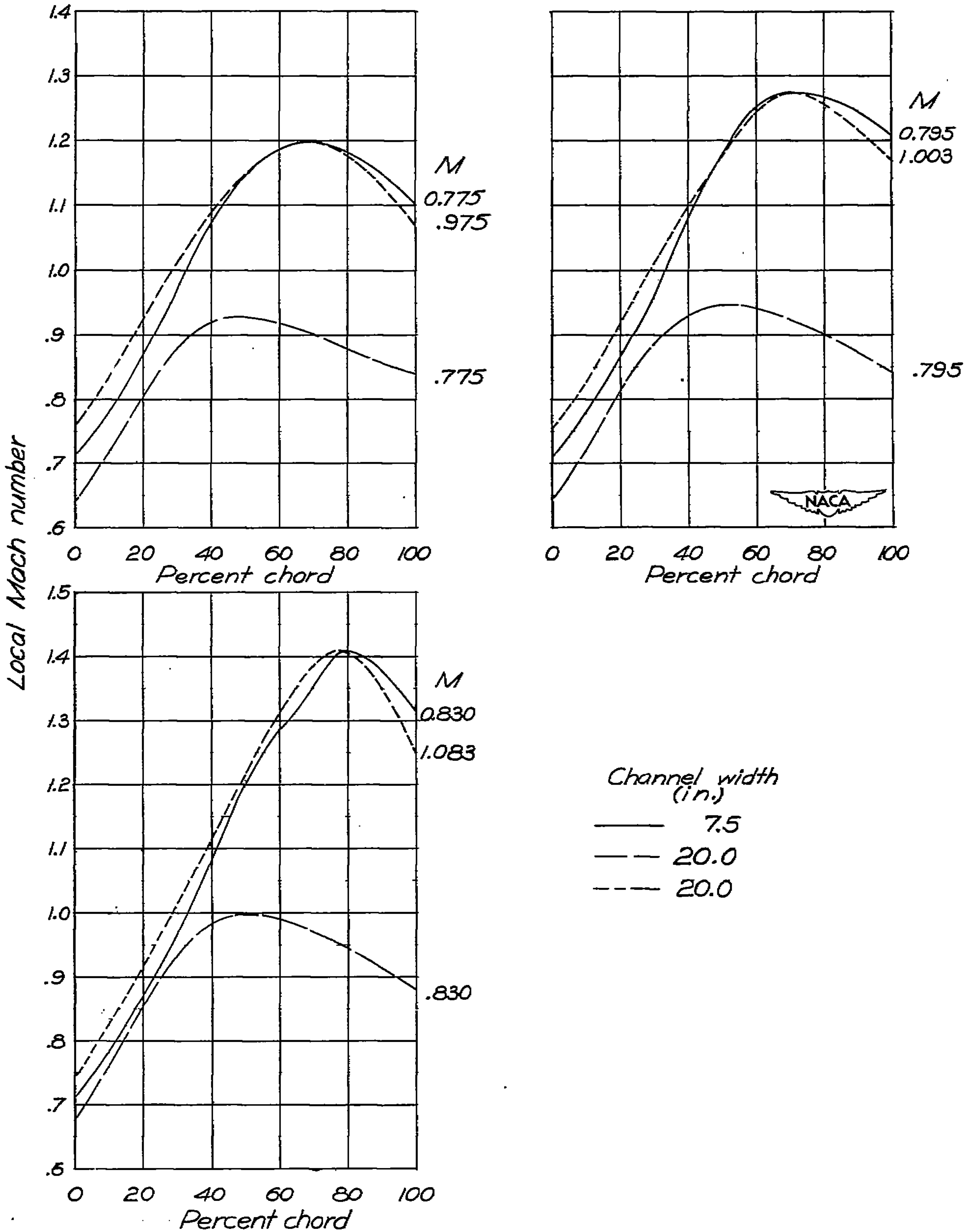


Figure 15.- Concluded.

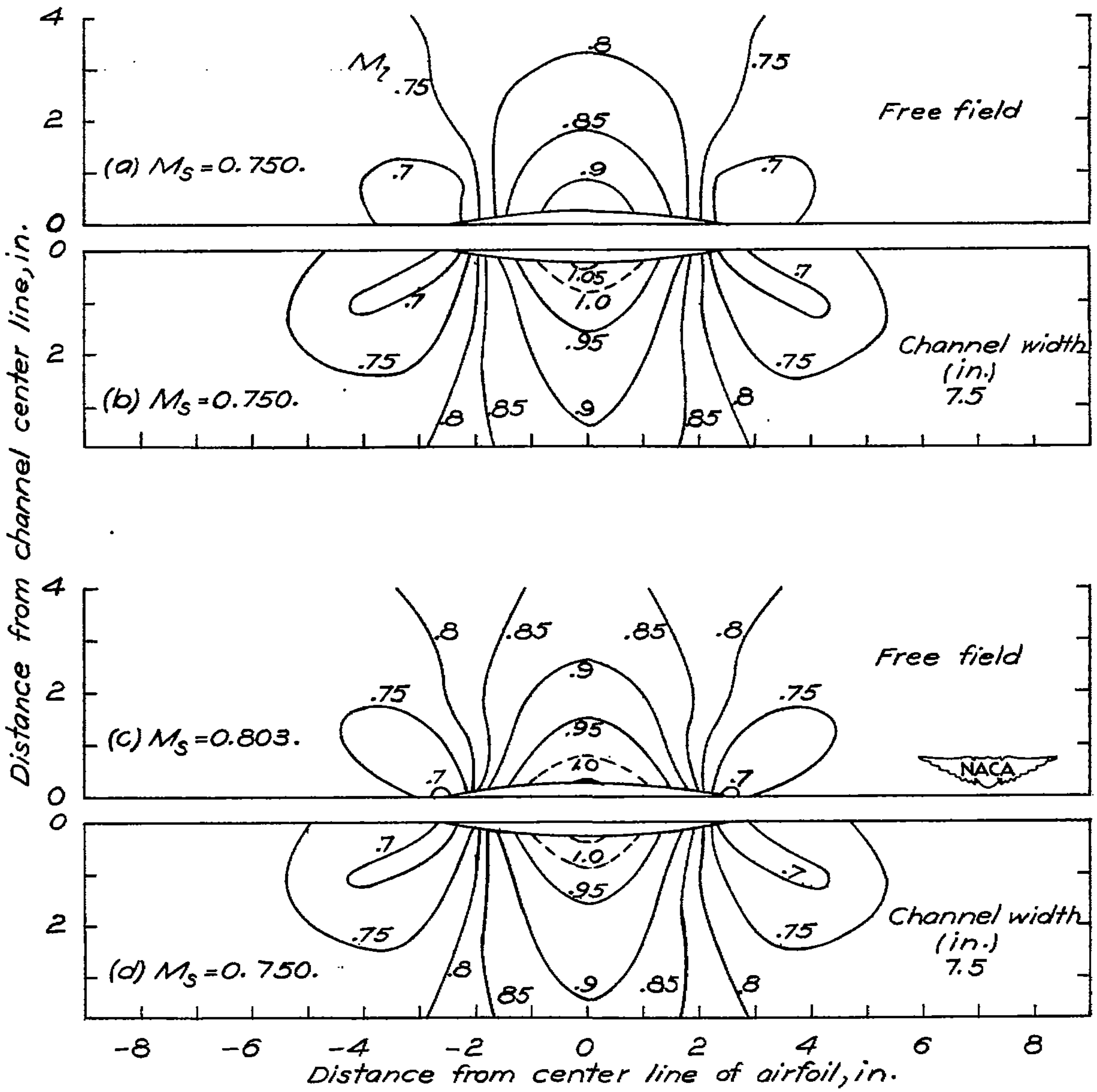


Figure 16.- Theoretical compressible-flow fields about a 10-percent-thick, 5-inch-chord, biconvex, circular-arc airfoil showing effects of solid constriction.

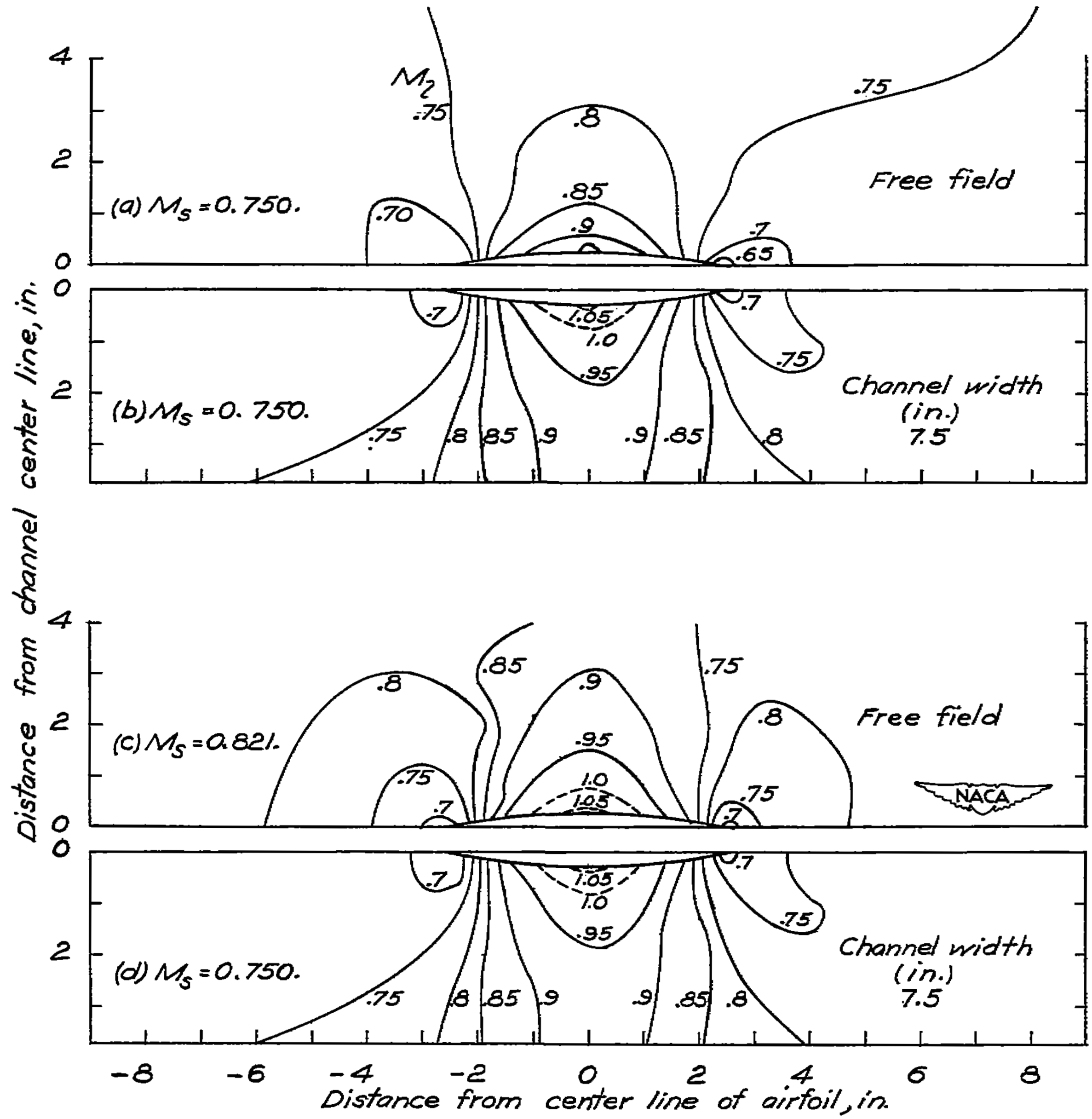


Figure 17.- Theoretical compressible-flow fields about a 10-percent-thick, 5-inch-chord, biconvex, circular-arc airfoil showing effects of solid and wake constriction.

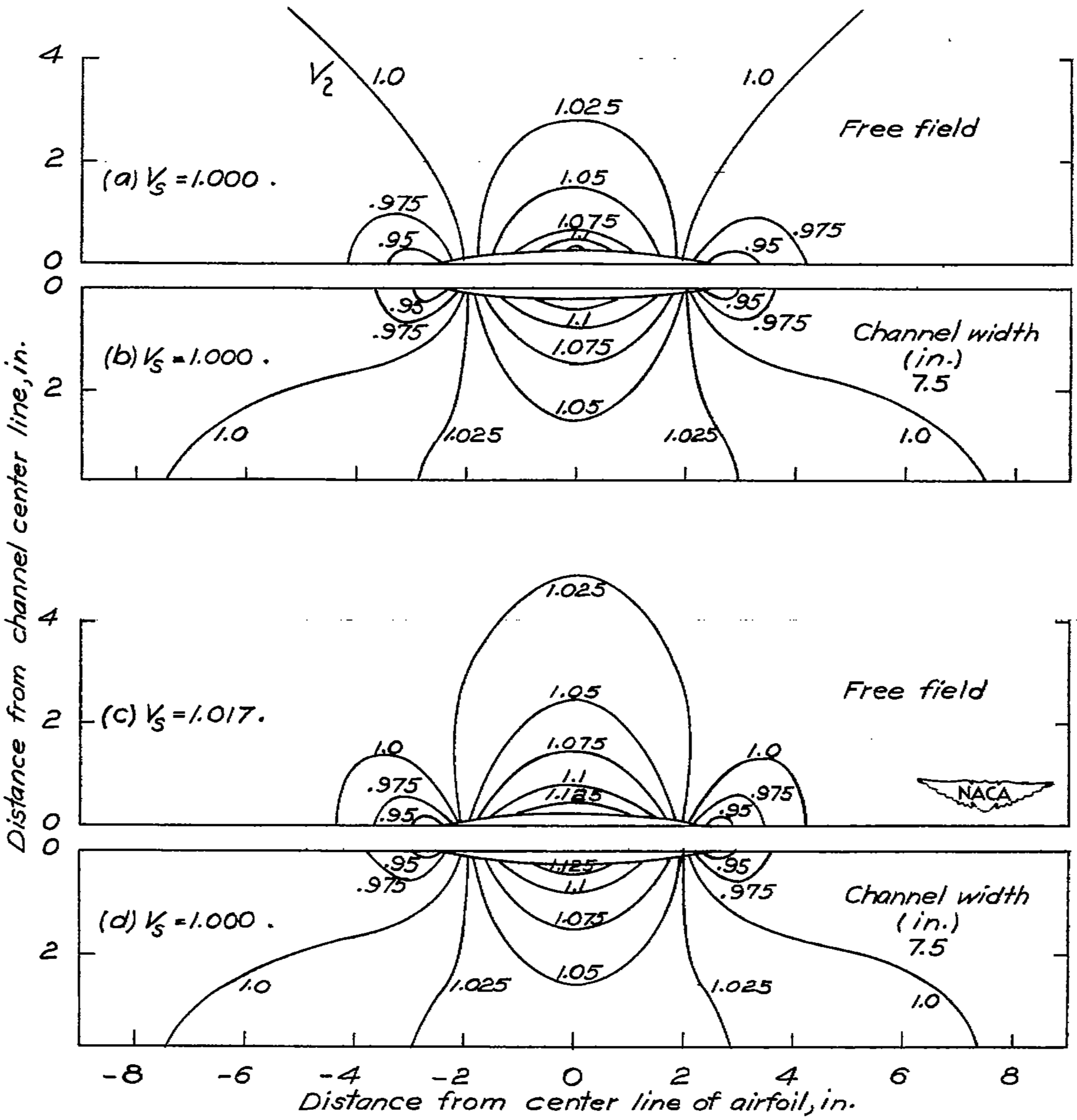


Figure 18.- Theoretical incompressible-flow fields about a 10-percent-thick, 5-inch-chord, biconvex, circular-arc airfoil showing effects of solid constriction.

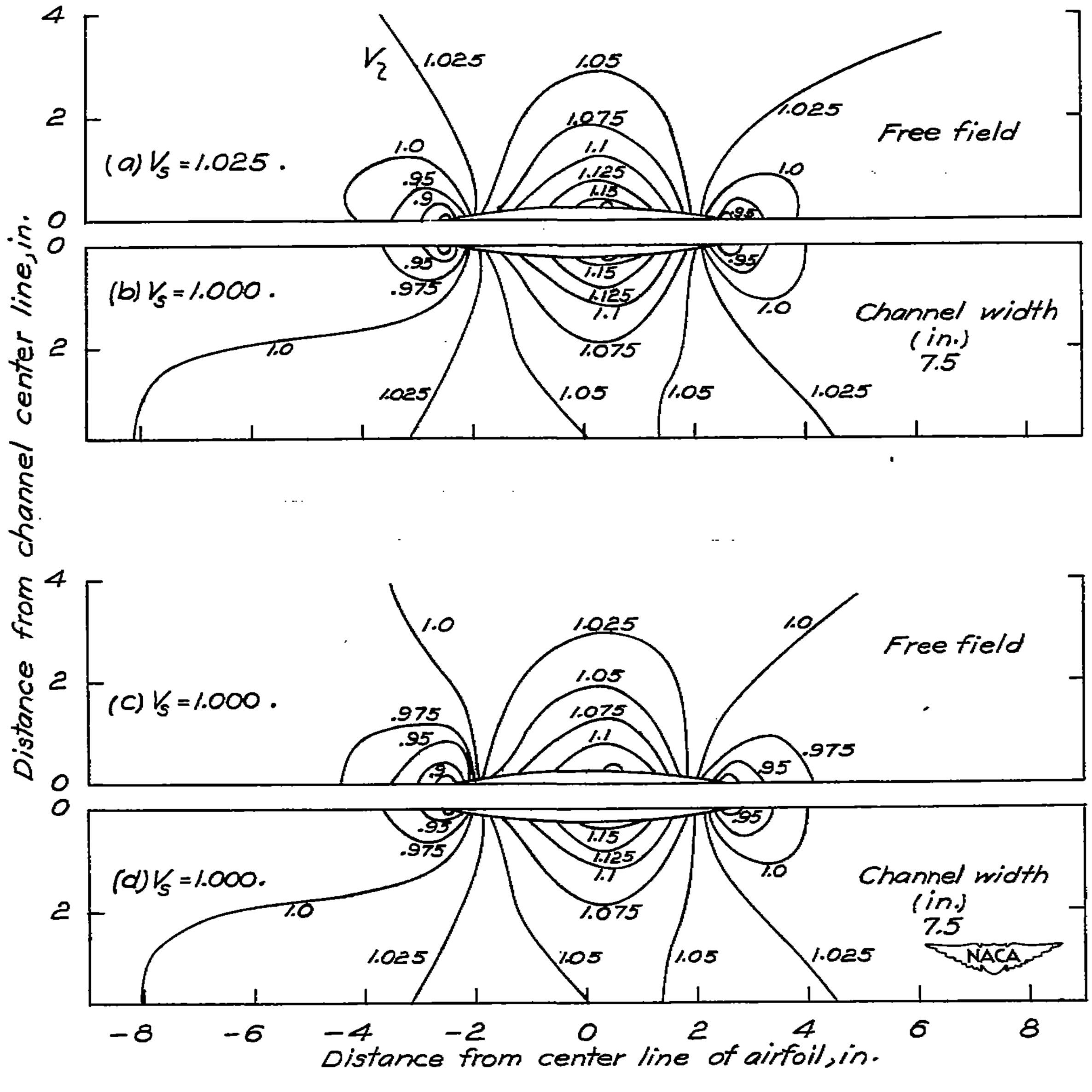
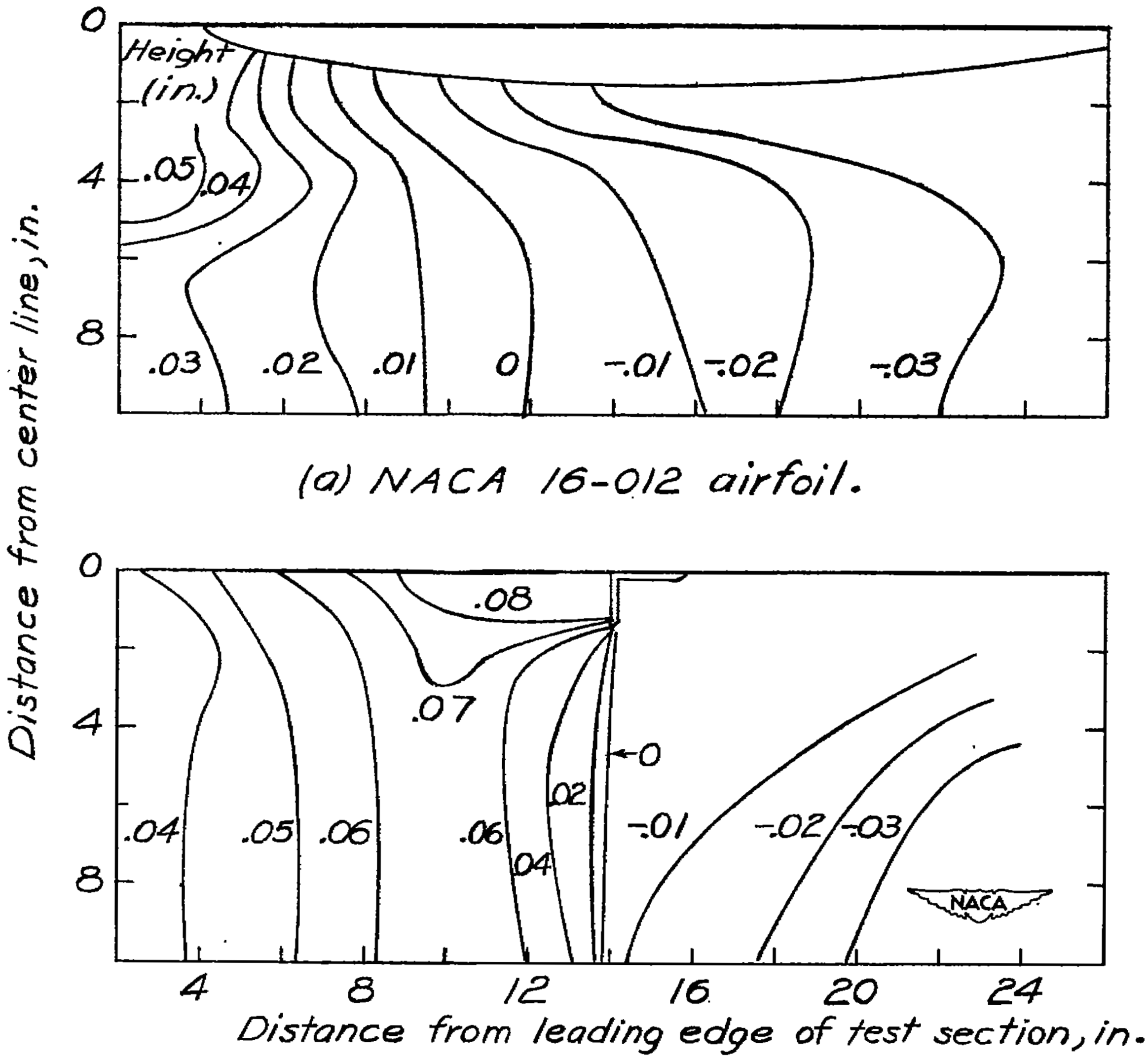
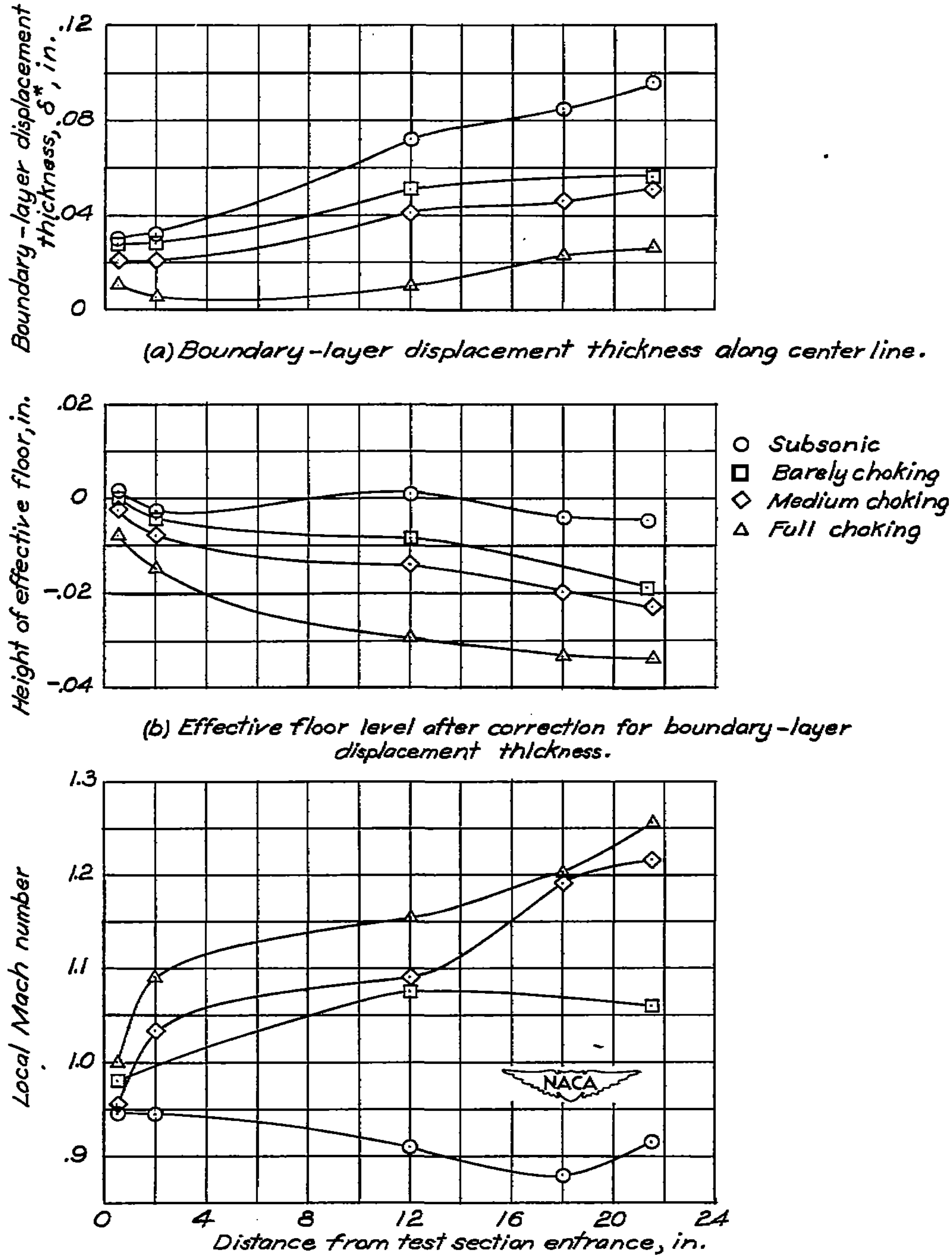


Figure 19.- Theoretical incompressible-flow fields about a 10-percent-thick, 5-inch-chord, biconvex, circular-arc airfoil showing effects of solid and wake constriction.



(b) 2.88-inch-wide flat plate normal to stream.

Figure 20.- Height in inches of displacement floor boundary-layer surface referred to the leading edge of the test section, channel choking.



(c) Local Mach number along center line.

Figure 21.- Boundary-layer displacement thickness changes for various tunnel-empty choking conditions.

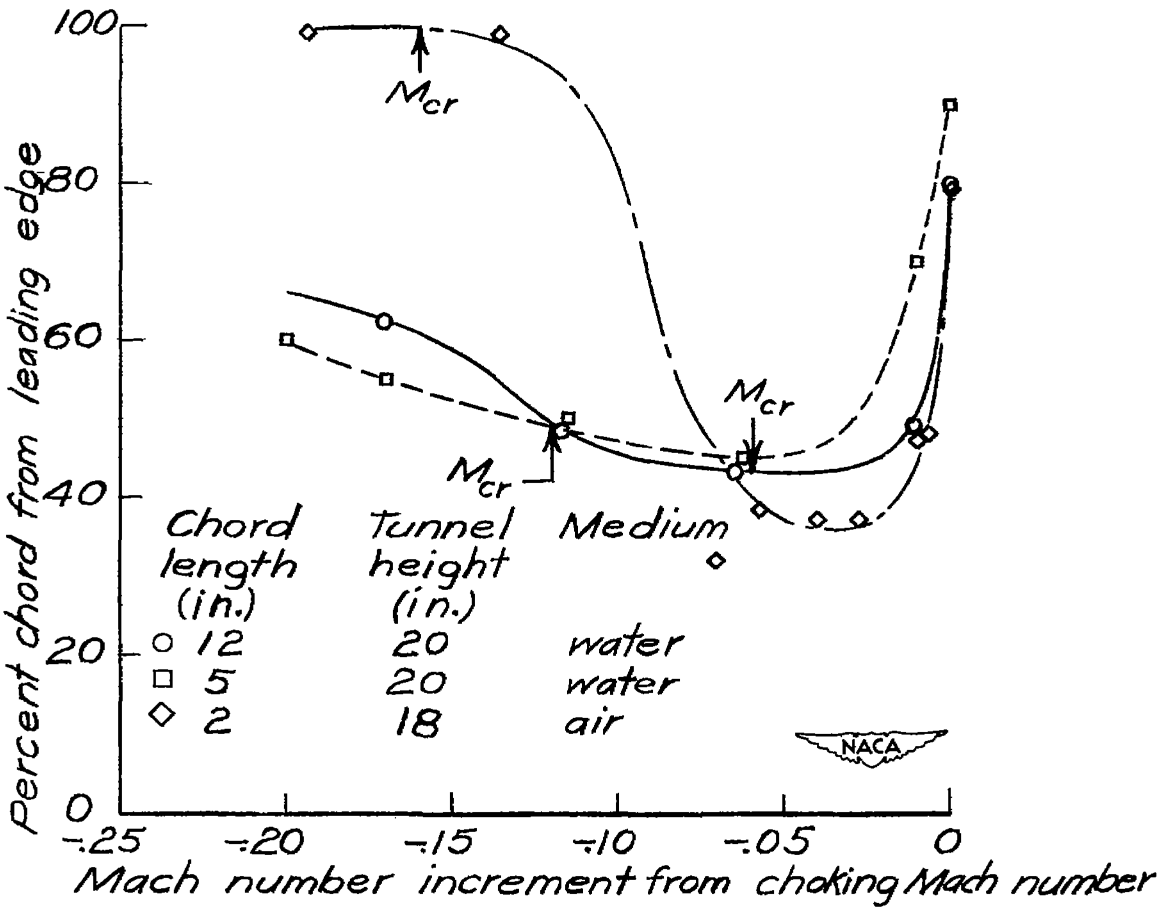


Figure 22.- Comparison of separation point positions on NACA 0012 airfoils in wind tunnels and in the water channel.

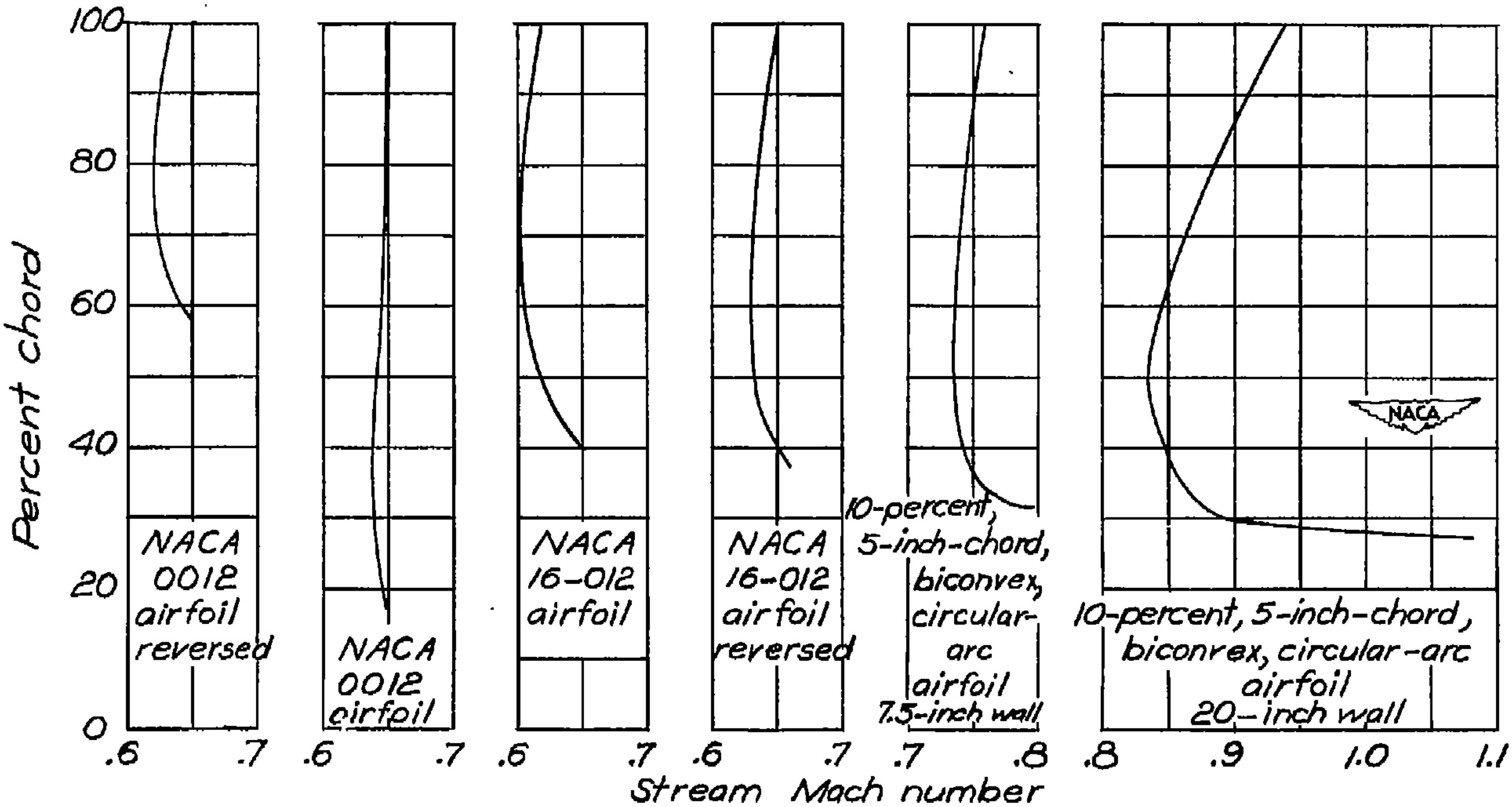


Figure 23.- Location of point of sonic velocity on surface of airfoils tested in the water channel.

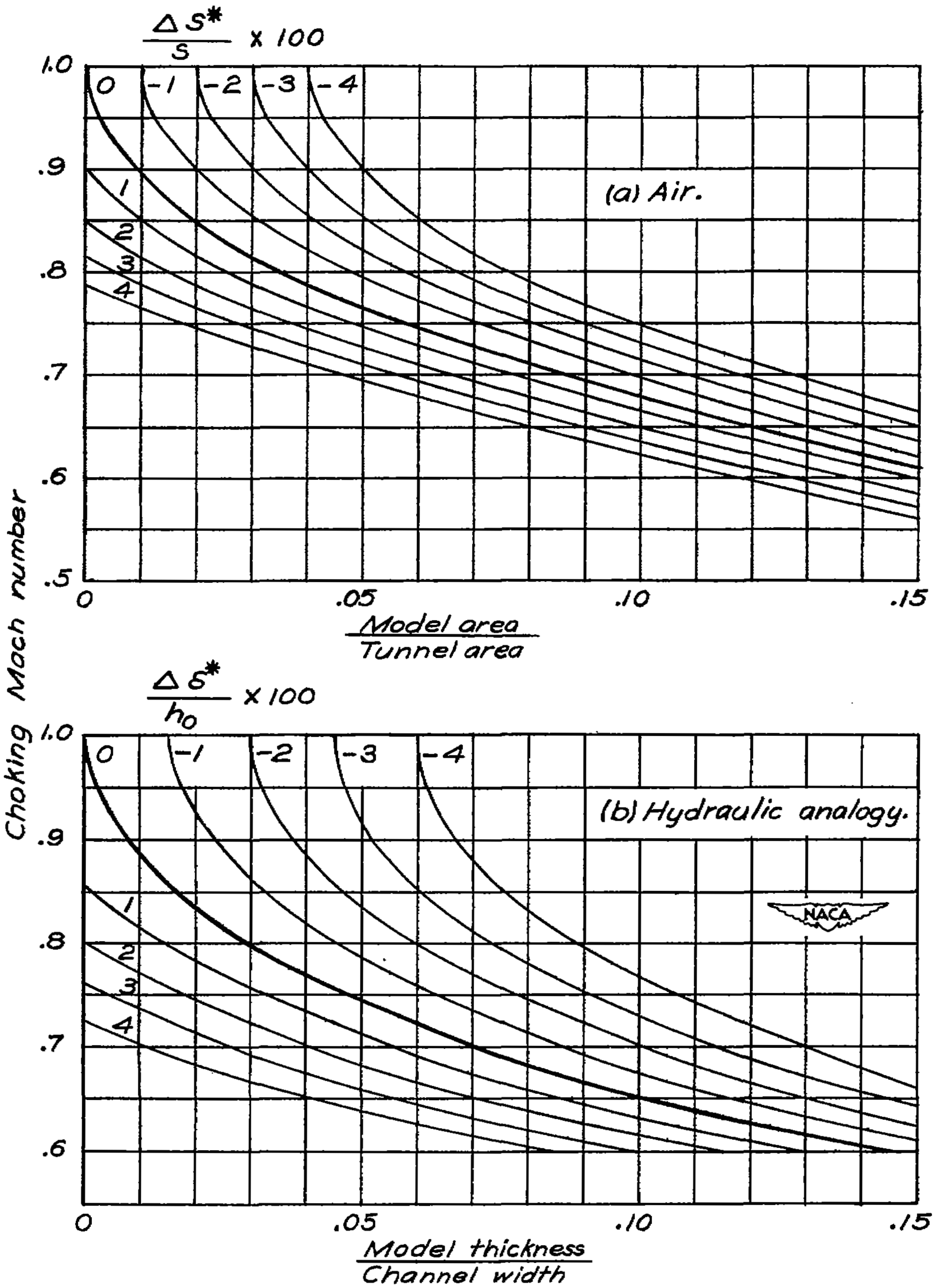


Figure 24.- Choking Mach number as a function of model size showing effect of changes in boundary-layer thickness. One-dimensional theory.

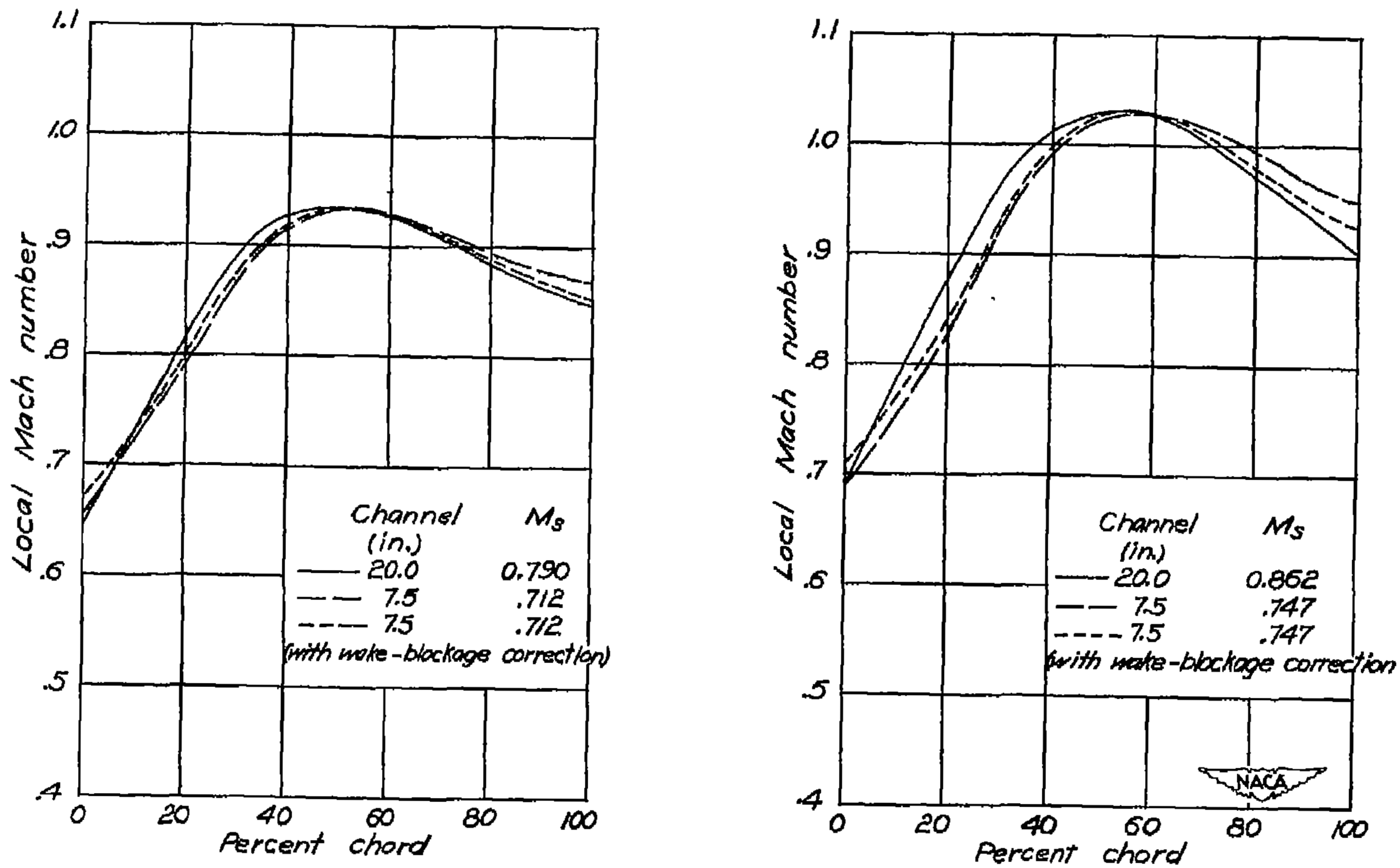
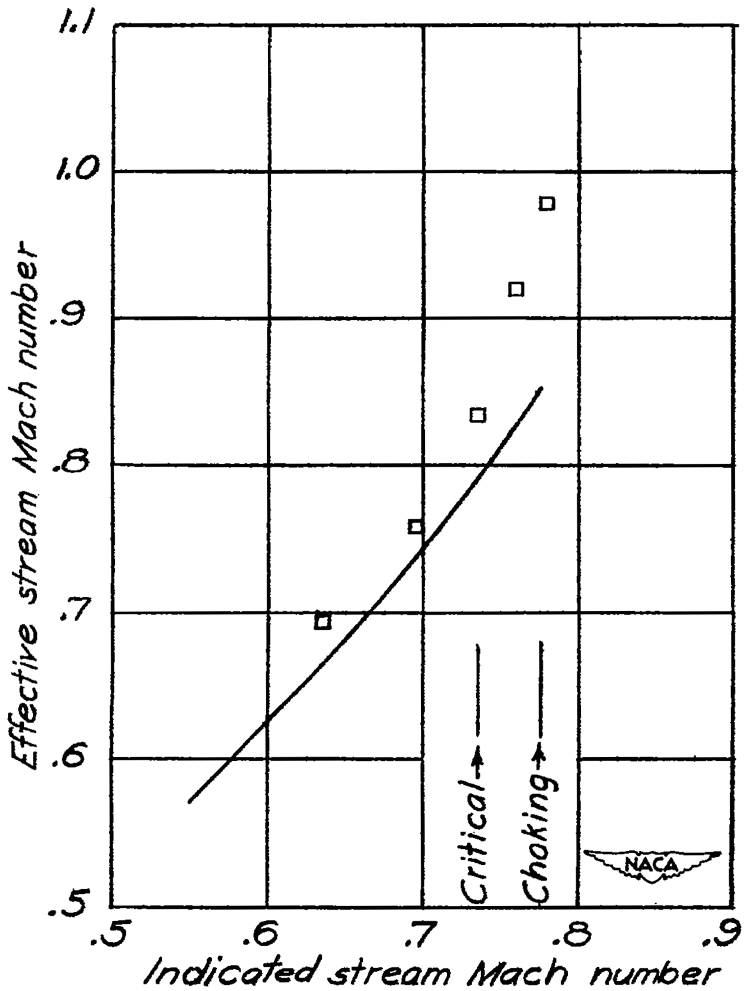


Figure 25.- Effects of wake-blockage corrections on the Mach number distribution about a 10-percent-thick, 5-inch-chord, biconvex, circular-arc airfoil in a 7.5-inch channel.



□ Corrections by matching the maximum Mach numbers
—— Theoretical corrections

Figure 26.- Comparison of the theoretical and experimental effective free-stream Mach numbers of the flow past a 10-percent-thick, 5-inch-chord, biconvex, circular-arc airfoil in a 7.5-inch channel.