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EXPERIMENTAL INVESTIGATION ON THE APPLICATION OF HYBRID LAMINAR FLOW CONTROL TO LARGE-SCALE SWEPT WING MODELS AT SUBSONIC SPEEDS

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Abstract

Carried out in the TsAGI T-107 subsonic wind tunnel (test section diameter d_{wt} = 2.7 m.) has been the experimental investigation of subsonic flow (M = 0.2–0.74; Re = $3.3 \cdot 10^6$–$9.4 \cdot 10^6$) about a large-scale half-model (span 1.67 m, chord 0.73 m) of a swept untapered wing ($\chi = 35°$) equipped with a local boundary-layer suction ($\bar{x}_s = 0.03$–0.11). Based on a measured pressure distributions, angles of attack were selected ($\alpha = -2°$ and $-4°$) to be rational from the standpoint of cross-flow disturbance's development. The extent of the laminar flow zone was determined through various methods. Based on the losses of the total pressure measured in the wake behind the mid-section of the swept portion of the model, the decrease in the section drag was estimated due to local boundary-layer suction. Several laws and suction rates as well as the chordwise extents of the suction region were tested with the aim of obtaining a maximum decrease in the section drag of the swept wing. The computational investigations on the optimal leading-edge shape and suction system parameters on the swept wing section model (χ = 30°) with large chord (b = 3 m) have been accomplished in the conditions of wind tunnel T-107. According to computational evaluations, the extent of laminar flow region may be approximately 50% of the wing chord at the Reynolds number Re ≈ $26 \cdot 10^6$.

1. Introduction

Decreasing drag of today's and future passenger, cargo and business aircraft is a complex and important problem of sub- and transonic aerodynamics. A perspective way of cardinal solution to this problem seems to be local (up to 15–20% of the chord) low intensive boundary-layer suction from the surface of the swept wing having specially designed contours of both the nose portion and the remainder the wing with aim of a significant increase in the laminar flow region's extent. To obtain the maximum gain in decreasing drag, it is advisory to provide local boundary-layer suction on other aircraft components too (for example, on the fin, stabilizer, engine nacelles).

It is known that application of distributed suction provides a significant increase in the laminar flow region both on a flat plate [1] and on airfoils and wings at low subsonic speeds [2,3]. In recent years, for flow laminarization on swept wings (and other components of aircraft) at subsonic speeds, the so-called hybrid laminar flow control (HLFC) has been considered as a promising solution [4–7]. Such a system provides low intensive suction on the nose of the wing (up to 15–20% of the chord) to suppress cross-flow disturbances and uses the effect of natural laminarization [3,8] for weakening longitudinal disturbances.

With the goal of the maximum realization of the HLFC effect, it is necessary to provide rapid acceleration of the flow over the upper surface of the nose portion [8,9], which localizes the rapid growth of cross-flow disturbances along a short region of the swept wing, and subsequent weak acceleration (or practically zero-gradient flow along a significant region of the surface in the case of a high sweep angle) to weaken the growth rate of the longitudinal disturbances (Tollmien-Shlichting waves) at high Reynolds numbers up to full-scale ones (Re ≈ $40 \cdot 10^6$ and greater).

The purpose of the present work is a search for the possibilities of expansion of the laminar flow zone on the wing with a sufficiently high sweep angle under conditions of an acceptable pressure distribution in the neighborhood of the nose region and elevated Reynolds num-

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bers. This is provided by means of local boundary-layer suction with low intensity at various laws of its chordwise distribution at subsonic speeds.

2. Model and methodology of the experiment

The test model represented a half-wing with a span of $l = 1675$ mm (for the test section's diameter being equal to $d_{wt} = 2700$ mm). It was mounted on the side wall of the wind tunnel (Fig. 1). The model consisted of a rectangular portion (span 670 mm) and a basic swept untapered portion ($\chi = 35°$) with a span of 1005 mm and streamwise chord of $b = 732$ mm. These positions were divided by a fence of 1088 mm in length with its maximum height being 560 mm. The swept portion had a symmetric airfoil section (perpendicular to the leading edge). The maximum streamwise thickness ratio was $\bar{c} = 9.8\%$, the thickness ratio perpendicular to the leading edge was $\bar{c}_n = 12\%$. The wing had three chambers for boundary-layer suction placed under the surface of the wing nose portion ($\bar{x}_s = 0.03$–0.11). Over the first chamber there was a stripe of perforations. Over each of other chambers there were two stripes of perforations. Each stripe had 5 rows of holes with a diameter of $d = 0.2$ mm. The distances between rows and the holes in a row were equal to 1 mm. The perforated stripes were spaced $\Delta\bar{x}_s = 0.02$ (2% of the chord) apart. In total there were 5 perforated stripes (25 rows, each having 1000 holes). The number of holes totaled 25000. In the course of the experiments the stripes were opened in a given consecutive order.

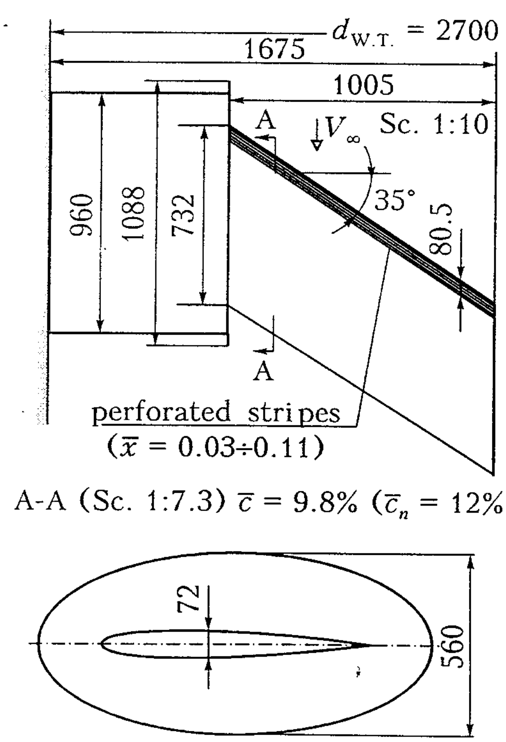

Fig. 1. *Swept wing half-model*

The present experiment was carried out in the TsAGI T-107 large subsonic wind tunnel (test section diameter $d_{wt} = 2700$ mm, length of the unperforated test section $l_{wt} = 3500$ mm). The initial degrees of flow turbulence for the Mach number range M = 0.1–0.8 were 0.1–0.4%. For boundary-layer suction, a self-contained high vacuum suction system was used (to about 200 mm H_2O).

Suction flow coefficient was determined for each of the three chambers using the relationship

$$C_{qi} = \frac{F}{S}\alpha\varepsilon\cdot\sqrt{\frac{\Delta p_i \cdot p_{1i}}{q_\infty p_\infty}},$$

where F – orifice area of the orifice plate ($d_p = 0.00914$ m, $F = 0.000066$ m^2); wing area served by suction ($S = 0.6$ m^2); α – orifice plate coefficient ($\alpha = 0.775$); Δp_i – pressure differential across the orifice plate; p_{1i} – pressure in front of an orifice plate; ε – correction for flow compressibility in a metric range (for the present experiment $\varepsilon = 1$). The suction flow coefficient $C_q = \Sigma C_{qi} = Q/\rho_\infty V_\infty S_s$, where Q is the total airflow through all holes, S_s is the portion of the wing area served by suction, ρ_∞ is the free-stream density, V_∞ is free-stream speed. In this case $S_s = S$.

The experiments with and without boundary-layer suction were performed for the following values and ranges of angles of attack, Mach numbers and Reynolds numbers: $\alpha = 0$; $-2°$; $-4°$; M = 0.2–0.74; Re = $3.3\cdot10^6$–$9.4\cdot10^6$. The tests were accompanied by measurements of the initial degree of flow turbulence (mass flow fluctuations) in presence of the model. The degree of turbulence was $\varepsilon_m = (0.20$–$0.25)\%$ in the flow core at Mach numbers of M = 0.2–0.5.

For determining the laminar-turbulent transition and drag of the swept wing section, various measurement methods were used in the present work. In particular, used in the experiment was a modification of the well-known kaolin-coating technique – the "quick kaolin" method developed at TsAGI [10]. For the same goal, the well-known method of the total pressure on the surface was also used together with comparatively new method of total pressure fluctuations. Measurements through both methods were made simultaneously. Measurements of the total pressure were made using a specially developed miniature probe 0.31 mm in height and 2.5 mm in width, whereas measurements of the total pressure fluctuations were carried out by means of microphone [10].

To determine the drag of the swept portion's mid-section ($\bar{z}$ =0.624), a pitot rake to measure total pressure was installed in the wake behind the mid-section at a distance of $\bar{x} = x/b = 0.88$ from the trailing edge. The rake had 80 probes spaced 2.5 mm ($\Delta\bar{y} = \Delta y/b = 0.0034$) apart and covering the wake's portion $\bar{h} = h/b = 0.27$. The true thickness of the measured wake was considerably less.

3. Analysis of the experimental results

The experimental investigations on the large-scale half-model of a swept wing (l = 1675 mm, b = 732 mm, χ = 35°) were performed both with local boundary-layer suction from the upper surface of the nose portion of the model (for angles of attack ranging from α = 0 to − 4°, Mach numbers from M = 0.2 to 0.74 and Reynolds numbers in the range from $3.3 \cdot 10^6$ to $9.4 \cdot 10^6$) and with no suction.

With the aim of selecting a rational angle of attack for suction experiment, prior to tests, static pressure along the upper surface was measured using a miniature probe at three angles of attack (α = 0; −2°; −4°) and M = 0.2–0.74 with no suction (C_q = 0). As an illustration, Fig. 2 shows pressure distribution at M = 0.3. It can be seen that at α = 0 a positive pressure gradient is observed on the main part of the upper surface, which is prohibitive for the aim of flow laminarization on a swept wing. At negative angles of attack, α = −2° and − 4°, such an adverse phenomenon is not observed. Because of this, further investigations of the effect of boundary-layer suction on the flow about a swept wing were performed at these negative angles of attack.

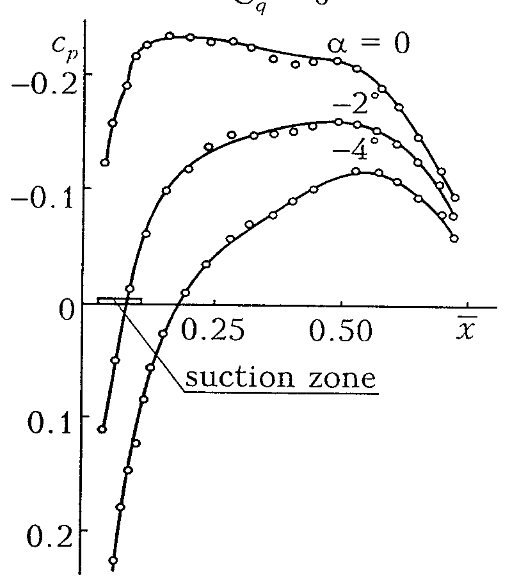

Fig. 2 *Pressure distribution along the upper surface of the middle section of the swept wing for various angles of attack.*

The preferred angle of attack turned out to be α = −2° providing a more quick acceleration of the flow about the upper surface near the nose and lower level of the negative pressure gradient (for example, $dc_p/d\bar{x} = -(0.10\text{–}0.13)$, M = 0.3, compared with the α = −4° case − $dc_p/d\bar{x} = -(0.26\text{–}0.30)$. Such conditions (at α = −2°) are favorable from the standpoint of decreasing the growth of cross-flow disturbances.

3.1 Laminar-turbulent transition. As has been noted above, in present studies the location of laminar-turbulent transition was determined by three methods.

The kaolin-coating method has shown that in the presence of suction with an optimum intensity $C_{q_{opt}} = 12 \cdot 10^{-5}$ the transition line was shifted downstream by more than 40% of the chord ($\bar{x}_{tr}$ = 78%) compared to the case of no suction

$$(C_q = 0,\ \bar{x}_{tr} = 35\%,\ \alpha = -4°,\ M = 0.2).$$

Based on the measurements of the local total (P_0') and static (p) pressure the relationship $\Delta\bar{q} = (P_0' - p)/q_\infty = f(\bar{x})$ was constructed, which is proportional to the local velocity squared in the boundary layer near the surface. It clearly shows not only the end of the transition zone ($\bar{x}_{tr}$) but also its chordwise extent ($\Delta\bar{x}_{tr}$), Fig. 3.

The third method of determination of the laminar-turbulent transition on the wing under study was the method of total pressure fluctuations $\tilde{P}_0' = \sqrt{\tilde{P}_{0i}'^{\,2}}/P_0$ (Fig. 4). A maximum of these fluctuations corresponded to the transition ($\bar{x}_{tr}$) and was equal to about 0.2 to 0.75% of the flow's total pressure for a Mach number range of M = 0.2–0.6 (Re = $3.3\cdot10^6$–$8.5\cdot10^6$).

A comparison of three methods of detecting the laminar-turbulent transition (at $C_q = 0$) reveals their satisfactory agreement (Fig. 5). In the optimum low-intensity suction regime ($C_q = C_{q_{opt}}$), the laminar zone in the mid-section ($\bar{z} = 0.5$) of the wing under study ($\chi = 35°$) becomes more than twice as large as in the no-suction case (Fig. 5).

3.2 Drag. Increasing the laminar flow region of a swept wing must be accompanied by decreasing drag. To determine the drag of the mid-section of the wing's swept portion ($\bar{z} = 0.624$)

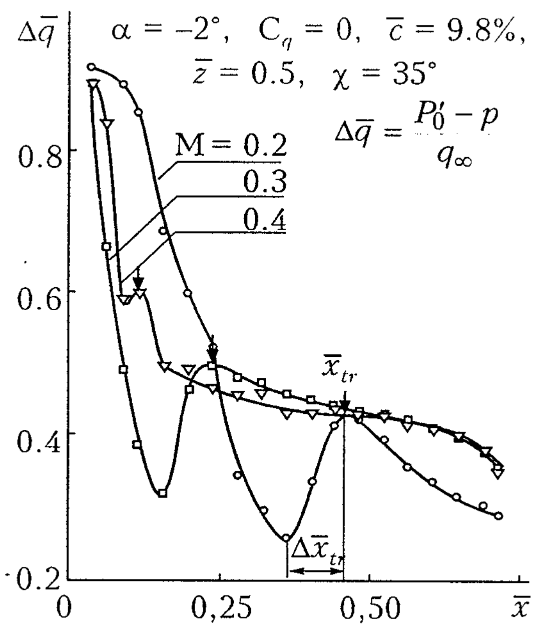

Fig. 3 *Transition identification by the total-pressure method*

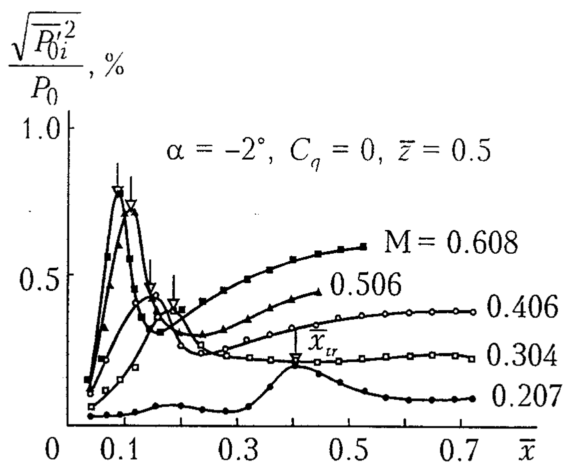

Fig. 4 *Relative total pressure fluctuations for various Mach numbers (with no suction)*

the losses of total pressure $\overline{H} = (P_0 - P_{0i})/q_\infty$ were measured at a distance of about the chord length ($\bar{x} = 0.88$) using the pitot rake. As evident from Fig. 6, the losses of total pressure in the wake decrease with increasing the suction flow coefficient C_q to $C_{q_{opt}} = 11\cdot10^{-5}$ ($\alpha = -2°$, M = 0.3, perforation coefficient $\sigma = F_{SUC} / S = 0.125\%$). Pressure losses increase with further increasing suction flow coefficient (for example, up to $C_q = 13.4\cdot10^{-5}$).

Drag of the mid-section of the swept wing was determined by the momentum method based on total pressure losses in the wake. To investigate quantitatively the effect of boundary layer suction on drag reduction, the dimensionless parameter $\bar{c}'_{xp} = c'_{x_s} / c'_{x_{C_q=0}}$ was introduced.

Studied in the present work was influence of various suction laws (from uniform to non-uniform), suction intensity and suction region extent (at various values of the perforation coefficients $\sigma = F_{SUC} / S$) on change in the sectional drag coefficient $\bar{c}'_{xp}$. The maximum positive effect in this experiment was obtained with the maximum value of the perforation coefficient ($\sigma = 0.125\%$, $\Delta\bar{x}_s = 0.08$) and the suction law close to the uniform one. By the way of illustration Fig. 7 shows the functions $\bar{c}'_{xp}(C_q)$ for M = 0.3–0.6 and Re = (4.8–8.5)$\cdot10^6$ at $\alpha = -2°$. Here we notice that the maximum sectional drag reduction (with $C_q = C_{q_{opt}}$) for the swept wing is as much as 14% (M = 0.3, Re = $4.8\cdot10^6$). The positive suction effect gradually decreases with simultaneous increase in the Mach and Reynolds numbers. The main reason for this appears likely to be an increase in the Reynolds number (from $4.8\cdot10^6$ to $8.5\cdot10^6$) rather than the compressibility effect. It should be noted that the maximum drag reduction occurs at low suction intensity. For example,

$C_{q_{opt}} = (11\text{-}4)\ 10^{-5}$ at $\alpha = -2°$ and M = 0.3–0.6. It is remarkable that the function $\bar{c}'_{xp}(C_q)$ has a minimum (at $C_q = C_{q_{opt}}$) followed (at $C_q > C_{q_{opt}}$) by decreasing suction efficiency (reduction of the gain in drag). Most likely in these experimental conditions the suction efficiency losses at $C_q > C_{q_{opt}}$ are caused by the allowable (critical) hole Reynolds number $Re_d = V_h d / \nu$ (where V_h is the flow velocity in the hole, d is the hole diameter, ν is the kinematic viscosity coefficient) being exceeded and subsequent irreversible flow turbulization.

Proposed recently in Ref.11 was a new suction parameter characterizing the onset of turbulization of the flow in the suction hole:

$$P = \frac{V_h d^2}{u_e \delta^{*2}} \geq 0.2, \text{ or } P = \frac{Re_d \bar{d}}{Re \cdot \bar{u}_e \bar{\delta}^{*2}}, \text{ where}$$

$\bar{V}_h = V_h / V_\infty$, $\bar{d} = d/b$, $\bar{\delta}^* = \delta^*/b$ and $\bar{u}_e = u_e / V_\infty$ is the relative velocity at the outer edge of the boundary layer. To put it another way, the critical value of the suction parameter P (P_{cr}) is proportional to the critical value of Re_d (Re_d^*).

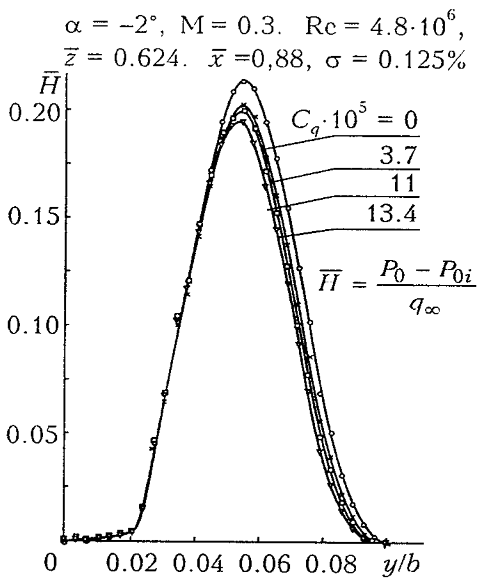

Fig. 6 *Total pressure losses in the wake behind the wing.*

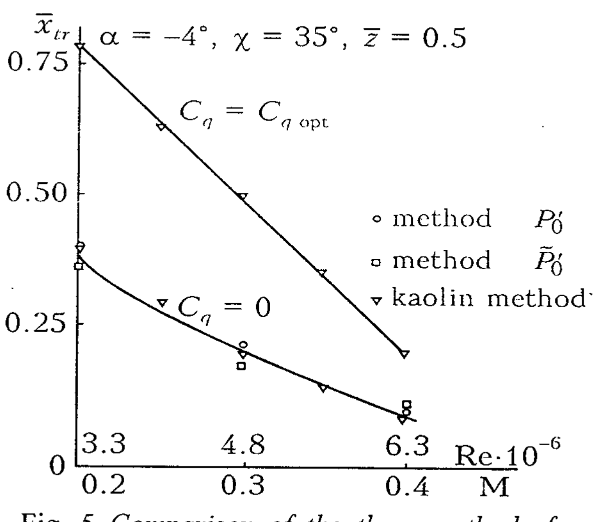

Fig. 5 *Comparison of the three methods for detecting laminar-turbulent transition on the upper surface of the swept wing.*

In our experiments it was found that an increase in Re from $4.8 \cdot 10^6$ ($M = 0.3$) to $8.5 \cdot 10^6$ ($M = 0.6$) leads to decreasing the mean value of Re_d^* from about 160 (perforation coefficient $\sigma = 0.075\%$, three perforation stripes, $N = 3 \times 5$) to about 80 ($\sigma = 0.125\%$, five perforation stripes, $N = 5 \times 5$) at $\alpha = -2°$.

From Fig. 8 it follows that the maximum drag reduction ($\Delta \bar{c}'_{xp}$) within $M = 0.3$–0.4 at $\alpha = -2°$ is more than twice that at $\alpha = -4°$. This results from both the greater flow acceleration at the forward upper surface and more than two times lesser negative pressure gradient over the main portion of the upper surface (Fig.2).

4. Computational investigations on the optimal leading-edge shape and suction system parameters on the swept wing section model ($\chi = 30°$)

In the second stage of the investigations aimed at reducing friction drag by using HLFC system, it is planed to carry out a set of experiments in the T-107 wind tunnel with the large-scale resized model of the constant chord section of the swept wing having a sweep angle of $\chi = 30°$ and a chord length b=3m. Boundary-layer suction will be applied to a 20% - chord forward portion of the model. To determine the leading edge shape and suction system parameters optimized from the laminarization standpoint for the T-107 environment, investigations were carried out with the use of various computational methods. Parametric calculations of the inviscid flow about the model were performed to modify the contours of the forward portion of the model. Fig. 9 shows the contours of the initial (0) and two modified (1 and 2) airfoil sections. The pressure distributions in the nose region over the upper surface in the 50% semispan wing section ($\bar{z} = 0.5$) were computed by the panel method with taking into consideration the tunnel wall interference effect, refer to Fig. 10*).

*) The flow about the model of the swept wing section in the T-107 wind-tunnel test section was calculated by Yu.L.Zhilin, L.G.Ivanteeva and L.L.Teperin.

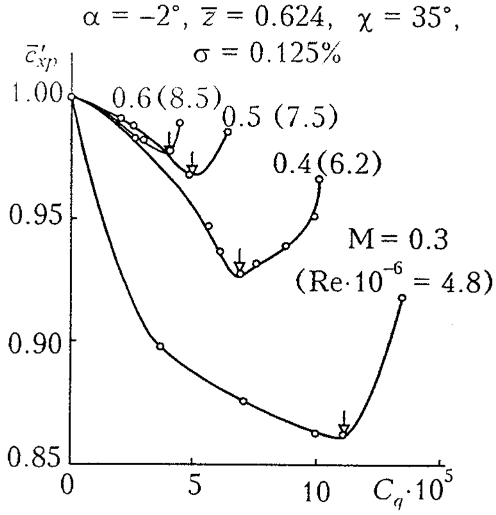

Fig. 7 *Influence of suction flow rate on the variation of the drag coefficient of the swept wing middle section for various Mach and Reynolds numbers.*

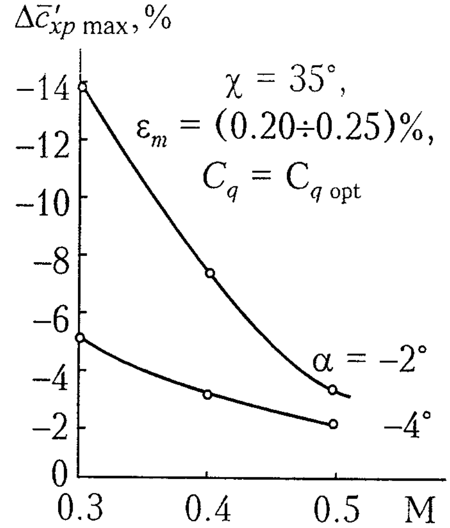

Fig. 8 *Maximum drag reduction for the middle wing section as a function of Mach number at optimum suction.*

Fig. 11 shows (for example) the computed constant pressure coefficient c_p counters lines over the upper surface of modified airfoil wing section (1). The constant pressure lines are nearly parallel to the wing leading edge line only within its close vicinity where the yawed-wing theory is applicable. Longitudinal and transverse pressure gradients at the main portion of the surface are of approximately the same magnitude, the near-wall flow is substantially three-dimensional. As it follows from the pressure distribution shown in Fig. 11, for the medium portion of the half-wing ($0.3 \le \bar{z} \le 0.7$) the yawed-wing theory can be taken as a primary approximation. This being so, the method described in [12] for numerical solution to the equations governing the flow in the boundary layer on the yawed wing was used to calculate boundary layer characteristics in this region.

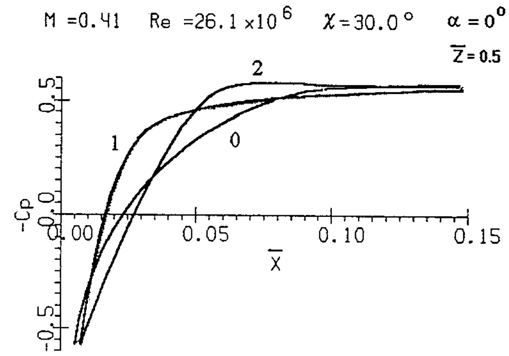

Fig. 10 *Pressure distributions in the nose region over the upper surface in the 50% semispan wing section with initial* (0) *and modified airfoils* (1) *and* (2).

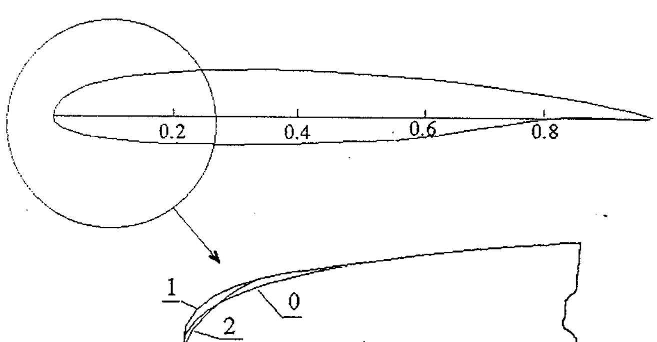

Fig. 9 *Airfoil nose shape modification:* 0 - *initial airfoil;* 1 *and* 2 - *modified airfoils.*

The effectiveness of boundary-layer suction was assessed on the basis of parametric calculations in terms of flow stability characteristics and criteria for the laminar-turbulent transition with the use of semi-empirical approaches reported in [1], [13], [14]. The results of this assessments are presented in Fig. 12. In the figure the calculated transition location $\bar{x}_{tr}$ is represented in terms of the total suction airflow coefficient C_q for various laws of the distributed suction airflow coefficient $c_q = \rho v_w / \rho_\infty V_\infty$ (where v_w is the distributed suction flow velocity) shown diagrammatically. The location $\bar{x}_{tr}$ of the transition was determined as the least

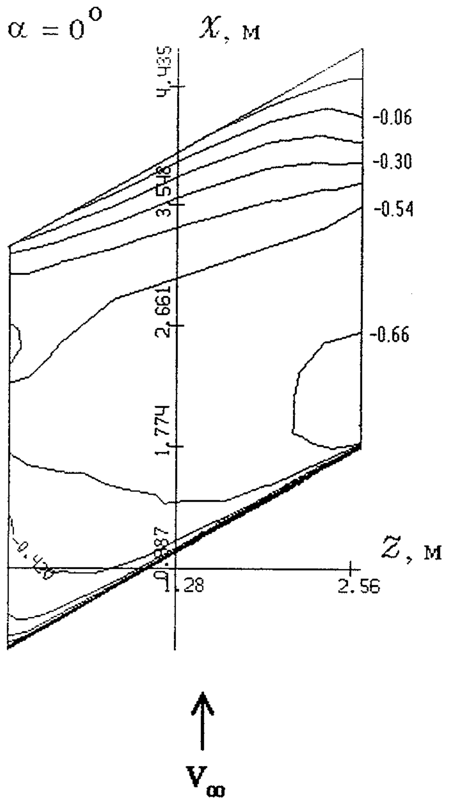

Fig. 11 *Computed constant pressure counters lines over the upper surface of modified airfoil wing section* (1) *in the wind tunnel test section coordinate system* (X , Z).

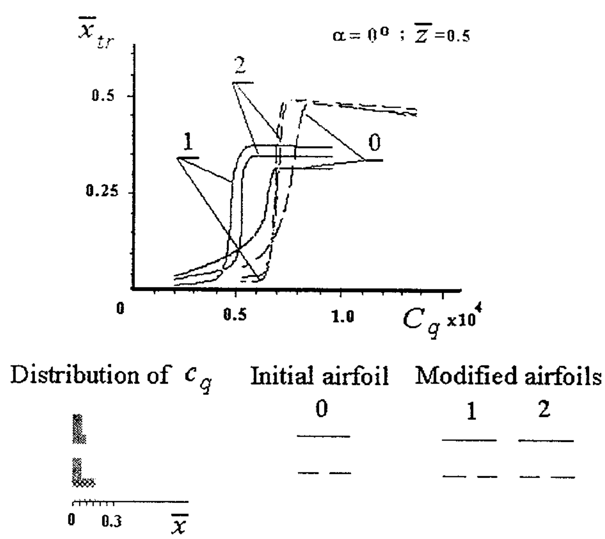

Fig. 12 *Calculated transition location* $\bar{x}_{tr}$ *in terms of the total suction airflow coefficient* C_q *for various laws of the distributed suction coefficient* C_q.

value among the three criteria from [1], [13], [14].

As can be seen, the modification of the airfoil's geometry (1) yields some advantages in flow laminarization using local air suction in the nose region both over the original geometry and modification (2), and it should be recommended for testing.

The most favorable suction intensity distribution for the recommended airfoil geometry is a two-level distribution with increased suction rate in the nose area ($\bar{x} \le 0.05$).

In spite of the fact that transverse pressure gradients in the medium portion of the wing are small, the application of yawing wing theory to computing velocities at the external edge of the boundary layer using the three-dimensional pressure field can lead to errors in assessing cross-flow stability based on approximate criteria from [13], [14] outside the neighborhood of the leading edge. For taking into account the effects of three dimensionality in the main order of magnitude, a method based on small perturbations theory was used for boundary layer computation [15].

Besides, for checking the validity of the approximate transition criteria, computations were performed of the boundary-layer's stability characteristics, basing on the methods of linear stability theory from [16].

Comparison of stability and the growth increment of cross-flow disturbances together with numerical analysis in the framework of small perturbations theory [15] performed for a number of check cases, substantiate estimates obtained from parametric computations.

Conclusions

1. Various laws, intensities and chordwise extents (variable perforation coefficient) of boundary-layer suction were studied. The most efficient for the tested wing ($\chi = 35°$) turned out to be the uniform (or close to uniform) suction with maximum chordwise extent ($\sigma = 0.125\%$)

2. By using low-intensity, local boundary-layer suction, maximum sectional drag reduction was obtained equal 14% at $\alpha = -2°$, $M = 0.3$, $Re = 4.8 \cdot 10^6$.

3. The computational investigations on the optimal leading-edge shape and suction system parameters on the swept wing section model ($\chi = 30°$) with large chord

(b = 3 m) have been accomplished in the conditions of wind tunnel T-107. According to computational evaluations, the extent of laminar flow region may be approximately up to 50% of the wing chord at the Reynolds number Re $\approx 26 \cdot 10^6$.

4. It is considered advisable to further proceed with searching studies on the extension of laminar flow zones both on swept airfoil section and on real-word swept wings for advanced passenger aircraft at full-scale Reynolds number.

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