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Effect of Small Angles of Attack on Turbulence Generation in Supersonic Boundary Layers on Swept Wings

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Abstract—We present the new (for Mach numbers M = 3 and 3.5) and generalizing (for Mach numbers from 2 to 4) results of experimental investigations on the effect of small angles of attack on laminar-turbulent transition in the supersonic boundary layer on a swept wing with the leading-edge slip angle of 72°. The angle-of-attack variation has a strong effect on the transition Reynolds number. The transition Reynolds number decreases with increase in the Mach number. The measurements were carried out by means of a constant-temperature hot-wire anemometer using the proven procedure of determining the transition location. The e^N method is used for the first time for numerically estimating the transition Reynolds numbers in the supersonic boundary layer on a swept wing with the leading-edge slip angle of 72°. The growth of the amplitudes of the steady and unsteady modes of the boundary layer crossflow are calculated in accordance with the linear stability theory, within the framework of the Lees–Lin system of equations. The numerical results indicate that, in accordance with the experimental results, laminar-turbulent transition in the boundary layer on the model swept wing is governed by the growth of stationary modes of the crossflow instability.

Keywords: experiments, supersonic boundary layer, swept wing, angle of attack, angle of slip, laminar-turbulent transition, transition Reynolds number, linear stability theory

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The investigations of turbulence origin and the development of new engineering approaches to the prediction and control of laminar-turbulent transition in three-dimensional boundary layers on the basis of the data obtained is at the center of attention of the specialists in many countries. These studies are of practical interest, since these flows are realized on swept wings of the aircraft. The scenarios of laminar-turbulent transition on swept wings are presented in [1–4]. We note that the process of turbulence generation in swept-wing boundary layers is qualitatively different from the two-dimensional case on a flat plate, where the transition is caused by the Tollmien–Schlichting (TS) wave instability. In the three-dimensional boundary layer on a swept wing, apart from the TS waves slightly growing due to the favorable pressure gradient, the crossflow instability can also occur; it is expressed in the form of steady and traveling disturbances growing downstream. The development of all these disturbances and their relative roles in the transition process depend strongly on the conditions in the flow past a model [4, 5].

For this reason, a small variation in the angle of attack of a swept wing can lead to a change of the transition Reynolds number. We will consider the results of experiments [6] performed on the model delta wing with the slip angle of 77.1° at M = 3.5, which corresponded to the flow regime with a subsonic leading edge. Using the flow visualization on the upper surface of the model it was obtained that with increase in the angle of attack from –2° to 3° laminar-turbulent transition is displaced upstream. We note that the wake behind the delta wing nose includes a region of nonuniform flow [7, 8]; for this reason, it seems better to deal with the data on transition near lateral edges of delta wings. As shown in [6], in this case transition occurs parallel to the lateral edge of the model. In [9] flow visualization was applied to obtain the data on transition on the NACA64A008 model wings with the sweep angles of 45° and 65° at M = 2 and the angle of attack varying from –4° to 4°. The cases of supersonic and subsonic leading edges were realized, respectively. It was obtained that an increase in the angle of attack leads to flow stabilization, that is, to an increase in the transition Reynolds number.

In [10] a thin model wing with two sweep angles was used at the Mach number 2. The sweep angle of the wing was 66° near the airframe and 61.2° near the wing end. The angle of attack varied from −1.5° to +5.5°. The surface hot-wire anemometer probes located at 70% along the wing span demonstrated a monotonic increase in the transition Reynolds number from $Re_{tr} \approx 0.6 \times 10^6$ to $Re_{tr} \approx 1.3 \times 10^6$ with increase in the angle of attack. However, when the probes were located at 30% along the wing span, a maximum in the $Re_{tr}(\alpha)$ distribution was fixed at the angle of attack $\alpha = 1.73°$. A slight variation in the angle of attack near this value led to a considerable upstream variation in the transition location.

In [11, 12] the effect of small angles of attack on the natural disturbance development in the boundary layer on a swept wing with a supersonic leading edge was studied for the first time and the results obtained were compared with the data calculated in accordance with the linear stability theory. Owing to the character of natural fluctuations (random and with uncontrolled wave characteristics), only qualitative agreement between the theoretical and experimental results could be obtained. The transition Reynolds numbers obtained in the experiments for different angles of attack of the model were compared with the calculated dependences of the N factor on the longitudinal coordinate or the Reynolds number obtained for the most rapidly growing waves. It was shown that the estimated location of laminar-turbulent transition based on the calculations according to the e^N method can also be carried out for boundary layers on the model swept wing having supersonic leading edges.

The experimental results considered in [11, 12] were obtained in the supersonic T-325 wind tunnel. The model wing with the leading-edge slip angle $\chi = 45°$ was used at the Mach number M = 2. This corresponds to the case of a supersonic leading edge, while in [13] the leading edge of a swept wing with $\chi = 72°$ was subsonic at M = 2 and 2.5 and supersonic at M = 4. It was obtained that on both model wings the variation in the angle of arrack has a fairly strong effect on the transition Reynolds number. We could not perform on a dull scale the measurements considered in [13] at M = 4. Owing to methodological problems, the transition points were obtained for only two angles of attack. For this reason, a demand arose in additional data for M = 3 and 3.5 which provided the transition of the flow regime past the leading edge from subsonic to supersonic, which was necessary for performing the numerical analysis of the experimental results.

On one hand, the boundary layer state can be better interpreted on the basis of hot-wire anemometer measurements of the fluctuations in the boundary layer. On the other hand, the measurements of transition using the total pressure probe on a polished model surface can scarcely be carried out by means of displacing it downstream. The model geometry is not plane. For this reason, there arises the problem of developing a technique for determining laminar-turbulent transition in the supersonic boundary layer on a highly-swept model wing using a hot-wire anemometer at a fixed probe location. This makes it possible to develop also some other techniques of measuring transition in this flow.

An interest in the experimental data on the laminar-turbulent transition location in swept-wing boundary layers obtained under low-noise conditions is obvious. In the supersonic T-325 wind tunnel of the Institute of Theoretical and Applied Mechanics the low-noise regimes are realized at the Mach numbers M = 2 and 2.5 [14]. For the greater Mach numbers the noise is considerably reduced and the level of background disturbances is stabilized with increase in the unit Reynolds number. For this reason, the experimental results can be used in developing the engineering methods of predicting the laminar-turbulent transition location.

This study continues an investigation of the effect of small angles of attack on the laminar-turbulent layer position on swept wings at supersonic flow velocities [11–13] and is devoted to the development of the technique for determining laminar-turbulent transition in supersonic boundary layers using a hot-wire anemometer on the model swept wing with $\chi = 72°$ at low angles of attack. The new data make it possible to generalize the results concerning the transition obtained in the T-325 wind tunnel to the model swept wing with $\chi = 72°$ at low angles of attack and compare them for the first time with the data calculated using the e^N method.

METHOD OF INVESTIGATION

The experiments were performed in the T-325 wind tunnel of the Institute of Theoretical and Applied Mechanics of the Siberian Branch of the Russian Academy of Sciences at the Mach numbers M = 3 and 3.5. The measurements were carried out on the model swept wing with a 3% airfoil near the base and spanwise-varied chord length (498 mm near the chord base and 200 mm at the wing end). The slip angle of the leading edge was 72°, which corresponded to a subsonic leading edge at M = 3 and a supersonic leading edge at M = 3.5. The measurements were performed on the upper surface of the model. The model schematics and the measurement region were presented in [13].

Disturbances in the flow were recorded by a constant-temperature hot-wire anemometer. The anemometer probe was made of a tungsten wire, 10 μm in diameter and 1.5 to 1.7 mm in length. The probe thread overheating was settled as 0.8. In this case, the measured disturbances corresponded to mass flow rate fluctuations [15]. The fluctuating and mean flow parameters were measured using an automated system of data gathering. The experimental data were processed using the fast Fourier transformation; the power spectra were determined from the complete oscillograms. The experimental setup, the automated system of data gathering, the procedure of data processing, and the technique of determining the r.m.s. values of the mass flow rate fluctuations $\langle m' \rangle$ are described in more detail in [15]. As a result of the measurements, the dependence of the disturbance amplitude on the Reynolds number $\mathrm{Re}_x = \mathrm{Re}_1 \times x$ was obtained. It is customary to relate a maximum in the mass flow rate fluctuation distribution with the laminar-turbulent transition location.

The calculations of the mean flow in the supersonic boundary layer on a wing with the leasing-edge sweep angle of 72° were performed for the experimental conditions, together with the calculations according to the linear theory of hydrodynamic stability. The numerical results were obtained for all Mach numbers. The standard procedure was realized, namely, first the mean flow was calculated in the *AnsysFluent* package and then the diagrams were calculated according to the linear stability theory within the framework of the full Lees–Lin system [16] for the conditions of the experiments performed in the T-325 wind tunnel. For calculating the mean flow in the wing boundary layer the wing geometry was first constructed (from the measurements of the actual model dimensions) and then the compressible flow was numerically simulated by means of integrating the complete three-dimensional equations of viscous heat-conducting gas motion. The approach realized and certain results of the mean flow calculations are presented in [17]. Within the framework of the linear stability problem the flowfield in the compressible boundary layer was presented in the form of the sum of the mean flow and a small perturbation. In this study, we consider the spatial stability problem, in which the frequency is assumed to be a real number, while the longitudinal wavenumber is complex. In analyzing the flow stability in the swept-wing boundary layer in local-parallel approximation an assumption is conventionally used that the amplitudes of localized disturbances increase in the external flow direction x, while the transversally periodic disturbances grow along the chord direction s [18]. In more detail the calculation procedure is described in [19].

EXPERIMENTAL RESULTS

The angle-of-attack effect on the location of laminar-turbulent transition in the three-dimensional supersonic boundary layer on the model swept wing with the slip angle of 72° was experimentally investigated at M = 3 and 3.5. Methodological tests were preliminarily performed for developing the measurement procedure. The growth curves were obtained, that is, the dependences of the integral r.m.s. mass flow rate fluctuations on the Reynolds number $\mathrm{Re}_x = \mathrm{Re}_1 \times x$, where x is the longitudinal coordinate. The fluctuation profiles were measured and the locations of laminar-turbulent transition were determined for several values of the angle of attack. In analyzing the experimental results the amplitude-frequency spectra were used, together with the statistical distributions of the fluctuations in the supersonic boundary layer on the model swept wing. The transition locations were measured at a fixed probe position, while the unit Reynolds number Re_1 was monotonically increased up to the values at which the transition location could be fixed from the fluctuation level maximum on the diagram displayed on the screen of a personal computer in the real-time mode. The disturbance growth diagrams were obtained at $z = 60$ mm for the following values of the angle of attack α and the distances from the leading edge of the model x: M = 3; $\alpha = -0.75°$ and $x = 80$ mm; $\alpha = -0.75°$ and $x = 70$ mm; $\alpha = 0°$ and $x = 80$ mm; $\alpha = 0.45°$ and $x = 80$ mm; M = 3.5; $\alpha = -0.8°$ and $x = 50$ mm; $\alpha = 0°$ and $x = 55$ mm; $\alpha = 0.45°$ and $x = 70$ mm; and $\alpha = 0.45°$ and $x = 80$ mm. The results of the measurements are presented in Fig. 1 for the Mach numbers M = 3 (a) and 3.5 (b).

The transition Reynolds numbers Re_{tr} were obtained from the data presented in Fig. 1; they correspond to the maxima in the Reynolds-number dependences of the r.m.s. fluctuations $\langle m' \rangle(\mathrm{Re}_x)$. In Fig. 2 we have plotted the dependences of the transition Reynolds number Re_{tr} on the angle of attack α of the model wing with $\chi = 72°$ at M = 3 and 3.5. The growth curves and the $\mathrm{Re}_{tr}(\alpha)$ dependences for M = 2, 2.5, and 4 were obtained in [13] and are also presented in Fig. 2. For the sake of comparison, in the same figure we have presented the results of the measurements made on the model wing at the slip angle of the leading edge

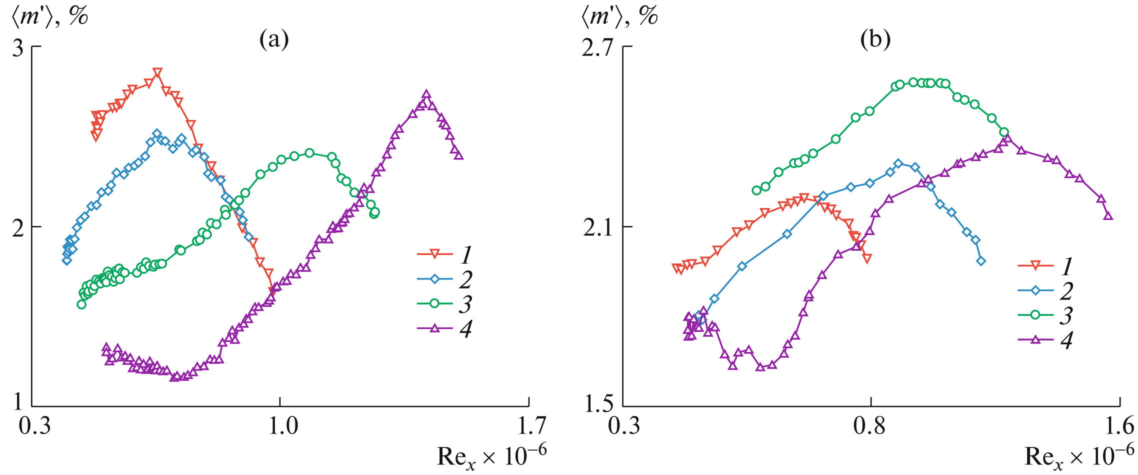

Fig. 1. Dependence of the mass flow rate r.m.s. fluctuations on the Reynolds number at $x = \mathrm{const}$ and $z = 60$ mm: (a) M = 3; *1*, $\alpha = -0.75°$, $x = 80$ mm; *2*, $\alpha = -0.75°$, $x = 70$ mm; *3*, $\alpha = 0$, $x = 80$ mm; and *4*, $\alpha = 0.45°$, $x = 80$ mm; (b) M = 3.5; *1*, $\alpha = -0.8°$, $x = 50$ mm; *2*, $\alpha = 0$, $x = 55$ mm; *3*, $\alpha = 0.45°$, $x = 80$ mm; and *4*, $\alpha = 0.45°$, $x = 70$ mm.

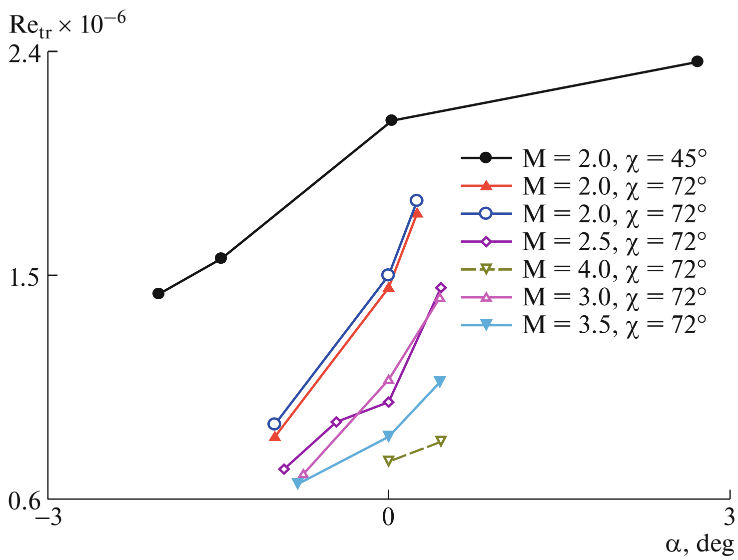

Fig. 2. Dependence of the transition Reynolds number on the angle of attack for different Mach numbers.

$\chi = 45°$ at M = 2 taken from [12]. The results for the model wing with $\chi = 45°$ are taken from [12] and the data for $\chi = 72°$ at M=2, 2.5, and 4 are taken from [13].

From the comparison of the data obtained at M = 2 for different models it can be seen that laminar-turbulent transition in the supersonic boundary layer on the swept wing with the subsonic leading edge occurs at lower Reynolds numbers than on the model with the supersonic leading edge. This result was obtained for the same freestream parameters. A variation in the angle of attack has a stronger effect on laminar-turbulent transition in the boundary layer of the swept wing with the subsonic leading edge than in the case of the model wing with the supersonic leading edge. Thus, on the model wing with the slip angle $\chi = 72°$ the transition Reynolds number $\mathrm{Re}_{\mathrm{tr}}$ increases from 0.9×10^6 to 1.8×10^6 with increase in the angle of attack approximately from $-1°$ to $0.3°$, while on the model with $\chi = 45°$ $\mathrm{Re}_{\mathrm{tr}}$ increases from 1.4×10^6 to 2.4×10^6 with variation in the angle of attack from approximately $-2°$ to $2.5°$. At the Mach numbers M = 2.5 and 3 an increase in the angle of attack from approximately $-1°$ to $0.5°$ leads to the variation in the transition Reynolds number from 0.7×10^6 to 1.4×10^6. A slight flow destabilization occurs at approximately M = 2.5 and laminar-turbulent transition begins approximately 10–20% earlier than for

Table 1

No.	M	T_∞, K	ϕ, deg	P_∞, Pa	$\mathrm{Re}_{1\infty}\times 10^{-6}$, m^{-1}	$\mathrm{Re}_{x,\mathrm{tr}}\times 10^{-6}$	$-\alpha_{i,\max}$, m^{-1}
1			−1	9240	9	0.90	104
2	2	164	0	15600	15.3	1.53	107
3			0.3	18024	17.6	1.76	121
4			−0.46	6000	10	0.91	77
5	2.5	131	0	6600	11	0.99	81
6			0.46	6600	11	1.45	82
7			−0.75			0.81	71
8	3	103	0	5400	13	1.12	76
9			0.45			1.42	80
10	3.5	86	0	2500	11	0.7	73
11	4	70	0	1420	10	0.75	62
12			0.46	1280	9	0.83	56

M = 2. At the Mach number M = 3.5 the variation in the angle of attack α from –0.8° to 0.45° leads to an increase in the transition Reynolds number from 0.7×10^6 to 1.0×10^6, while at M = 4 the variation of α from 0 to approximately 0.5° leads to an increase in the transition Reynolds number $\mathrm{Re}_{\mathrm{tr}}$ from 0.7×10^6 to 0.8×10^6. A decrease in the transition Reynolds number with increase in the Mach number obtained for the model wing with the sweep angle $\chi = 72°$ is in good agreement with the data for the model wing with $\chi = 45°$ at zero incidence [20]. The dependences of the transition Reynolds number on the Mach number can be different in form for different aerodynamic setups, which is due to their design features, which include, as minimum, flow homogeneity and the noise level in the test section, as well as the setup dimensions [21].

RESULTS OF THE CALCULATIONS

The mean flow in the supersonic boundary layer on the model wing was calculated for the experimental conditions, together with an analysis of its stability within the framework of the linear theory. The calculations of the boundary layer stability on the model swept wing were carried out for different Mach number values $2 \leq M \leq 4$ and the unit Reynolds number at the flow stagnation temperature $T_0 = 293$ K. The angles of attack α, at which the model was set, were taken into account. The parameters of all calculated cases are brought together in Table 1. The columns of the Table 1 contain the freestream Mach number M, the static temperature of the flow, the angle of attack of the model α, the static pressure P, the unit Reynolds number Re_1, and the transition Reynolds number $\mathrm{Re}_{\mathrm{tr}} = \mathrm{Re}_1 \times x_{\mathrm{tr}}$ determined in the experiments. The maximum rate of the spatial growth of crossflow instability mode $-\alpha_{i,\max} = \max\limits_{f,\beta^*}(-\alpha_i)$ was calculated for $x = 50$ mm.

The mean boundary layer flow profiles obtained as a result of three-dimensional numerical simulation in the *Ansys Fluent* package along the line of symmetry of the computation domain $z = 0$ were approximated using smoothing splines. Then an analysis of the flow obtained was carried out on the basis of the linear stability theory. We note that the resolution of the computation grid is fairly good: thus, within the boundary layer thickness there were from 30 ($x = 10$ mm) to 140 ($x = 140$ mm) gridpoints. In [17] the formulation of the problem and the profiles of the longitudinal $U = U(y)$ and transverse $W = W(y)$ mean flow velocities are presented, together with the temperature profiles $T = T(y)$, for the Mach numbers M = 2 and 2.5 at zero angle of attack. In this study, the calculations were carried out for the Mach numbers M = 2, 2.5, 3, 3.5, and 4 and for all angles of attack presented in Table 1. In this case, a modified computation procedure was used, as concerned the choice of the computation grids; it was verified against the known data and optimized for reducing the computation time.

By way of illustration, in Fig. 3 we have plotted the growth curves of the most rapidly growing stationary ($f = 0$) instability vortices in the crossflow as functions of $\mathrm{Re}_x = \mathrm{Re}_1 \times x$; they are calculated for the case in which M = 2 and $\chi = 72°$. The analogous calculations were carried out for all cases in Table 1, that is, the Mach numbers and angles of attack.

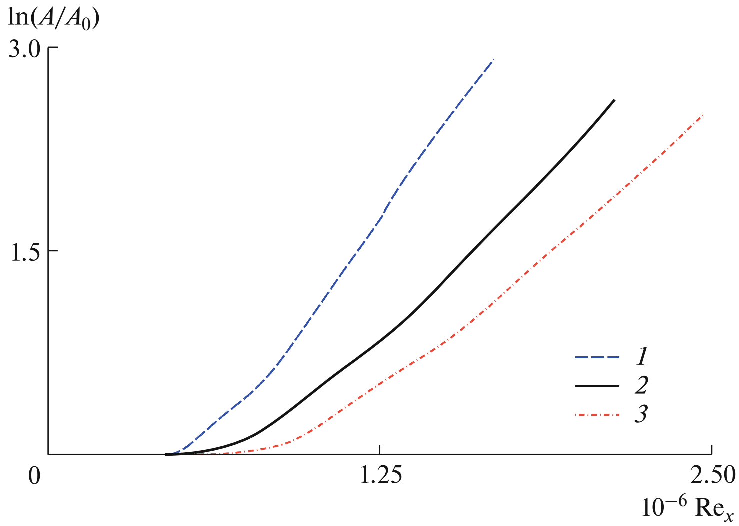

Fig. 3. Curves of the disturbance amplitude growth for stationary ($f = 0$) vortices of the crossflow instability calculated by the e^N method for M = 2 and $\chi = 72°$, where *1* corresponds to $\alpha = -1°$, *2*, 0°, and *3*, 0.3°.

Clearly that in the chosen coordinates the most rapidly growing vortex amplitude occurs for a negative angle of attack (curve *1*), while in the experiment the least value of the transition Reynolds number $\mathrm{Re}_{\mathrm{tr}}$ was obtained precisely for a negative value of the angle of attack. Generally, the behavior of the calculated curves *1*, *2*, and *3* in Fig. 3 is in qualitative agreement with an increase in $\mathrm{Re}_{\mathrm{tr}}$ with the angle of attack.

By way of illustration of the calculations performed, in Fig. 4 we have plotted the boundary layer stability diagrams in the form of isolines of the dimensional spatial growth rates $-\alpha_i$ [m^{-1}] calculated for the longitudinal coordinate (distance from the leading edge of the model) $x = 50$ mm. The data are presented for the same flow parameters and angles of attack as in Fig. 3. The isolines are plotted in the plane of the dimensionless transverse wave number β and the dimensional frequency f [kHz]: $-\alpha_i = -\alpha_i\,(\beta, f)$. It is only the domain $\beta > 0$ corresponding to the most unstable crossflow disturbances propagating counter the secondary flow that is presented on the diagrams. Clearly that at different angles of attack α (Figs. 4a–4c) the instability domain is occupied by the frequency range $0 \leq f < 140$ kHz which lies in the transverse wavenumber range $0 < \beta < 2$. The stability diagrams were calculated for all the cases in Table 1. All the diagrams look like those presented in Fig. 4. A quantitative decrease in the greatest spatial growth rates of disturbances can be observable with increase in the Mach number. The values of these rates are presented in Table 1.

The position of the growth rate maximum for the crossflow instability mode is shown in the diagrams of Fig. 4; in the case in which M = 2 and $\alpha = 0$ (Fig. 4b) its value $\alpha_{i,\,\max} \approx 10^7$ m^{-1}. It corresponds to the frequency $f_{\max} \approx 36$ kHz and $\beta_{\max} \approx 0.8$ rad/mm. We will consider an increase in the angle of attack of the model wing from negative to positive values (Figs. 4a to 4c). Clearly, this increase leads to a certain increase in the maximum local rates of the disturbance growth $-\alpha_{i,\,\max}$. We also note that the stationary ($f = 0$) vortices of the crossflow instability located on the abscissa axis (Figs. 3a–3c) have maximum growth rates $-i\alpha_{i,\,0,\,\max} < 30$ m^{-1}. This is considerably less than the growth rates $-\alpha_{i,\,\max}$ of the unsteady ($f > 0$) modes. Thus, it might be expected that transition on the swept wing due to the crossflow instability will be determined by an enhancement of unsteady high-frequency fluctuations. We will compare the dependences of the transition Reynolds number on the angle of attack of the model obtained in the experiment at M = 2 on the swept wing with the slip angle $\chi = 72°$ (Fig. 2) with the results of the calculations according to the linear theory of stability of the stationary ($f = 0$) vortices (Fig. 3) and unsteady high-frequency crossflow fluctuations. This makes it possible to conclude that in this case the transition is controlled by the stationary mode enhancement. This inconsistence in the conclusions of the theory and the experiments can be attributed to the nonlinearity of the processes observable in the experiments. Thus, in [19] the linear stage of the crossflow instability was theoretically studied and the theoretical results were compared with the data of the experiments [22] performed at M = 2 on the wing with a 7.8% circular-arc airfoil and the leading-edge slip angle of 40°. It was shown that the theoretical data are in good agreement with the experimental results for the transverse scales of the unstable vortices in the secondary flow. How-

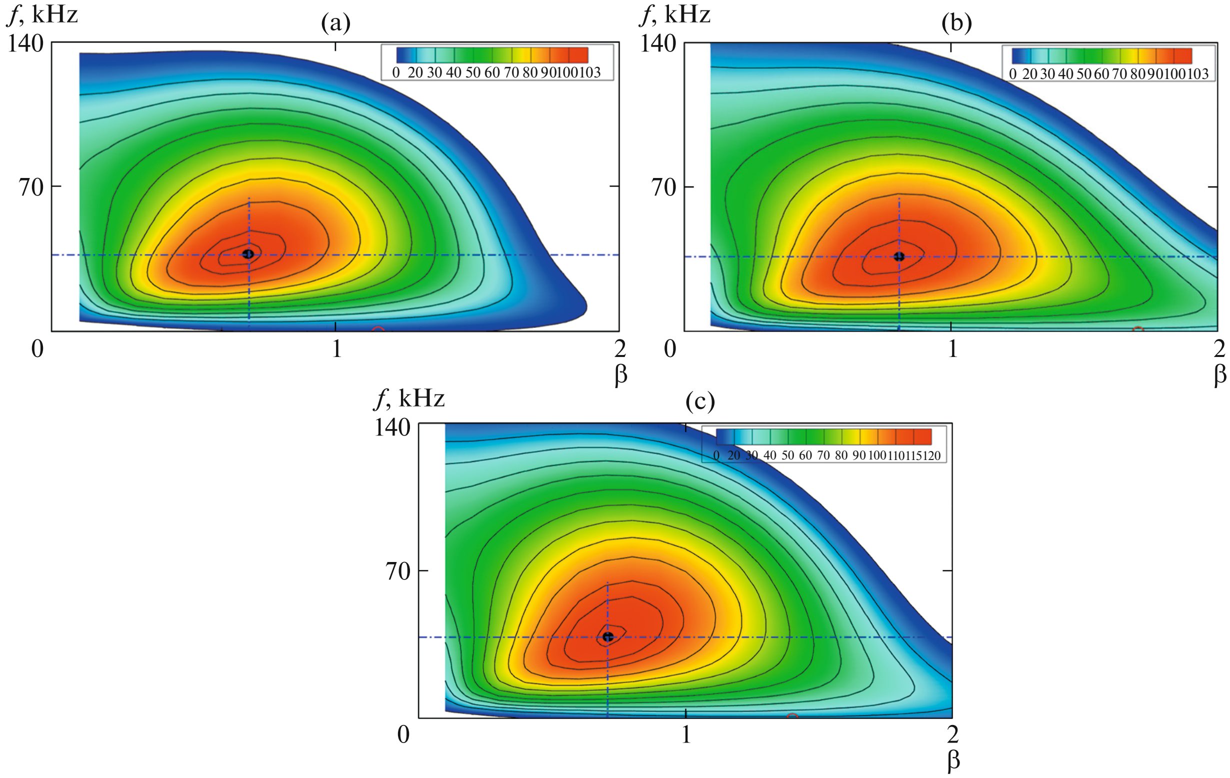

Fig. 4. Stability diagrams for the model swept-wing boundary layer $-\alpha_i$ [m^{-1}] calculated for the longitudinal coordinate (distance from the leading edge of the model) $x = 50$ mm at M = 2 and $\chi = 72°$, where (a) relates to $\alpha = -1°$, (b) 0°, and (c) 0.3°.

ever, the calculated disturbance growth rates are approximately two times higher than those obtained in the case of the experimental data processing. This difference is attributed to the nonlinearity of the processes observable in the experiments. In [11] the experimental growth rates are also compared with the results of calculations for various angles of attack. Good agreement was obtained between the experimental and theoretical data for zero angle of attack. At the same time, this agreement could not be obtained for the angles of attack of $\pm 1°$. The theoretical values of the growth rates are approximately twice as large as the experimental ones. We note that, as shown in [23], the nonlinear processes on the model swept wing with $\chi = 45°$ at M = 2 and $\mathrm{Re}_1 = 5 \times 10^6$ m^{-1} are observable for $\mathrm{Re}_x \geq 0.6 \times 10^6$ in the case of natural disturbances. The experimental data for zero angle of attack were obtained for Re_x in the 0.3×10^6– 0.6×10^6 range, that is, in the linear stage of the disturbance development. At the same time, the experimental values for the angles of attack $\pm 1°$ are in the range of nonlinear development ($0.6 \times 10^6 < \mathrm{Re}_x < 1.1 \times 10^6$). The nonlinear processes in the three-dimensional boundary layer on a swept wing bring into action the secondary instability mechanisms, which lead to the earlier transition.

In Fig. 5 we have plotted the curves of the disturbance amplitude growth for the most rapidly growing stationary ($f = 0$) vortices of the crossflow instability calculated by the e^N method as functions of Re_x on the swept wing with $\chi = 72°$ for different Mach numbers and angles of attack. Here, curve *1* is plotted for M = 2 and $\alpha = -1°$, *2* for M = 2 and $\alpha = 0$, *3* for M = 2 and $\alpha = 0.3°$, *4* for M = 2.5 and $\alpha = -0.46°$, *5* for M = 2.5 and $\alpha = 0$, *6* for M = 2.5 and $\alpha = 0.46°$, *7* for M = 3 and $\alpha = -0{,}75°$, *8* for M = 3 and $\alpha = 0$, *9* for M = 3 and $\alpha = 0.45°$, *10* for M = 3.5 and $\alpha = 0$, *11* for M = 4 and $\alpha = 0$, and *12* for M = 4 and $\alpha = 0.46°$.

The comparison of the curves shows that an increase in the angle of attack from negative to positive values leads to the situation in which the stationary vortices start to grow downstream and grow more slowly. An increase in the freestream Mach number leads to an opposite result, namely, an earlier loss of stability and more rapid growth of vortex amplitudes (cf. the results 2, 5, 8, 10, and 11 in Fig. 5). Thus, the data presented in Fig. 5 generalize the conclusion made on the basis of the data presented in Figs. 2, 3,

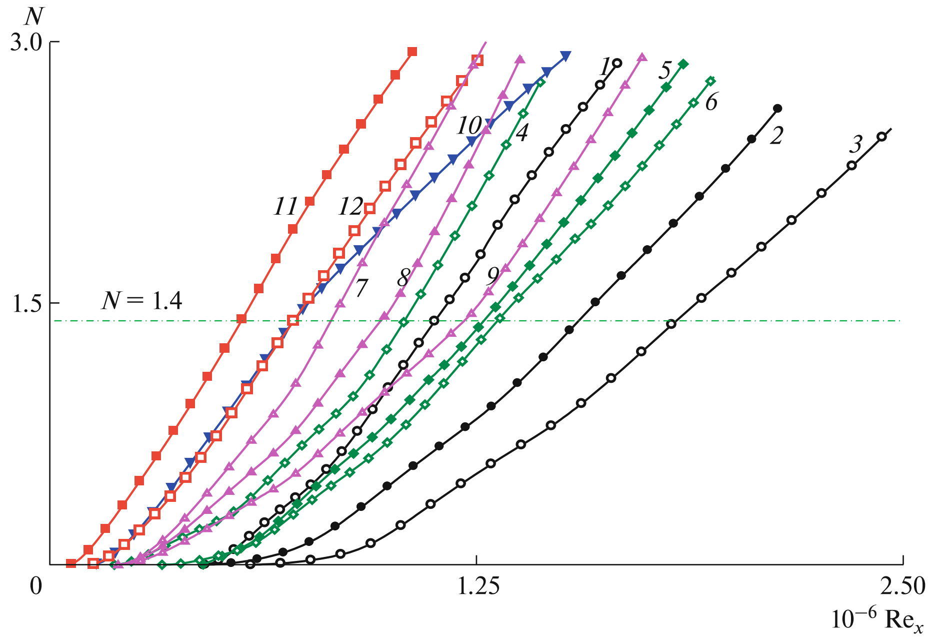

Fig. 5. Growth rates of the disturbance amplitudes for the stationary ($f = 0$) crossflow instability vortices calculated using the e^N method, where the number of a curve corresponds to the number of the case in Table 1.

and 4 for all values of the Mach number considered within the framework of this study. Thus, it may be concluded that the stationary disturbances play the decisive role in the process of laminar-turbulent transition on the model swept wing with the leading-angle slip angle of 72° under the conditions of the present experiments.

SUMMARY

An experimental investigation of the effect of small angles of attack on laminar-turbulent transition in the supersonic boundary layer on a swept wing with the leading-edge slip angle of 72° is performed. New results are presented for the Mach numbers M = 3 and 3.5 and the generalizing results are given for the Mach numbers from 2 to 4. The measurements were carried out using a constant-temperature hot-wire anemometer. The procedure of determining the transition position is developed. Variations of the angle of attack have a strong effect on the transition Reynolds number for all Mach numbers. An increase in the Mach number leads to a decrease in the transition Reynolds number. It is shown that laminar-turbulent transition in the boundary layer on a wing with a subsonic leading edge is more sensitive to variations in the angle of attack, as compared with the case of flow past a model wing with a supersonic leading edge. In the latter case the correlation between the results for the models with $\chi = 45°$ (M = 2) and $\chi = 72°$ (M = 3.5 and 4) can be observable. The data on the influence of small angles of arrack on the transition Reynolds number for each Mach number are obtained under the same conditions, as concerns the noise level in the test section of the supersonic T-325 wind tunnel.

The calculations of the mean flow in a supersonic boundary layer and those according to the linear theory of hydrodynamic stability are performed under the experimental conditions for the wing with the leading-edge sweep angle of 72°. The e^N method is used for the first time for numerically estimating the transition Reynolds numbers in the supersonic boundary layer on a swept wing with the leading-edge slip angle of 72°. The calculations of the amplitude growth of stationary and nonstationary modes of the crossflow in the boundary layer are carried out according to the linear stability theory within the framework of the complete system of Lees–Lin equations. It is shown that the amplitude of vortices increases most rapidly at negative angles of attack. With increase in the angle of attack the growth rate of stationary disturbances decreases. For zero angle of attack the stationary modes of disturbances are more stable at Mach number 2, while with increase in the Mach number the growth rate increases. An analysis of these data and the transition Reynolds numbers obtained in the experiments makes it possible to conclude about the governing role played by stationary disturbances in the laminar-turbulent transition process on the model swept wing with the leading-edge slip angle of 72° under the conditions of the experiments carried out in this study.

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CONFLICT OF INTEREST

The authors declare that they have no conflicts of interest.

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