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EVOLUTION OF DISTURBANCES IN A LAMINARIZED SUPERSONIC BOUNDARY LAYER ON A SWEPT WING

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Experimental data on stability of a three-dimensional supersonic boundary layer on a swept wing are presented. The experiments are performed on a swept wing model with a lenticular profile with a 40° sweep angle of the leading edge at a zero angle of attack. The supersonic boundary layer on the swept wing was laminarized with the use of distributed roughness. A pioneering study of interaction of traveling and stationary disturbances is performed. Some specific features of this interaction are identified. The main reason for turbulence emergence in a supersonic boundary layer on a swept wing is demonstrated to be secondary crossflow instability.

Key words: *supersonic boundary layer, swept wing, transition, stationary disturbances, stability.*

Introduction. The problem of turbulence emergence and development of methods of transition control in three-dimensional boundary layers has been studied by many researchers. Creating a small commercial supersonic aircraft of the new generation is supposed to involve new technologies, in particular, passive control of the laminar–turbulent transition (flow laminarization) in the boundary layer with the use of microscale roughness distributed over the surface of a swept wing near the leading edge.

It was shown [1, 2] that instability of the boundary layer on a swept wing can be controlled with the use of distributed roughness at subsonic flow velocities. The method of passive control has the following features. Microscale roughness elements are applied parallel to the leading edge of the wing, at a distance equal to 1–5% of the wing chord. Based on calculation results for the most unstable stationary mode, the spanwise step between the roughness elements is chosen, which should be approximately $(0.50–0.55)\lambda_{\rm st}$ ($\lambda_{\rm st}$ is the wavelength of the most unstable stationary mode in the direction parallel to the leading edge of the wing). The roughness elements used in [1, 2] were cylinders 6 μm high, which were located near the leading edge of the swept wing. A change in the spanwise distance between the roughness elements was found to affect the position of the laminar–turbulent transition region. For instance, the use of distributed roughness with a 12-mm step along the wing span (or with a step multiple to this value) made the transition region approach the leading edge approximately by 35%, while the use of an 8-mm step increased the laminar flow region by 11%.

Distributed roughness was first used for passive control of the transition in a supersonic boundary layer on a swept wing in the experiments [3, 4] with the use of the method developed for subsonic velocities [1, 2]. Saric and Reed [3, 4] reported that they used surface microroughness to delay the transition to turbulence in a three-dimensional boundary layer on a wing model with a subsonic leading edge. For a supersonic leading edge, the boundary layer was observed to remain laminar on the entire model. Even the use of roughness elements with a step $\lambda_{\rm st}$ did not lead to boundary layer tripping. It should be noted that the values of the transition Reynolds numbers given in [3, 4] are overrated by an order of magnitude. Zuccher et al. [5] obtained reliable values of the transition Reynolds numbers, which contradict the research results [3, 4]. In addition, Zuccher et al. [5] made an attempt to control the transition on a wing model with a subsonic leading edge (i.e., repeat the results of [3, 4]). In this case, however, both with and without roughness elements, the transition occurred in the vicinity of the leading

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edge. Though no reliable data on the location of the boundary layer transition on a swept wing were obtained in [3, 4], the method of controlling the state of a three-dimensional supersonic boundary layer with the use of distributed roughness rises no doubts. In addition, Saric et al. described flight experiments that demonstrated the possibility of using the method of passive control of the transition. The study was performed on a wing model with a supersonic leading edge ($\chi = 30°$) attached under the fuselage of the F-15B aircraft at Mach numbers M = 1.85 and 0.91. With the use of roughness elements (identical to those in the experiments [3, 4]), the transition at M = 1.85 could be delayed until the transition Reynolds number reached the value $\mathrm{Re}_{\mathrm{tr}} \approx 8 \cdot 10^6$.

The study of stability of a supersonic boundary layer on a swept wing [7] showed that the transition to turbulence here is caused by interaction of stationary (large-scale) and traveling (small-scale) disturbances. Hence, for controlling the laminar–turbulent transition in this case, it seems simpler and more effective to affect stationary structures whose character of evolution exerts a greater effect on the location of the transition region.

Semionov et al. [8] proposed an original shape of microroughness elements in the form of streamwise structures distributed along the wing span, which allowed the transition on a swept wing with a supersonic leading edge to be delayed by 40%. The wing model used in the experiments had a lenticular profile and sweep angles of the leading and trailing edges $\chi = 40°$.

To reach effective control of the transition in a supersonic boundary layer on a swept wing with the use of distributed roughness, one has to understand the processes proceeding in such a layer. The majority of investigations of the mechanism of turbulence emergence on an airfoil were performed at subsonic velocities [9–11]. Exceptions are the works described in [7, 12, 13], where stability of a supersonic boundary layer on a swept wing was studied. Ermolaev et al. [12] considered natural traveling disturbances, and Semionov et al. [7] and Kosinov et al. [13] studied the evolution of artificial wave trains and their interaction with a stationary disturbance. Obviously, these experimental data are insufficient for understanding the sophisticated process of the transition to turbulence in a supersonic boundary layer on a swept wing.

The present paper describes the experimental results on evolution of disturbances in a supersonic boundary layer on a swept wing, which was artificially laminarized with the use of distributed streamwise roughness elements. This study allowed us to reach better understanding of evolution and interaction of traveling and stationary disturbances in a supersonic boundary layer on a swept wing and to identify some features of the transition to turbulence in a three-dimensional boundary layer.

Experimental Equipment. The experiments were performed in a T-325 supersonic wind tunnel based at the Khristianovich Institute of Theoretical and Applied Mechanics of the Siberian Division of the Russian Academy of Sciences. The experiments were performed at a Mach number M = 2 with a wing model, which had a lenticular profile and sweep angles of the leading and trailing edges $\chi = 40°$. The model was mounted at a zero angle of attack in the central section of the wind-tunnel test section. The model with a 7.7% profile had the following geometric parameters: length 0.26 m, width 0.2 m, and maximum thickness 20 mm.

The transition was controlled by using distributed streamwise roughness elements [8], which were applied at a distance of 10 mm from the leading edge. The roughness elements were applied with the use of varnish and special templates. The width of the roughness elements was 2 mm, and their length was 10 mm. The distance between the roughness elements was 2 mm. The arrangement of the roughness elements is illustrated in Fig. 1.

The disturbances induced by the flow were recorded by a constant-temperature hot-wire anemometer. The mean and fluctuating characteristics of the flow were measured by an automated data acquisition system. The fluctuating hot-wire signal (in the band of frequencies up to 350 kHz) was measured by a 12-bit analog-to-digital converter with a sampling frequency of 750 kHz, and the mean voltage in the bridge diagonal was measured by an Shch-1516 voltmeter. The length of the signal sample was 65,536 points of the analog-to-digital converter. The discrete Fourier transform was used to determine the frequency spectra of disturbances.

Experimental Results and Analysis. It was found [8] that the use of distributed streamwise roughness lead to flow laminarization on a smooth wing [where the laminar–turbulent transition occurs at $\mathrm{Re}_{\mathrm{tr}} \approx (0.9\text{–}1.0) \cdot 10^6$], and the transition Reynolds number reaches the value $\mathrm{Re}_{\mathrm{tr}} \approx 1.35 \cdot 10^6$. It was in such a laminarized boundary layer that evolution of disturbances was studied.

The measurements were performed in several spanwise sections at x = 45, 60, 75, and 90 mm and the Reynolds number per meter $\mathrm{Re}_1 = 12.2 \cdot 10^6$ m^{-1}, and also at x = 90 mm and $\mathrm{Re}_1 = 14.4 \cdot 10^6$ m^{-1}, which corresponds to the values of the Reynolds number $\mathrm{Re}_x = 0.55 \cdot 10^6$, $0.73 \cdot 10^6$, $0.92 \cdot 10^6$, $1.10 \cdot 10^6$, and $1.30 \cdot 10^6$.

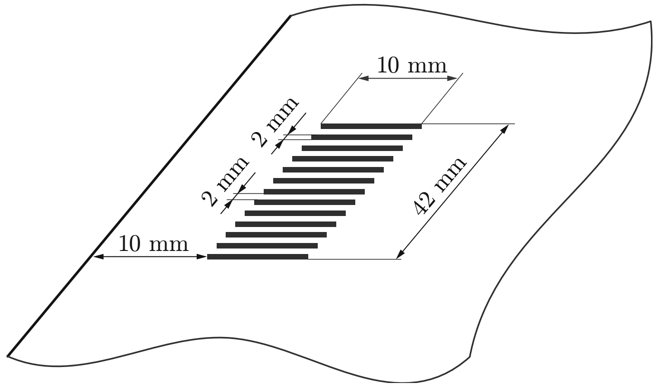

Fig. 1. Arrangement of distributed roughness elements.

The x coordinate was counted from the leading edge of the model in the streamwise direction. The measurements were performed in the section with the maximum fluctuations in the boundary layer at a constant value of the y coordinate by moving the hot-wire probe along the z' coordinate parallel to the leading edge of the model.

Figure 2 shows the mass flow and its root-mean-square fluctuations as functions of the spanwise coordinate z' for several sections (in the layer with the maximum fluctuations). Evolution of traveling and stationary disturbances from the linear stage of their development to the transition region and also interaction of these disturbances are illustrated. In the first section ($x = 45$ mm), one can only start to identify stationary disturbances generated by distributed roughness (Fig. 2a). A tendency to certain periodicity is observed. Though the distance between the end of the distributed roughness elements and the first section is fairly large (25 mm), periodicity in the spanwise direction cannot yet be clearly identified. Note that spanwise periodicity could not be identified on a smooth wing surface either [12].

In other sections, periodicity of stationary structures is clearly expressed. Periodicity of disturbances along the wing span is approximately 4 mm, which correlates well with the period of the distributed roughness elements. Similar results were obtained for distributions of the fluctuating component, where periodicity of maximums and minimums is also observed, and the maximums in the distributions of disturbances in the mean flow rigorously correlate with the minimums of fluctuations in the boundary layer for the sections $x = 60$ and 75 mm (Figs. 2b and 2c). The section $x = 90$ mm (Figs. 2d and 2e) corresponding to the region of nonlinear development of disturbances displays rapid reconstruction of the flow. In the section corresponding to $\mathrm{Re}_x = 1.1 \cdot 10^6$ (Fig. 2d), the mass flow distribution remains periodic, whereas the distribution of mass flow fluctuations becomes different. We can even state that the number of periods is doubled. At $\mathrm{Re}_x = 1.3 \cdot 10^6$ (Fig. 2e), the location of the maximum of the distribution of disturbances in the mean flow coincides with the location of the maximum of the distribution of fluctuations in the boundary layer. Similar processes were observed in experiments at subsonic velocities. It was found that the maximum of disturbances at the stage of their linear development is located in the region of the mean velocity shear, while there is no such correspondence in the region of nonlinear evolution [14]. A rapid change in the structure of traveling disturbances in the region of their nonlinear evolution was also noted in the experiments [7], where the evolution of controlled disturbances at $\mathrm{M} = 2$ was studied on the same wing model as that used in the present work. It follows from Fig. 2 that the amplitude of both traveling and stationary disturbances first increases and then decreases at the end of the transition region. A similar result was obtained in [7].

Additional information on disturbance evolution can be obtained by analyzing the amplitude–frequency spectra of disturbances. Figure 3 shows the spectra of disturbances obtained by applying the discrete Fourier transform. Note that the spectra corresponding to the points with the maximum or minimum root-mean-square fluctuations of the mass flow differ from each other (see Fig. 2). For comparison, therefore, Fig. 3 shows the spectra corresponding to the points with the maximum fluctuations of disturbances in the section. At the initial stage of disturbance evolution, the spectra have two maximums (curves 1–3). The first maximum is observed at a frequency $f \leqslant 5$ kHz, and the second maximum corresponding to frequencies of increasing traveling disturbances [7, 15] is observed in the region of frequencies $f = 10$–60 kHz. As the Reynolds number increases, intense excitation and growth of fluctuations occur in a frequency band between 10 and 60 kHz. In addition, high-frequency modes can

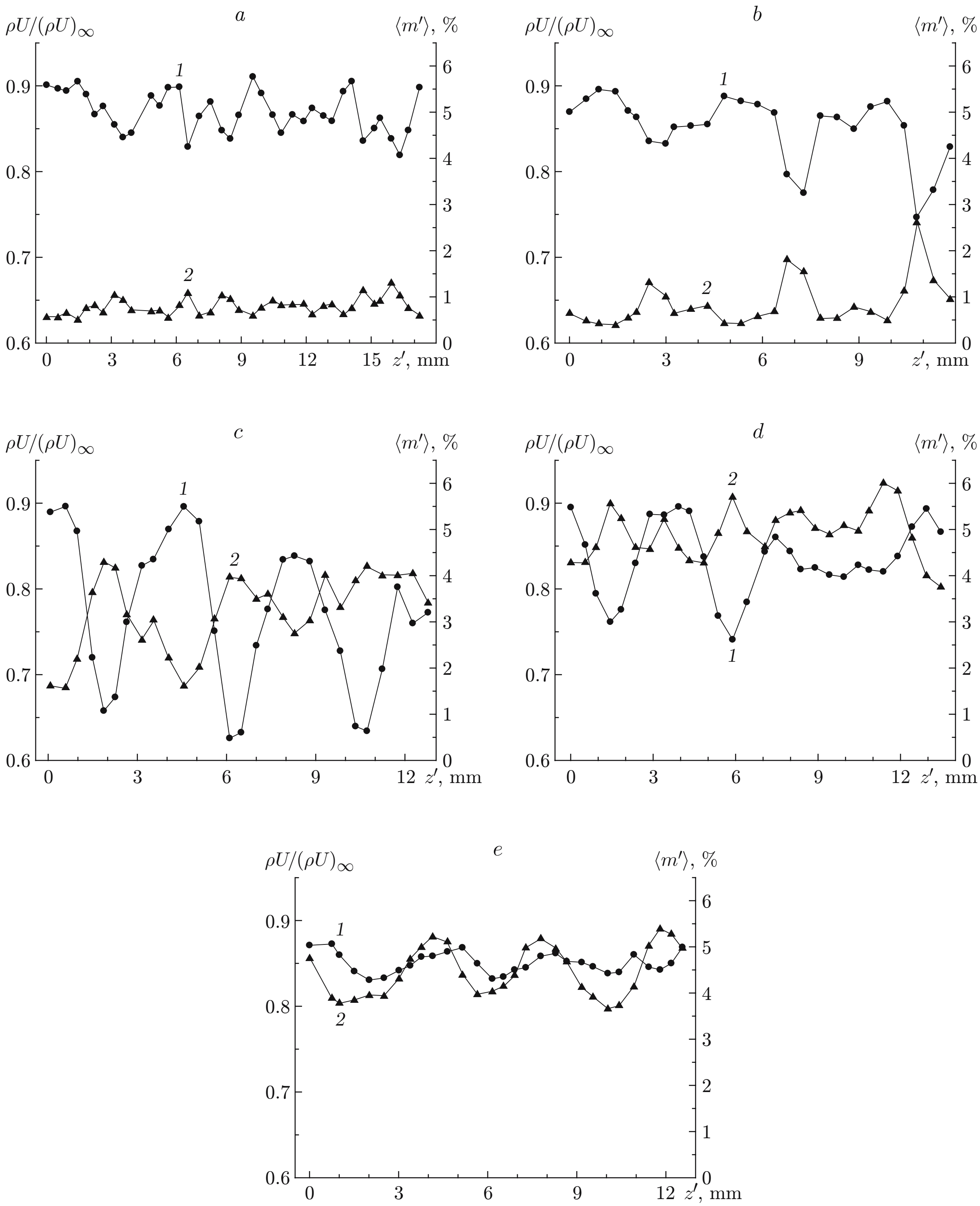

Fig. 2. Distributions of the mass flow (curves 1) and its root-mean-square fluctuations (curves 2) along the wing span for $\mathrm{Re}_x = 0.55 \cdot 10^6$ (a), $0.73 \cdot 10^6$ (b), $0.92 \cdot 10^6$ (c), $1.1 \cdot 10^6$ (d), and $1.3 \cdot 10^6$ (e).

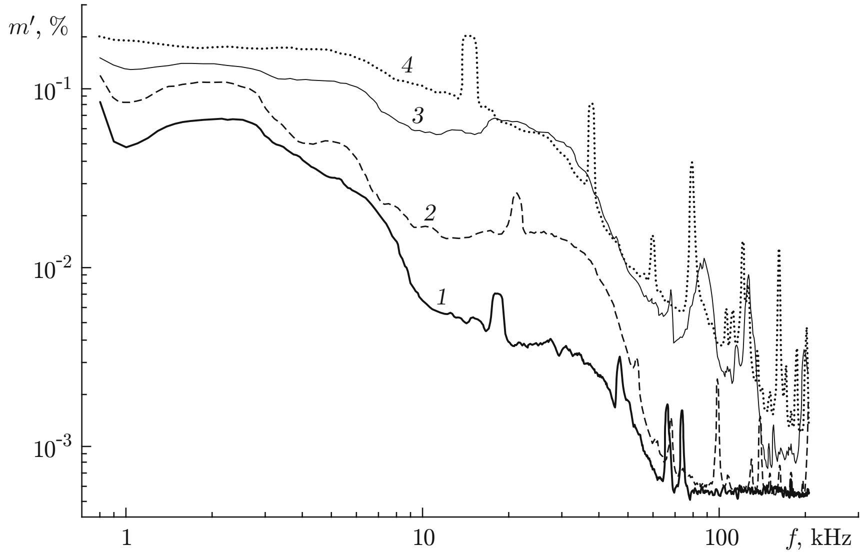

Fig. 3. Amplitude–frequency spectra of disturbances for different values of the Reynolds number: $\mathrm{Re}_x = 0.55 \cdot 10^6$ (1), $0.73 \cdot 10^6$ (2), $0.92 \cdot 10^6$ (3), and $1.1 \cdot 10^6$ (4).

be identified in the spectra. Excitation of high-frequency disturbances growing in the downstream direction was observed in experiments, which is consistent with the calculated results on secondary crossflow instability [16]. Hence, we can argue that secondary crossflow instability plays the governing role in the transition of the laminar flow in a supersonic boundary layer on a swept wing model. It should be noted that secondary instability at subsonic velocities is responsible for the transition on a swept wing [9, 17]. It was demonstrated [9] that secondary high-frequency fluctuations start growing when the amplitude of stationary disturbances reaches approximately 10% of the free-stream velocity. This is also the process that occurs in the case considered.

At $\mathrm{Re} = 1.1 \cdot 10^6$ and $1.3 \cdot 10^6$ (near the transition region), the amplitude–frequency characteristics almost coincide and are similar to the spectra for a turbulent boundary layer on a swept wing [15]. Rapid changes in the structure of traveling disturbances were also observed in these same sections (see Fig. 2). Similarly, it was found for controlled traveling disturbances [7] that rapid reconstruction of the wave structure (especially at the subharmonic frequency) occurs in the region of nonlinear evolution, which was not observed in the case of subsonic velocities. At the same time, the evolution of controlled disturbances at the initial stage of their development is similar to the evolution of traveling waves in the case of subsonic velocities [18], and the specific features typical of a supersonic boundary layer only are mainly manifested near the transition region.

Conclusions. Evolution and interaction of stationary and traveling disturbances in a supersonic boundary layer on a swept wing are studied. The experimental results are in qualitative agreement with the results obtained at subsonic velocities. The main reason for turbulence emergence in a supersonic boundary layer on a swept wing is confirmed to be the secondary crossflow instability. It is demonstrated that a rapid change in the structure of disturbances in the region of their nonlinear evolution is a typical feature for a supersonic boundary layer on a swept wing.

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