Experimental Study of Origin of Turbulence and Laminar-Turbulent Transition in Inhomogeneous Supersonic Boundary Layers

A.D. Kosinov[1,2], N.V. Semionov[1], A.A. Yatskikh[1,2], Yu.G. Yermolaev[1], G.L. Kolosov[1], V.L. Kocharin[1], A.V. Panina[1], V.Ya. Neyland [3], A.V. Vaganov[3], A.Yu. Noev[3], A.S. Skuratov[3], S.V. Aleksandrov[3], V.I. Shalaev[4]

Abstract

The paper presents the results of experimental studies on laminar-turbulent transition at supersonic speeds performed within the framework of ITAM-TsAGI cooperation during the last decade. Experiments were carried out in the T-325 supersonic wind tunnel of ITAM SB RAS at Mach numbers 2, 2.5 and 4. For measuring the pulsations and the characteristics of the mean flow, a constant temperature hot-wire anemometer was used. The measurements were carried out on models of triangular wings with different swept angles and radii of blunting. The dimensions and characteristics of the inhomogeneities of the flow along the span of a triangular flat wing and the region of the laminar-turbulent transition in the center of symmetry of the model are determined. The most significant flow defects were fixed in the vicinity of the axis of symmetry of the triangular wing and at some distance from the center, which decreased with increasing angle of attack. Excitation of high-intensity perturbations by an external Mach wave in the boundary layer of the delta-wing model with blunt lateral edges was found at Mach numbers 2, 2.5, 4. As a result of the measurements, the impact areas and mass flow pulsations in subsonic, supersonic leading edges were tested. It is shown that the maximum value of pulsations is 12-15% and varies only slightly from the conditions of the flow pattern of the delta wing.

Keywords: laminar-turbulent transition, experiment, supersonic flow, delta wing, swept cilinder

1. Introduction

Study of laminar-turbulent transition in inhomogeneous boundary layers, realized by lifting surfaces of aircraft, recently associated with increased interest. The results of these studies are necessary for the development of high-speed aircraft. A review of the studies on the transition to the surface of triangular wings and on the attachment line of their leading edges is given in [1-3]. It is shown that the transition of a laminar boundary layer to a turbulent boundary layer on a delta wing depends on a large number of parameters. The main ones are Mach number, unit Reynolds number, angle of attack, sweep angle and blunt radius of the leading edges of the wing. But, despite the practical importance, at the present time there are no systematized studies of laminar-turbulent transition on triangular wings in supersonic flow, and the available data often contradict each other. Unfortunately, the available data on the laminar-turbulent transition is not enough to perform some experimental development of perspective aircrafts. First of all, it refers to the boundary layers arising in the flow around delta wings with a finite radius of bluntness of leading edges.

The reasons for this difference in the experimental data can be caused by the complex nature of the flow around the models and high level of turbulence of flow in supersonic wind tunnels. Thus, in [1], formation of three types of regions of increased values of heat fluxes on the windward surface outside the edges and the apex of the triangular wing was recorded. The first region is associated with the laminar turbulent transition of the boundary layer, the second is due to the effect of blunting

[1] *Khristianovich Institute of Theoretical and Applied Mechanics SB RAS, Novosibirsk, Russia., E-mail kosinov@itam.nsc.ru*
[2] *Novosibirsk State University, Novosibirsk, Russia*
[3] *Central Aerohydrodynamic Institute, Zhukovskii, Moscow Region, Russia*
[4] *Moscow Institute of Physics and Technology, Dolgoprudnyi, Moscow Region, Russia*

the apex of the wing, and the third region is observed at some distance from the axis of symmetry. Similar results were obtained in [4].

Note that research of supersonic delta wing flow with the help of hot wire is not conducted. A detailed study of the mean flow and disturbances evolution in supersonic boundary layers on triangular wings can clarify the questions that have arisen.

2. Set-up of the experiments

The experiments were conducted in T-325 low noise supersonic wind tunnel of the ITAM SB RAS at Mach 2, 2.5 and 4. The measurements were carried out on models of triangular wings with different radii of blunting. The main sets of experiments were fulfilled on two models of delta wings. First model was a flat triangular wing with a swept angle of 55 ° and a blunt leading edge (radius of blunting 2 mm). The length of the model is 350 mm and width - 200 mm. The model was installed in the working part of the supersonic wind tunnel at angles of attack 0°, 5° or 10° by turning off the plugs of the test section. The angle of attack of the model was controlled by a quadrant whose accuracy was ± 1 '.Second model was a 60% flat delta wing with a swept angle of 55 ° and blunted nose and lateral edges (radius of bluntness 7.5 mm). General views of the models are shown in Fig. 1 a, b respectively.

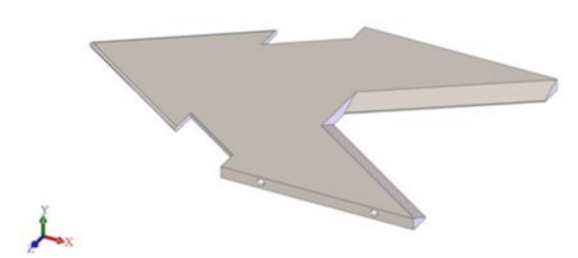

Fig. 1a. Model of a flat triangular wing

Fig. 1b. 3D picture of 60% model of the wing with a sweep angle 55°

The measurements of the pulsations and mean flow characteristics were performed with the help of a constant-temperature hot-wire anemometer. Single-wire tungsten probe of diameter 10 μm and length 1.2 mm was used. The overheat ratio of the wire was 0.8, and the measured disturbances corresponded to mass-flow pulsations. The pulsation and mean flow characteristics were measured by an automated data acquisition system [5]. The measurement system consists of instrumentation apparatus in CAMAC standard, controller CC 32 and the base PC. The fluctuation signal from the hot-wire anemometer was measured by a 12-bit A/D converter with a digitization on time 1.33 μs. Mean voltage component from the output of hot-wire anemometer was fixed by a digital voltmeter Agillent 34401A. The length of each realization was 65536 points. The sensor in the experiments was moved in three co-ordinates: x, y, z, where x coordinate is measured downstream from the nose of the, the coordinate Z - from the symmetry line relative to the top of the wing perpendicular to the flow, the coordinate Y - normal to the model surface. The coordinates x = 0, z = 0 correspond to the nose of the models. Accuracy of the coordinates measurements for x and z is 0.1 mm, and for y - 0.01 mm.

Processing of the experimental data is described in detail in [3, 5]. The absolute values of the mass-flux fluctuations <m'> were calculated by the method described in [5]. The interaction of stationary weak shock waves with the flow in the vicinity of the attachment line at the leading edges of the models was studied using two-dimensional roughness elements of predetermined height and width placed on the wall surface of the test section or nozzle [5].

3. Results

An experimental study of regions of inhomogeneity of the flow and transition on a flat triangular wing with a swept angle of 55· and blunted leading edge (radius of blunting 2 mm) was carried out at Mach numbers 2, 2.5, and 4, and at angles of attack 0°, 5° and 10°. In this case, it is necessary to control the supersonic flow around the wing model at large angles of attack. Shadow visualization showed that problems with installing a supersonic flow regime around the model at an angle of attack only occur at M = 2 and α = 10°, so measurements were not carried out in this case. The position of the laminar-turbulent transition on the axis of symmetry of the model of a flat triangular wing with blunt edges was determined using a hot-wire anemometer. The measurements were carried out on a flat triangular wing at a fixed position of the hot-wire probe x = 80 mm, a unit Reynolds number Re_1 was changed (the coordinate x was measured from the model's leading edge, and the value z = 0 corresponds to the symmetry of the model). The dependence of the mass flow pulsations <m'> on the Reynolds number $Re_x = Re_1 \times x$ for various Mach numbers and angles of attack are shown in Fig. 2. The position of the end of the laminar-turbulent transition corresponds to a maximum in the dependence of mass flow pulsations <m'> (Re_x). It is obtained that the maximum transition Reynolds numbers Re_{tr} are observed at zero angle of attack and at M = 2.5, the increasing of the angle of attack leads to decreasing of Re_{tr}.

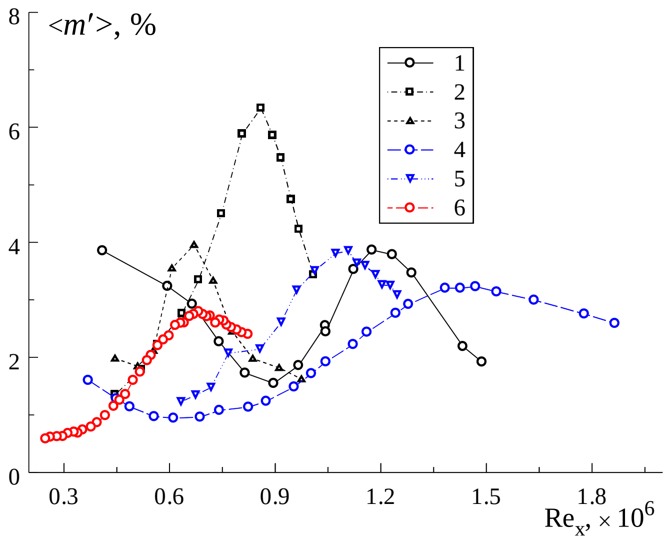

Fig. 2. Curves of the disturbances growth; 1 — M = 2.0, α = 0°; 2 — M = 2.0, α = 2°39′; 3 — M = 2.0, α = 4°55′; 4 — M = 2.5, α = 0°; 5 — M = 2.5, α = 9°40′; 6 — M = 4.0, α = 0°

To determine the dimensions and characteristics of the flow inhomogeneity in the central part of the model, we carried out hot-wire measurements with respect to the transverse coordinate in the cross section x = 80 mm for the selected values of Mach number and angles of attack (M = 2, α = 0°, M = 2.5, α = 0°, α = 10°, M = 4, α = 0°, α = 10°). The main attention was paid to the study of the flow singularity in the center of the model. The most complete data were obtained at M = 2.5, and therefore they are given in the paper. Fig. 3 shows the distribution of mass flow ρU as a function of the transverse coordinate z obtained for different distances y from the model surface. At z <-58 mm, the sensor is probably behind the shock wave. The changes in the mass flow observed in the distributions indicate a significant inhomogeneity of the flow at y = const. The most significant defect of the flow is found in the center of the model and in the region of the lateral edge. In the region z = 0 mm, the inhomogeneity of the mean flow increases as the model surface is approached.

Note that with respect to the normal coordinate, the boundary of the inhomogeneity of the flow can not be determined exactly from these measurements. It can be assumed that in the center there is an increase in the dimensions of the inhomogeneity to 5 mm or more. When displaced from the axis of symmetry ($z \approx$ 17 mm), this heterogeneity has a lower height. The corresponding distributions of mass flow pulsations along the transverse coordinate, measured at different distances from the model surface, are shown in Fig. 4. These distributions also have a significant flow heterogeneity throughout the span of the wing. A strong increase in the level of pulsations when approaching the surface of the model is observed in the vicinity of $z \approx$ 17 mm. In the center and on the side of this region ($z \approx$ 32 mm), an increase in the level of pulsations upon approaching the wall is also observed, but approximately half as large. Note that according to the average stress the greatest defect of the flow was observed at the center of the model, and by the pulsations the inhomogeneity is "more active" in the vicinity of $z \approx$ 17 mm. The most significant defect of the flow is found in the center of the model. More detailed results are described in [2].

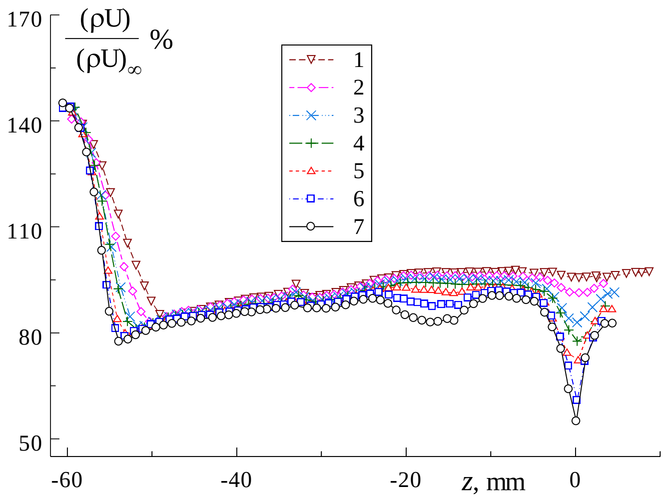

Fig. 3. The distributions of mass flux ρU along the transverse coordinate z at M = 2.5, α=0°, Re_1=8.2×10^6 m^{-1}; 1 — y = 4 mm; 2 — y = 3.2 mm; 3 — y = 2.4 mm; 4 — y = 2.0 mm; 5 — y = 1.6 mm; 6 — y = 1.2 mm; 7 — y = 1 mm

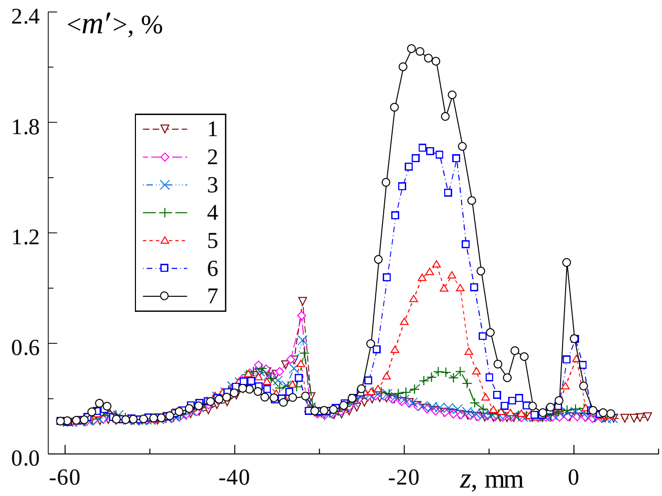

Fig. 4. The distributions of mass flow pulsations <m'> along the transverse coordinate z at M = 2.5, α=0°, Re_1=8.2×10^6 m^{-1}; 1 — y = 4 mm; 2 — y = 3.2 mm; 3 — y = 2.4 mm; 4 — y = 2.0 mm; 5 — y = 1.6 mm; 6 — y = 1.2 mm; 7 — y = 1 mm

The complexity of the observed phenomena accompanying the laminar-turbulent transition and the lack of data for supersonic edges due to the effect of the blunt radius on the flow turbulence process require that experiments be performed at different thicknesses of the model. So experimental study of the structure of a shear flow was carried out on the over a sixty percent model of a flat triangular wing with blunt leading edges (Fig. 1b) at Mach number M=4.0. The unit Reynolds number in all the experiments on this sцузе-wing model was Re_1 = 10.5 × 10^6 m^{-1}.

The flow around the triangular wing model forms a complex three-dimensional flow. It is not possible to obtain the complete field of the mean flow over the model in experiments. Therefore, a number of control cross-sections were chosen for performing the measurements. In Fig. 5 shows the distribution of mass flow pulsations <m'> and the normalized average mass flow rate ρU as a function of the transverse coordinate z at x = 90 mm, at different distances y from the model surface. Near the wall, intense pulsations of mass flow are detected (Fig. 5), which decrease with increasing y, but they remain large near the symmetry line relative to the wing tip, which is analogous to the previous results [2]. Despite the considerable magnitude of pulsations near the wall and in some other distributions along y, the nature of the change in the statistical characteristics of pulsations as a whole over the cross sections makes it possible to assert that the flow past the model is laminar. In this case, there are regions of nonlinear perturbation interaction at sufficiently small amplitudes. The results for the mean flow measurements, presented in Fig. 5, show a decrease in the region of the inhomogeneous flow with increasing y. The inhomogeneity of the flow is preserved only in the region of the line of symmetry (in the neighbourhood of z = 0).

A comparison of the mass flow distributions <m′> and the normalized average mass flow rate ρU as a function of the transverse coordinate z at y = 0.7 mm at different distances from the nose of the model are shown in Fig. 6. It is obtained that the growth of pulsations in the z-sections begins at a distance of 25-30 mm from the lateral edge (shock wave). This may mean that the transition in the shear flow on the model take place almost parallel to the lateral edges, which is confirmed by the results of the measurement of the transition by thermal indication coatings or other temperature-sensitive methods. A sharp peak in pulsations at z = 0 at x = 120 mm may indicate a laminar-turbulent transition in this region.

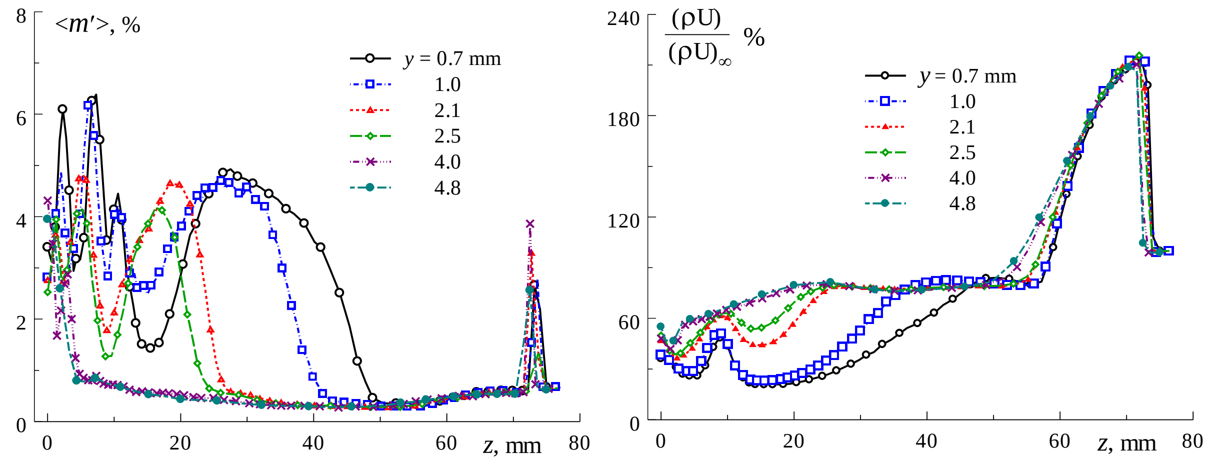

Fig. 5. Mass flow pulsations <m′> and the normalized mean mass flow ρU distributions over z–coordinate at x=90 mm at different distances y from the model surface.

the characteristics of the mean flow at y = 0.7 mm for the cross sections indicate a more filled flow profile at x = 120 mm compared to x = 60, 90 mm. The position of the lateral shock wave is fixed both from the results of measuring the mean flow in the form of a sharp drop in the average mass flow rate, and according to the data for pulsations, in the form of sharp peaks on the right on the graphs. Several measurement points were performed in a free flow in front of the shock wave to determine the boundary conditions along the mean flow. These points are plotted at the end of the graphs on the right.

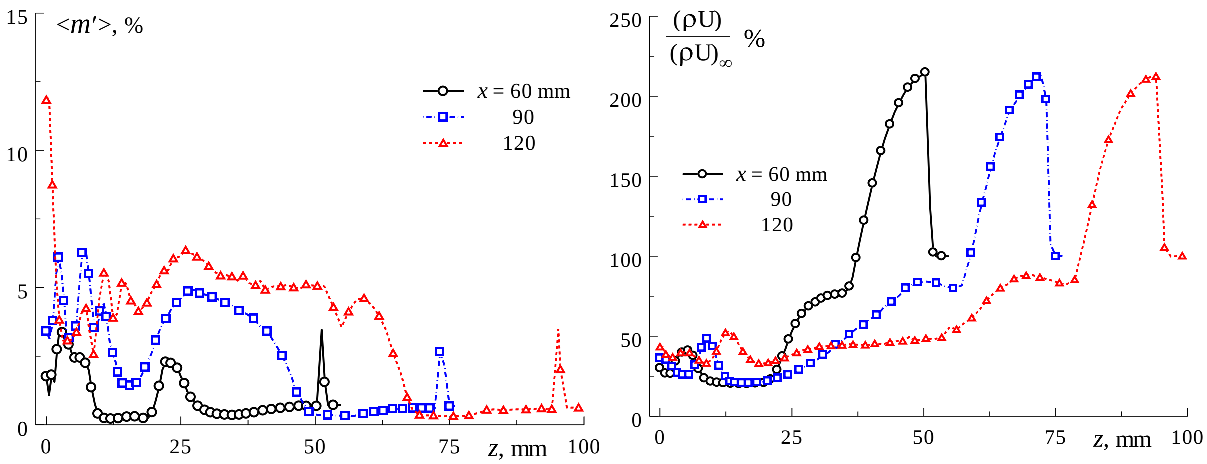

Fig. 6. Mass flow pulsations <m′> and the normalized mean mass flow ρU distributions over z–coordinate at y = 0.7 mm at different distances from the nose of the model

Fig. 7 shows the pulsations of mass flow <m′> and the normalized mean mass flow rate ρU at x = 60, 90 and 120 mm, respectively, measured near the leading edge. The profiles were measured at the end of the region of uniform flow between the onset of pulsation nonlinearity and the lateral edge. There is an increase in pulsations at the maximum with increasing distance from the model nose. Since these measurements were made at points practically parallel to the leading edge, the thickness of the boundary layer was the same. The position of the main maximum in pulsations (Fig. 7) for the coordinate x = 90 and 120 mm is observed at the same values of y, and for x = 60 mm closer to the surface of the model. The mean mass flow profiles for the coordinates x = 90 and 120 mm are also similar. Apparently, this behavior of the pulsation profiles and the mean mass flow rate at x = 60 mm is due to the not yet fully formed flow due to the model nose.

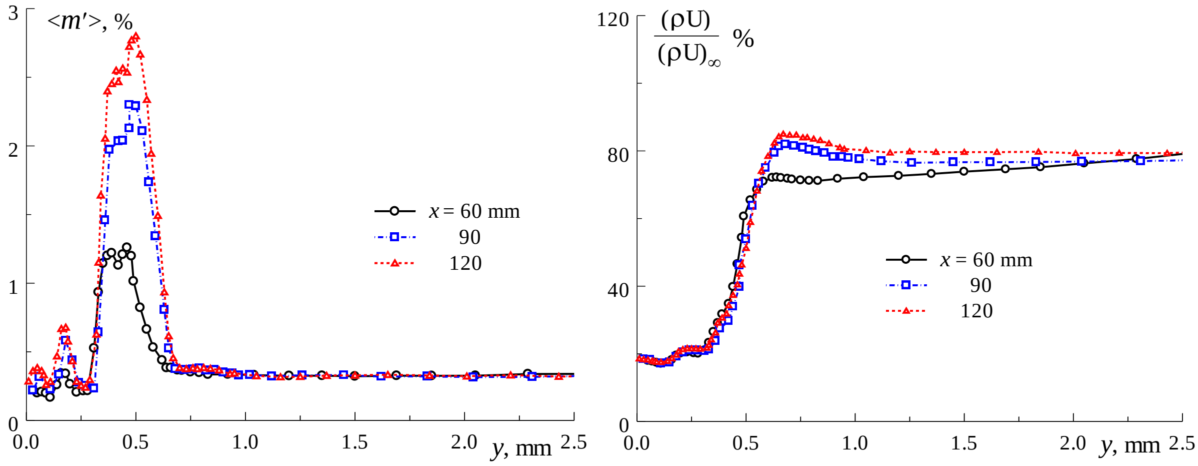

Fig. 7. Profiles of mass flow pulsations <m′> and the normalized mean mass flow ρU

Experimental studies of the transition onset in the supersonic boundary layers carried out at ITAM SB RAS over the past five years have made it possible to establish a special scenario for a laminar-turbulent transition arising due to the interaction of stationary weak shock waves with the flow in the vicinity of the attachment line at the leading edges of the models. This scenario introduces its eigen peculiarities in the process of turbulence in the supersonic boundary layer, which was not previously described and has not been investigated in detail. Such steady-state disturbances in supersonic flow of the T-325 wind tunnel were detected during the installation of models in the region of windows in test section for optical flow visualization with the simultaneous measurement of flow parameters in the boundary layer of a delta wing by a hot-wire anemometer or a Pitot tube [2]. In experimental study of flow around a delta wing with a cylindrical leading edge and swept angle of 55° at a Mach number M = 4 (supersonic leading edge, radius of blunting 2 mm) found abnormally high levels of mass flux pulsation in the boundary layer of model, probably due to interaction of the incident external Mach wave with a boundary layer on the attachment line. Their maximum values within the boundary layer reached about 20% of the local value of the mean mass flux. The high-intensity pulsations with the width of 3 - 4 mm, were observed downstream up to 30 - 40 mm. And they were causing the laminar-turbulent transition in the boundary layer. The main part of the pulsations energy was in the low frequencies (up to 3 kHz). Obviously, such a phenomenon may cause an additional heat and vibration strain on the plane.

In experiments [3, 5-7], it was possible to simulate this situation using two-dimensional roughness elements of controlled height and width placed on the lateral surface of the walls of the test section or nozzle. It was found that these stationary disturbances, observed during shadow visualization in the form of Mach waves, are generated even by a very small protrusion or ledge of the optical windows on the side wall of the wind tunnel (which in fact always existed). In model experiments, the height of the two-dimensional surface roughness was less than 0.01 mm, and its width (along the stream) was about one thickness of the turbulent boundary layer on the walls of the test section. Hot-wire measurements of the artificially supersonic flow perturbed in this manner performed in the test section made it possible to establish that the shape of the stationary perturbation (in form of the normalized mean mass flow) resembles the N-wave. From shadow visualization, it possible to observe a pair of Mach waves generated by the front and back edges of a 2D sticker on the surface of the T-325 nozzle.

An example of results of investigation of the radiation field from the two-dimensional sticker placed on the sidewall of the test section of the wind tunnel at Mach 2 and unit Reynolds number 8.3×10^{6} m^{-1} are shown in Fig. 8. The measurements are made in the free stream. The distance (L – 10 mm) from the left edge of the sticker to the measurement section was 148 mm. The results of the measurements presented in Fig. 8 are functions of mass flow pulsations <m′> and the normalized mean mass flow ρU over the transversal z–coordinate. Transversal mean flow distortion at Mach 2 is similar to that of N-wave. From pulsation distribution shown in Fig. 8., it follows that the conditions for the undisturbed flow correspond to the data for z > –2.5 mm. Here **1** – the free flow region unperturbed by Mach waves; **2** – the region of the passage of a weak shock wave from the downstream (right) edge of the sticker; **3** – the region of the oncoming stream located between the Mach waves; **4** – the area of passage of a weak shock wave from the upstream (left) edge of the sticker.

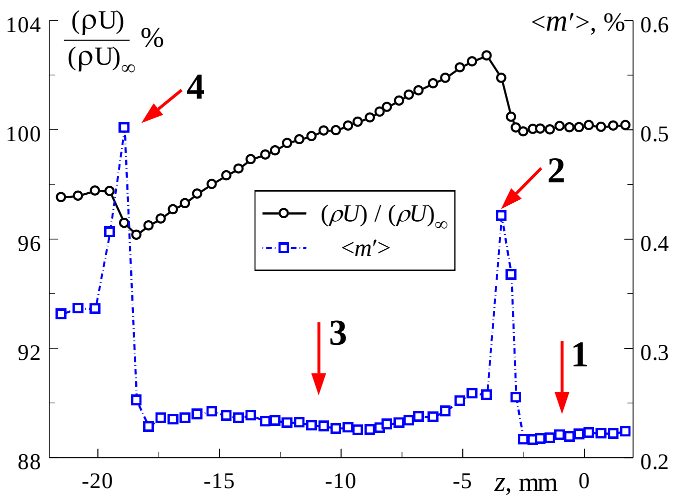

Fig. 8. Mass flow pulsations <m′> and the normalized mean mass flow ρU distributions over z–coordinate in free flow at M = 2.

Amplitude-frequency perturbation spectra are shown in Fig. 9. Here the data are presented for the regions of passages for the Mach waves of the measuring sections (regions 2 and 4). As can be seen from Figure 4, with the passage of a weak shock wave in the incoming stream, low-frequency pulsations with frequencies up to 10 kHz increase. The greatest increase in the disturbances is observed from the Mach wave emanating from the left edge of the sticker.

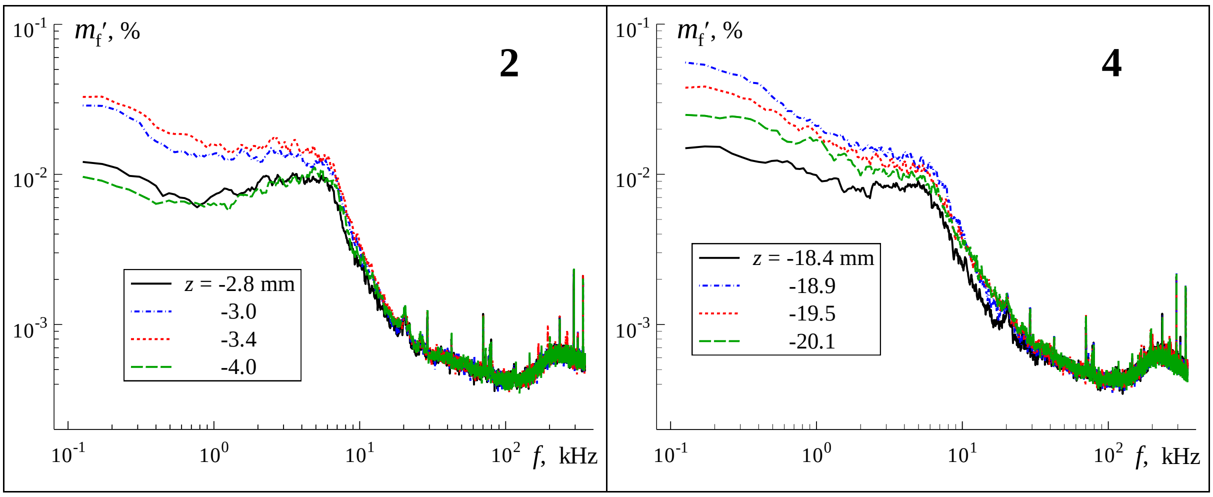

Fig. 9. Pulsation amplitude-frequency spectra in typical probe positions at Mach 2

The excitation of disturbances by weak shock waves in the boundary layer of flat triangular wing with a swept angle of 55 ° and a blunt leading edge (radius of blunting 7.5 mm) were experimentally investigated at Mach numbers 2, 2.5, 4. The studied cases therefore corresponded to subsonic, near sonic and supersonic leading edge. It has found out that the mass flow pulsation reached 12-15% and varied only slightly from the flow conditions. Thus, the effect of the interaction of the incident weak shock wave with the bow shock can be not very significant. The streaks occurrence was always detected in the boundary layer. Example of obtained data is presented in Fig. 10. Fig. 10 shows a comparison of the distributions of the mass flux pulsations <m'> and normalized mean mass flux ρU depending on the transverse coordinate z at M = 2.0, x = 165 mm, y = 0.8 mm for the case of a smooth wall of the test section of wind tunnel and in the presence of inhomogeneity on the sidewall. Similarly to the previous experiment with the wing model with swept angle 55°, measurement data detected the additional areas of disturbed flow: for mean flow - 17 mm in width from z = 33 mm and z = 50 mm and for the pulsation - width of about 17 mm from the z = 35 mm to z = 52 mm. Area of disturbed flow over the pulsations contains several high-intensity peaks, the amplitude of one of them reaches 12.5%. More detailed results are described in [6].

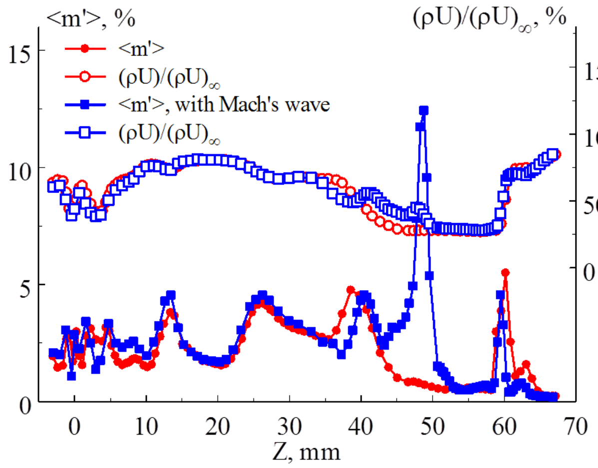

Fig. 10. Comparison of distributions of mass flux pulsations <m'> and normalized average mass flux ρU, depending on the transverse coordinate z at M = 2.0.

The results of measurements on the flat triangular wing with a swept angle of 55 ° and a blunt leading edge (radius of blunting 7.5 mm) are presented below, using inhomogeneity on the surface of the side wall of the test section in the form of a flat ledge at M=4. The longitudinal coordinate of the centerline of the 2D inhomogeneity (an insulating tape 0.15 mm thick, 20 mm wide and about 130 mm long) was x_t = -355 mm. This distance is chosen so that the Mach waves fall on the lateral edge of the wing in the area of the measured control cross sections (Fig. 7). Fig. 11 shows the distributions of mass flow pulsations <m'> and the normalized average mass flow rate ρU as a function of the transverse coordinate z at x = 90 mm, y = 0.7 mm. When the Mach waves generated by the insulating tape hit the lateral edge of the wing, additional regions of the disturbed flow are detected above the surface of the model: for the average flow, 20 mm from z = 30 mm to z = 50 mm, and for pulsations, 25 mm from z = 27 mm to z = 52 mm. The difference in the width of the regions of the disturbed flow can be explained by the nonstationary character of the generation of Mach waves from roughness on the side wall of the test section (there is a detachment in front of the ledge and behind the ledge, even with a boundary layer thickness on the wall of 14 mm). The region of the disturbed flow by pulsations contains several high-intensity peaks, and the amplitude of some of them reaches 11-12%.

For this region of perturbed flow, amplitude-frequency spectra and statistical diagrams were obtained. We should especially note a significant change in the spectra of pulsations, in which a strong increase in the low-frequency part of the spectra is observed, caused by perturbations from Mach waves incident on the leading edge. The density distributions corresponding to these pulsations indicate a nonlinear character of the perturbations.As can be seen from the presented data, the appearance of high-intensity pulsations noted in [2] occurred for all Mach numbers, and, apparently, is caused by the interaction of the Mach wave with the boundary layer on the spreading line. And the remoteness of the lateral shock wave from the surface of the leading edge allows us to state that it does not affect the process of generating high-intensity pulsations.

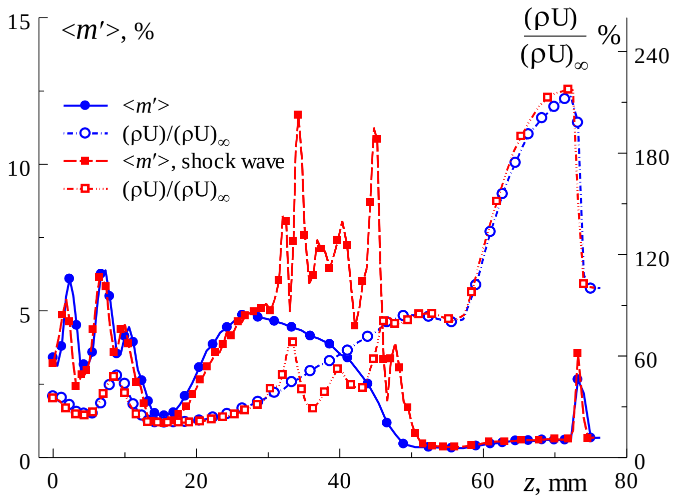

Fig. 11. Comparison of distributions of mass flux pulsations <m'> and normalized average mass flux ρU, depending on the transverse coordinate z at M = 4.0, x = 90 mmm, y = 0,5 mm

4. Conclusions

An experimental study was made of the location of the laminar-turbulent transition and of the mean and pulsation flow characteristics of flow past a flat triangular wing with blunt leading edges. Experiments were carried out in the T-325 low-turbulence supersonic wind tunnel of ITAM SB RAS at Mach number M =2, 2.5 and 4. For measuring the pulsations and the characteristics of the mean flow, a constant temperature hot-wire anemometer was used.

The dimensions and characteristics of the inhomogeneities of the flow along the span of a triangular flat wing and the region of the laminar-turbulent transition in the center of symmetry of the model are determined. The most significant flow defects were fixed in the vicinity of the axis of symmetry of the triangular wing and at some distance from the center, which decreased with increasing angle of attack. It is obtained that the maximum transition Reynolds numbers Re_{tr} are observed at zero angle of attack and at M = 2.5, the increasing of the angle of attack leads to decreasing of Re_{tr}.

Anomalously high levels of mass-flow pulsations (up to 20 % of the local value of mean mass flow) were registered in the boundary layer of a flat triangular wing.at M=4. It was shown that the production of those high-intensity disturbances was attributed to the action of a weak external shock wave on the boundary layer along the attachment line on the leading edge of the wing. The main part of the pulsations energy was in the low frequencies (up to 3 kHz). Obviously, such a phenomenon may cause an additional heat and vibration strain on the plane.

The interaction of stationary weak shock waves with the flow in the vicinity of the attachment line at the leading edges of the models was studied using two-dimensional roughness elements of predetermined height and width placed on the wall surface of the test section or nozzle. Excitation of high-intensity perturbations by an external Mach wave in the boundary layer of the delta-wing model with blunt lateral edges was found at Mach numbers M = 2, 2.5, 4. As a result of the measurements, the impact areas and mass flow pulsations in subsonic, supersonic leading edges. It is shown that the maximum value of pulsations is 12-15% and varies only slightly from the conditions of the flow pattern of the delta wing.

Acknowledgments

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