Experimental study of the laminar-turbulent transition on models of wings with subsonic and supersonic leading edge at M = 2

N V Semionov, Yu G Yermolaev, A D Kosinov, V L Kocharin, A V Panina, A N Semenov, S A Shipul and A A Yatskikh

1Khristianovich Institute of Theoretical and Applied Mechanics SB RAS, 4/1, Institutskaya str., Novosibirsk, 630090, Russia
2Novosibirsk State University, 1, Pirogova str., Novosibirsk, 630090, Russia
3Institute of Geology and Mineralogy, SB RAS, 3, Ac. Koptyuga ave., Novosibirsk, 630090, Russia

Abstract. The paper is devoted to an experimental study of laminar-turbulent transition in supersonic boundary layers on swept wings at Mach 2. The experiments were fulfilled in the low nose supersonic wind tunnel T-325 of Khristianovich Institute of Theoretical and Applied Mechanics on the models with supersonic or subsonic leading edges. The transition location was determined using a constant temperature hot-wire anemometer. While experiments for the case of a supersonic leading edge are a continuation of previous studies, then for the case of a subsonic leading edge, the first data have been obtained that show the possibility of carrying out studies on the development of unstable perturbations. It was obtained that transition take place parallel to the leading edge for both models. The destabilizing effect of the surface curvature of the model on the position of the laminar-turbulent transition for the case of a supersonic leading edge was confirmed. This experimental data is the room for further investigation.

1. Introduction
The study of the laminar-turbulent transition on a swept wing at supersonic flow rates is of practical interest for the development of high-speed aircraft.

There are only a few experiments in which the position of a laminar-turbulent transition was measured on three-dimensional swept wing models in supersonic flow. Supersonic flow around a swept wing is classified according to the Mach number along the normal to the leading edge. The leading edge is called subsonic if Mach number along the normal is M < 1; sound if M = 1, and supersonic if M > 1. Experiments on delta wing with 74° swept angle were fulfilled at Mach number range from 2.8 to 5.3 in the ballistic facilities [1]. So experiments were made for the cases of subsonic and supersonic leading edge (depend from Mach number). The effects of Mach number, temperature ratio, unit Reynolds number, leading-edge diameter, and angle of attack were investigated. The transition was measured by the optical method and so with low accuracy. It was obtained that for the subsonic leading edge Reₜ=3×10⁶ at M=2.78, Reₜ=3.3×10⁶ at M=3.35, for the supersonic leading edge Reₜ=4.8×10⁶ at M=4.1 and Reₜ≈(3÷4)×10⁶ at M=5.3. It is noted in [1] that a high error did not allow to reveal the dependence of Reₜ in the range of angles of attack from -3° to 4°. At the same time the obtained values of Reₜ in the
wind tunnel experiments [2] were approximately 3 times smaller than in [1] in the range of Mach numbers from 2 to 4. The effect of swept angle on laminar-turbulent transition in supersonic boundary layers on flat plate was studied in [2]. It was obtained that at all supersonic Mach numbers, increases in sweep angle reduced the transition Reynolds number. In [3], the results of measurements on a model of a delta wing with a swept angle $\chi=77.1^\circ$ (subsonic leading edge) at $M=3.5$ are presented. It was noted that the transition was observed parallel to the lateral edge of the model and depend from unit Reynolds number.

Transition measurements using hot-film sensors and an infrared camera were performed in [4] on a wing model with a subsonic leading edge at $M=2$ (sweep angles of the inner and outer parts of the wing were $66^\circ$ and $61.2^\circ$, respectively). An effect of angle of attack was studied in two sections along the wing span in the range of angles from $-1.5^\circ$ to $5.5^\circ$. It was obtained monotonic increase of transition Reynolds number $Re_{tr}=0.6\times10^6$ to $Re_{tr}=1.3\times10^6$ with increasing angle of attack at $70\%$ semispanwise probe station. Contrariwise at the $30\%$ semispanwise probe station the longest laminar region was detected at $\alpha=1.7^\circ$, and a slight deviation in the angle of attack from $1.7^\circ$ moved the transition location significantly upstream. But at zero angle of attack transitional Reynolds number was approximately $0.8\times10^6$ in both cases. It was shown that unit Reynolds number had no effect on the transition location [4].

The results of flight measurements of $Re_{tr}$ on a swept wing at $M=2$ are presented in [5]. The leading edge of the wing had the same two swept angles as in [4]. The Reynolds numbers of the transition are not given in the work, and from the presented data we can only say that the transition occurs near the leading edge.

The influence of flow parameters on the position of the laminar-turbulent transition, such as the Mach number [6], unit Reynolds number [7], the influence of external perturbations [8] and small angles of attack [9], was studied at the Institute of Theoretical and Applied Mechanics of the Siberian Branch of the Russian Academy of Sciences on the model of a wing with a supersonic leading edge.

It was shown that with an increase in the Mach number, $Re_{tr}$ decreases [6], with an increase in the unit Reynolds number, the Reynolds number of the transition also increases [7], and with an increase in the angle of attack, $Re_{tr}$ monotonically increases [9]. At a high noise level caused by an increase in perturbations, the effect of flow parameters on the laminar-turbulent transition is practically not fixed [8].

In the course of studies on the influence of various flow parameters on the laminar-turbulent transition in the supersonic boundary layer of the swept wing, it was revealed that in all these cases, $Re_{tr}$ varies from $0.7\times10^6$ to $2.4\times10^6$. The higher the noise level, the more $Re_{tr}$ tends to a value of $0.7\times10^6$. This paper is a continuation of our experimental research on the wing model with supersonic leading edge [6-9] and represents the direction of future studies.

2. Setting up experiments

The experiments were performed in a low-noise supersonic wind tunnel T-325 ITAM SB RAS at Mach number $M=2$. The level of disturbances in the test section of the wind tunnel does not exceed $0.2\%$ [10]. First model is a symmetrical wing with a $45^\circ$ sweep angle, a $3$ percent-thick circular-arc airfoil. The model length is $0.4$ m, its width is $0.2$ m, and the maximum thickness is $12$ mm. Scheme of the model is presented in [11]. Photo of another model with swept angle $\chi=72^\circ$ and lenticular profile is presented in figure 1. Swept angles $\chi=45^\circ$ and $\chi=72^\circ$ corresponds to a supersonic and subsonic leading edge at $M=2$. The experiments were performed at zero angle of attack. Perturbations in the flow were recorded by a constant-temperature hot-wire anemometer (CTA). The experimental data were obtained in two ways. First, a change in the position of the coordinate with a hot-wire anemometer along the model (the $x$ coordinate changes) with a constant mode of operation of the wind tunnel ($Re_{tr}=$const). The second method is to change the operating mode of the wind tunnel when the hot-wire probe is installed in a certain section ($x=$const, $Re_{tr}$ changes). These methods are interchangeable, if a complex geometry model is used; method two is more convenient for obtaining data.
The flow characteristics were measured using an automated data acquisition system [12]. Using an Agilent 34401A digital voltmeter, the DC component of the voltage was measured from the output of the hot-wire anemometer E. The ripple signal in the diagonal of the hot-wire anemometer bridge was digitized by a 12-bit analog-to-digital converter (ADC) and then recorded into a computer. The experimental data were processed using the fast Fourier transform, and the power spectra were determined from complete waveforms.

**Figure 1.** Photo of wing model with $\chi = 72^\circ$ in the test section of T-325.

### 3. Results

As noted above, studies of the influence of flow parameters on the development of disturbances and on the position of the laminar-turbulent transition were performed on a wing model with a supersonic leading edge [6 - 9, 11]. But some questions remained open, for example, how does the transition occur along the wingspan.

The measurement of the transition position on the wing with a swept angle of $45^\circ$ is presented in figure 2. These measurements were carried out using a Pitot tube. The Pitot probe were installed in a fixed position along two coordinates $x$ and $z$, which are indicated on the graph, and during the experiment, the unit Reynolds number ($Re_1$) changed. The measured dependences of the normalized...
pressure are close in form, and the obtained estimates of the transition Reynolds number have a value of \((2 \pm 0.1) \times 10^6\).

The results of measuring the position of the laminar-turbulent transition on a swept wing with a subsonic leading edge (the sweep angle is 72°) are shown in figure 3. Measurements on this model were carried out both by a hot-wire anemometer sensor and Pitot probe. Measurements with a hot-wire anemometer were made only at one point: \(x=110\) mm, \(z=-35\) mm. The hot-wire anemometer sensor was installed in the layer just below the ripple maximum along the vertical coordinate \(y\), and measurements were made with a variation of the unit Reynolds number. Hot-wire anemometer transition measurement data show that the emergence of turbulence on a swept wing with a subsonic leading edge occurs at \((1.5 \pm 0.1) \times 10^6 Re_t\). Pitot probe measurements give close estimates of the transition position.

The dependence of the laminar-turbulent transition on the radius of curvature of the surface is shown in figure 4. A flat plate has an infinite radius of curvature, and a profile of 7.8% is about 0.9 m. Transition measurements using a hot-wire anemometer were performed only for a model with a relative profile thickness of 3%. All other results were obtained using measurements with a Pitot probe. For a flat plate, the transition occurs at \(Re_tr=1.5 \times 10^6\), for a model with a relative profile thickness of 3% \(Re_tr=2.1 \times 10^6\), for 5% - \(Re_tr=1 \times 10^6\), for 7.8% - \(Re_tr=0.9 \times 10^6\). Measurements were taken at different points. We can conclude that with an increase in the relative thickness of the profile, the unit Reynolds number decreases monotonically. In order to confirm or refute this conclusion, measurements are needed that were made at one point with different models of the swept wing.

The curve of the increase in pulsations of the mass flow is shown in figure 5. The measurements were carried out at a distance of 70 mm from the leading edge of the wing with a sweep angle of 72°. The measurements were made by a hot-wire anemometer. The transition from laminar to turbulent flow was not achieved, since at an increased high-pressure head (\(Re_t>15 \times 10^6\) m\(^{-1}\)) the thread of the hot-wire anemometer sensor was torn. Considering that in the model of a swept wing with a subsonic leading edge, it occurs at \(Re_tr=(1.5\pm1.6) \times 10^6\) m\(^{-1}\), then at \(x=70\) mm the transition should occur at \(Re_t=(22\pm24) \times 10^6\) m\(^{-1}\). However, from these data we can conclude that the onset of turbulence occurs at \(Re_t>5.6 \times 10^6\) m\(^{-1}\), which is slightly less than in the case of transition measurements on the same model at \(x=110\) mm (see figure 3).

It was obtained that transition take place parallel to the leading edge for wings with subsonic and supersonic leading edge. The destabilizing effect of the surface curvature of the model on the position of the laminar-turbulent transition for the case of a supersonic leading edge was confirmed. This experimental data is the room for further investigation.

\[\text{Figure } 4. \text{ The effect of surface curvature of the model on the laminar-turbulent transition.}\]

\[\text{Figure } 5. \text{ Growth curve of mass flow pulsations.}\]
Acknowledgments
The study was conducted at the Joint Access Centre “Mechanics” of Khristianovich Institute of Theoretical and Applied Mechanics SB RAS. The work is supported by Russian Foundation for Basic Research (grant number 17-19-01289).

References
[1] Chapman G T 1961 Transition of the laminar boundary layer on a delta wing with 74 deg. sweep in free flights at Mach numbers from 2.8 to 5.3 NASA TN D-1066
[2] Jillie D W and Hopkins E J 1961 Effects of Mach number, leading-edge bluntness, and swept on boundary layer transition on a flat plate NASA TN D-1071
[3] Cattafesta L N III, Iyer V, Masad J A, King R A and Dagenhart J R 1995 AIAA J 33 2032
[4] Sugiura H, Yoshida K, Tokugawa N, Takagi S and Nishizawa A 2002 J. of Aircraft 39 996
[5] Kwak D-Y, Tokugawa N and Yoshida K 2007 WEHSFF conference (Moscow)
[6] Semionov N V, Kosinov A D and Yermolaev Yu G 2011 Journal of Physics: Conference Series 318 032018
[7] Ermolaev Y G, Kosinov A D, Semionov A N, Semionov N V and Yatskikh A A 2018 Thermophysics and Aeromechanics 25, 659
[8] Dryasov A D, Yermolaev Yu G, Kosinov A D, Semionov N V and Semionov A N 2016 Vestnik NSU. Physics 11 16 (in Russian)
[9] Yermolaev Yu G, Kosinov A D, Kosorygin V S, Semionov N V, Semionov A N, Smorodsky B V and Yatskikh A A 2017 Siberian Physics J 12 35 DOI: 10.25205/2541-9447-2017-12-3-35-40 (in Russian)
[10] Kosinov A D, Semionov N V and Yermolaev Yu G 1999 Disturbances in test section of T-325 supersonic wind tunnel Preprint 6-99
[11] Kosinov A D abd Semionov N V 2019 AIP Conference Proceedings 2125 030105 DOI: 10.1063/1.5117487