Observation of quasi-stationary flow regimes in a supersonic wind tunnel

SERGEY GAPONOV, ALEXANDER KOSINOV, NICKOLAY SEMIONOV,
Khristianovich Institute of Theoretical and Applied Mechanics SB RAS (ITAM)
Institutskaya 4/1, 630090 Novosibirsk
RUSSIA
Email: gaponov@itam.nsc.ru, web page:http://www.itam.nsc.ru

Abstract: - Study of supersonic flow onset slowly in time was fulfilled by Schlieren method. The experiments have been conducted for two initial installations of flat delta-wing. Different stages of the normal shock wave passage along the model with increasing of the total pressure are observed. Appearance of normal shock wave, bow shock wave, the flow separation zone near the nose bluntness, origin of the λ-shock wave and supersonic flow are detected. The results of the flow visualization show the high quality of the flow in wind tunnel T-325 test section.

Key-Words: - experiment, supersonic flow, Schlieren method, shock wave, flat delta wing

1 Introduction

In the study of supersonic flow around blunt models it is necessary to control the start and stop of the wind tunnel, as well as the establishment of the flow regime in the course of the experiment in order to save these bodies for the investigation. Highest aerodynamic load model has at starting and stopping of the wind tunnel. Flow regime can change during experiments at angle of attack varying. For the supervision of a supersonic flow in the test section applies shadow visualization.

Supersonic low noise wind tunnel T-325 put into operation in 1968 [1, 2]. First runs were made at M=1.5. It was obtained, that obstruction of the flow in test section (enough big models and traversing equipment) led to transient flow regime. In experiments [3] it was found that the problem with the setting of regime of supersonic flow around model at an angle of attack takes place only at M = 2 and α = 10°. Chocking in the test section of wind tunnel was observed. Note that the increasing of unit Reynolds number (Re_1) led only to a shift in position of the normal shock wave downstream, and it was remained above the surface of the model. So, it is necessary to supervise flow regime in test section especially in experiments with big model or model at angle of attack. And shadow visualization allows controlling a flow around model [4].

This paper presents the results of supersonic flow visualization around flat delta wing at M=1.5, which also shows the quality of the flow in the test section due to the perfect contour of the nuzzle inserts.

2 Experimental Set-up

The experiments were conducted in a supersonic wind tunnel T-325 ITAM SB RAS at Mach 1.5. Model of the triangular flat plate with blunt leading edges is used. The model was mounted at zero angle of attack almost in center of the test section. Shadow picture of the flow is obtained by using two telescopes with diameter apertures of 160 mm, which is equipped T-325 supersonic wind tunnel (Fig. 1). Wireless Internet Camera DCS-3420 with lens MC Volna-9 is used to transfer the shadow picture of the flow in the second computer to display a picture in control room (Fig. 2). CCD camera, telescope and general view of the experiment with the Schlieren method set-up are shown in Fig. 1, 2. Shadow movie with a frequency of 7 frames per second is recorded during the experiment. White LED as the light source is used.

Fig. 1. Photo of optical system for Schlieren method

Fig. 2. General view of the experiment with the shadow visualization

3 Results and Analysis

From the experience it is generally recognized, that is difficult to get a supersonic flow around a model in test section at Mach 1.5. In T-325 we found out that at the slow pressure increasing in the settling chamber (from very low unit Reynolds numbers) normal shock wave can be held in the vicinity of the model nose and then can go further downstream leaving supersonic flow after itself. The supersonic wind tunnel has also nuzzles for Mach 2, 2.5, 3, 3.5 and 4. However only at Mach 1.5 there is the possibility to see the supersonic flow onset slowly in time. Two experiments have been conducted for two different initial model installations. During wind tunnel run the model was rotated around flow axe so in the end of both experiments the model has the same position in test section. Obtained results are shown in figures 3–6.

Different stages of the normal shock wave passage along the horizontally mounted model in test section with increasing of the total pressure are shown in Figure 3. Planar geometry of shock wave (Fig 3a) may indicate on the high quality of the aerodynamic contour of the test section and uniformity of the flow. Further total pressure increasing led to a shift the position of the normal shock downstream until the supersonic flow is set.

This process is accompanied by the formation of the flow separation zone near the nose bluntness, which is starting say in Fig. 3b and is clearly visible in Fig. 3c. Outside contour of normal shock wave is kept but there is sharp boundary between bow shock wave and initial normal shock wave. Near-wall flow separation around the model was accompanied by shock wave up to the establishment of a supersonic flow in the test section by changing its spatial configuration. Formation of the bow shock wave starts from the front of the model a gradual increase in the region of supersonic flow. This happened almost without changing the geometry of the initial wave near the spout, which grows to its final shape while maintaining the previous geometry and flatness of the initial shock wave. Further increasing of the total pressure leads to the formation of the λ-shock wave (Fig. 3d, e). Nose separation zone in this case is disappeared however new one is appeared. Visible distortion flatness of the normal shock is associated with the boundary surface effects and asymmetric with respect to the installation of the model-enforcement test section vertically. Supersonic flow regime is shown in Fig. 3f.

Similar measurements are made for vertical installation of model. The passage of the normal shock wave along the model with increasing of the total pressure is shown in Figure 4. Origin of the bow shock wave is almost the same (see Figs 4a–4d). There are some differences in bow shock associated with the geometry of the model. Outside flatness of normal shock wave is kept also but there is no sharp boundary between bow shock wave and initial normal shock wave at the beginning (see Figs 4c–4d). We can say that this difference there is up to the λ-shock wave formation and then sharp boundary between bow shock wave and initial normal shock wave is appeared. Partially the λ-shock wave is visible in Fig. 4e.

Regimes of supersonic flow past a flat delta wing model obtained in its rotation, shown in Fig. 5. These photos show that in this case there are no any features in bow shock contour. It is very important that it was possible to follow the inverse process during stopping run of the wind tunnel. It was found that the slow decrease of the total pressure the normal shock wave can be held in the vicinity of back side of the model. Further decreasing of total pressure led to a shift in position of the normal shock upstream. Back way of the normal shock wave along the model in the test section of the wind tunnel is shown in Figure 6. It was detected the all stages shown in Fig. 3, but in the reverse sequence.

4 Conclusion

Study of supersonic flow onset slowly in time was fulfilled by Schlieren method. The experiments have been conducted for two initial installations of flat delta-wing. Different stages of the normal shock wave passage along the model with increasing of the total pressure are observed. Appearance of normal shock wave upstream from the model, bow shock wave, the flow separation zone near the nose bluntness, origin of the λ-shock wave and supersonic flow are detected. It was possible to follow the inverse process during stopping run of the wind tunnel. It was detected the all stages in the reverse sequence The results of the flow visualization show the high quality of the flow in wind tunnel T-325 test section. It should be note that some features of the origin of bow shock are still unclear and should be numerically investigated.

References:

[1] G.I. Bagaev, V.A. Lebiga, V.G. Pridanov, V.V. Chernykh, T-325 low-turbulence supersonic wind tunnel, *Aerophysical studies.* Novosibirsk: Inst. Theor. and Appl. Mech. SB AS, 1972. pp. 11-13. (in Russian)

[2] V.G. Pridanov, A.M. Kharitonov, V.V. Chernykh, Combined influence of Mach and Reynolds numbers on transition in the boundary layer, *Fluid Dynamics*, 1974, №1, pp. 126-128.

[3] A.V. Vaganov, Yu.G. Yermolaev, A.D. Kosiniv, N.V. Semionov, V.I. Shalaev, Experimental study of the flow structure and the transition in the boundary layer of the delta wing with blunted leading edges at Mach 2, 2,5 and 4, *Proceedings of Moscow Institute of Physics and Theknology*. 2013. Vol. 5. №3. pp. 164 – 173. (in Russian)

[4] G.S. Settles, *Schliren and shadowgraph techniques. Visualizing phenomena in transparent media*. Berlin: Springer, 2006.

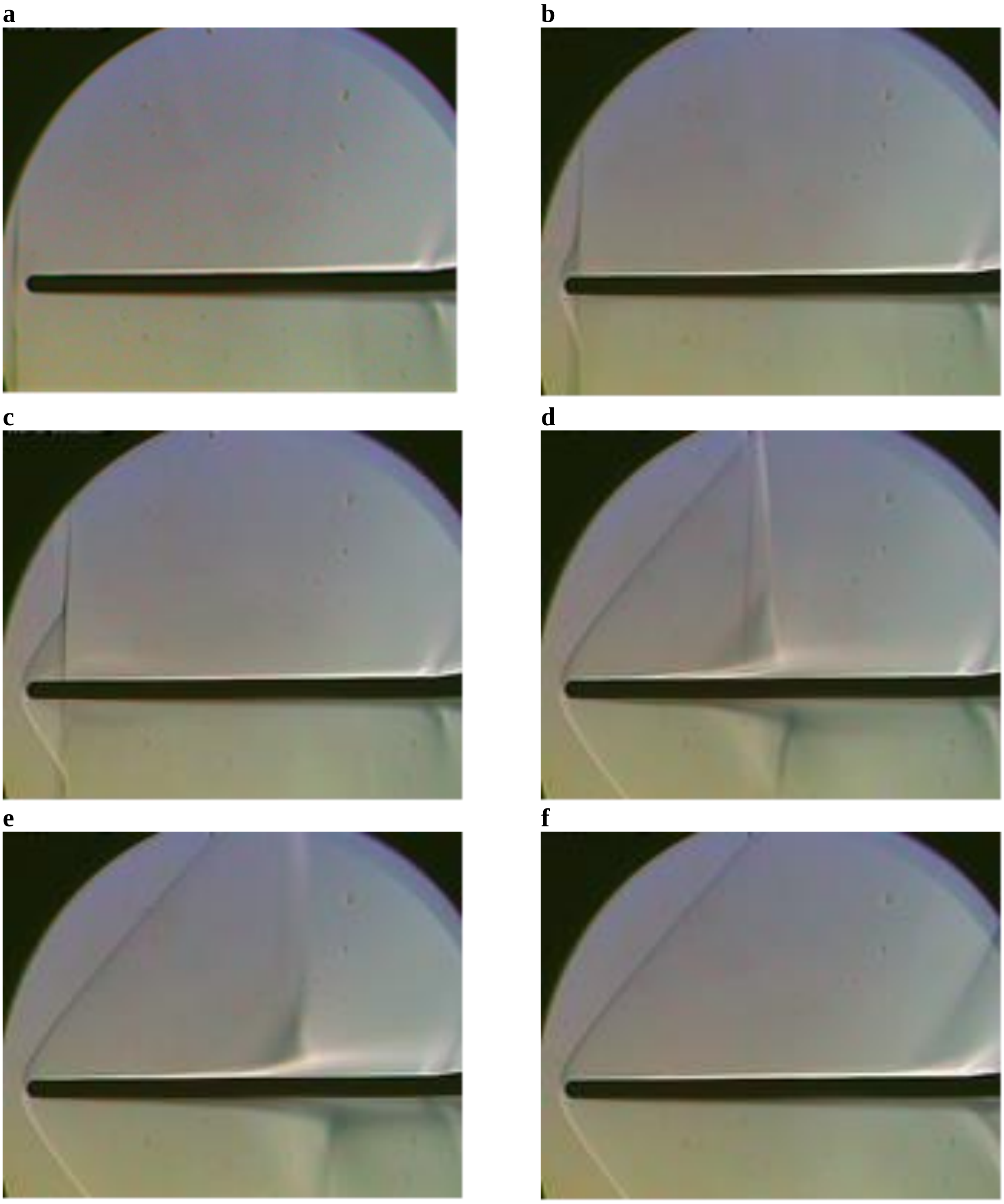
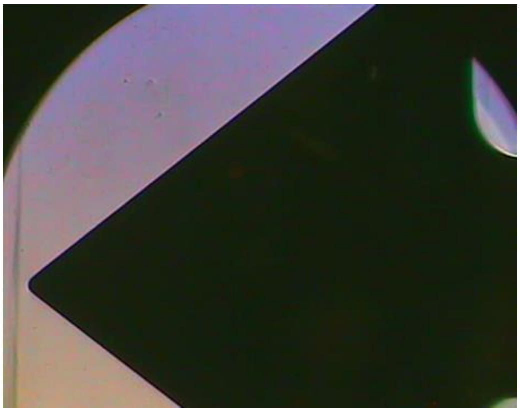


Fig. 3. Photos of shadow visualization illustrated dislocation of normal shock wave along the model in the test section of the wind tunnel with increasing of the total pressure, horizontal model installation.

a **b**




c **d**


e **f**

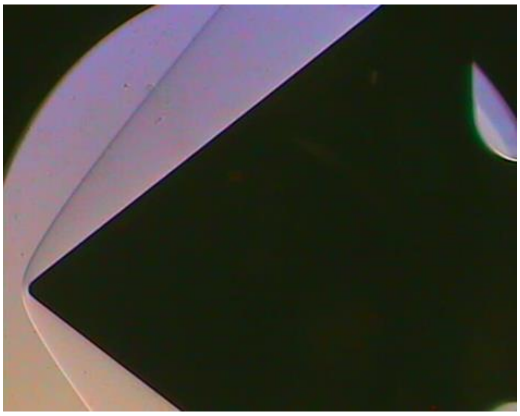
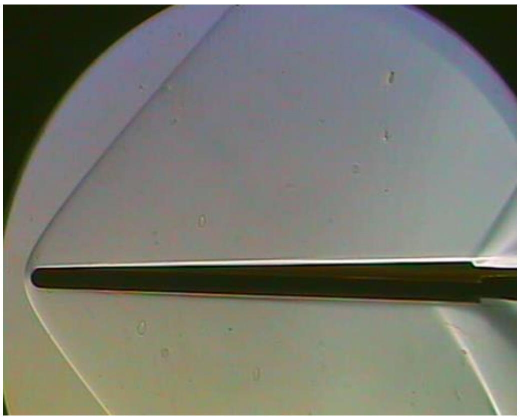
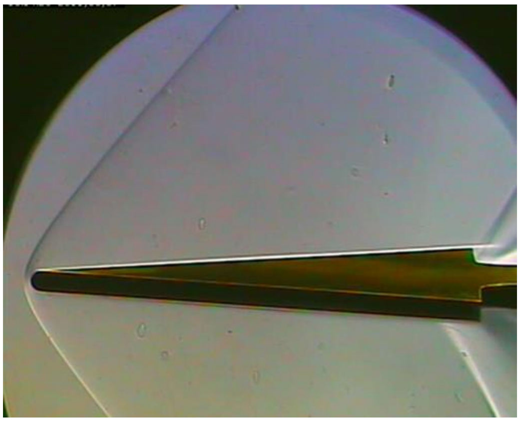



Fig. 4. Photos of shadow visualization illustrated dislocation of normal shock wave along the model in the test section of the wind tunnel with increasing of total pressure, vertical model installation.

a **b**

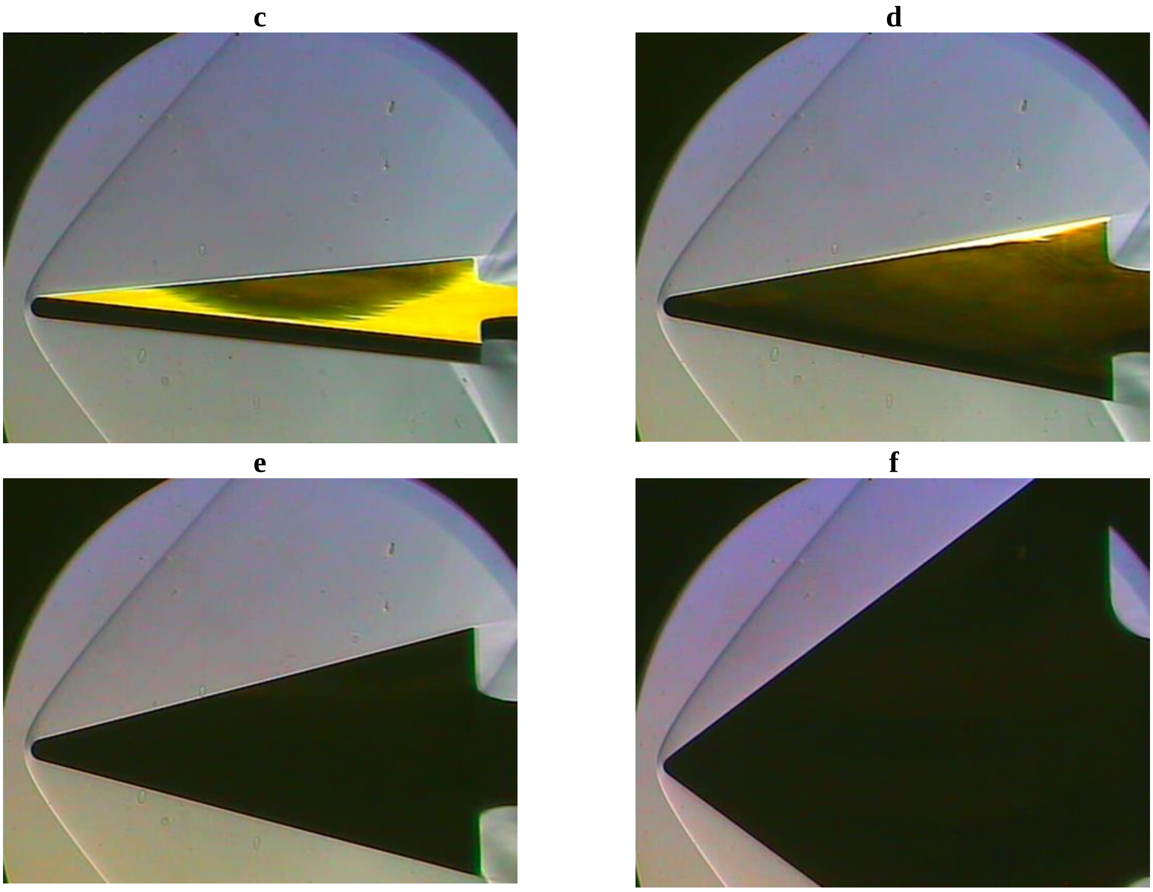
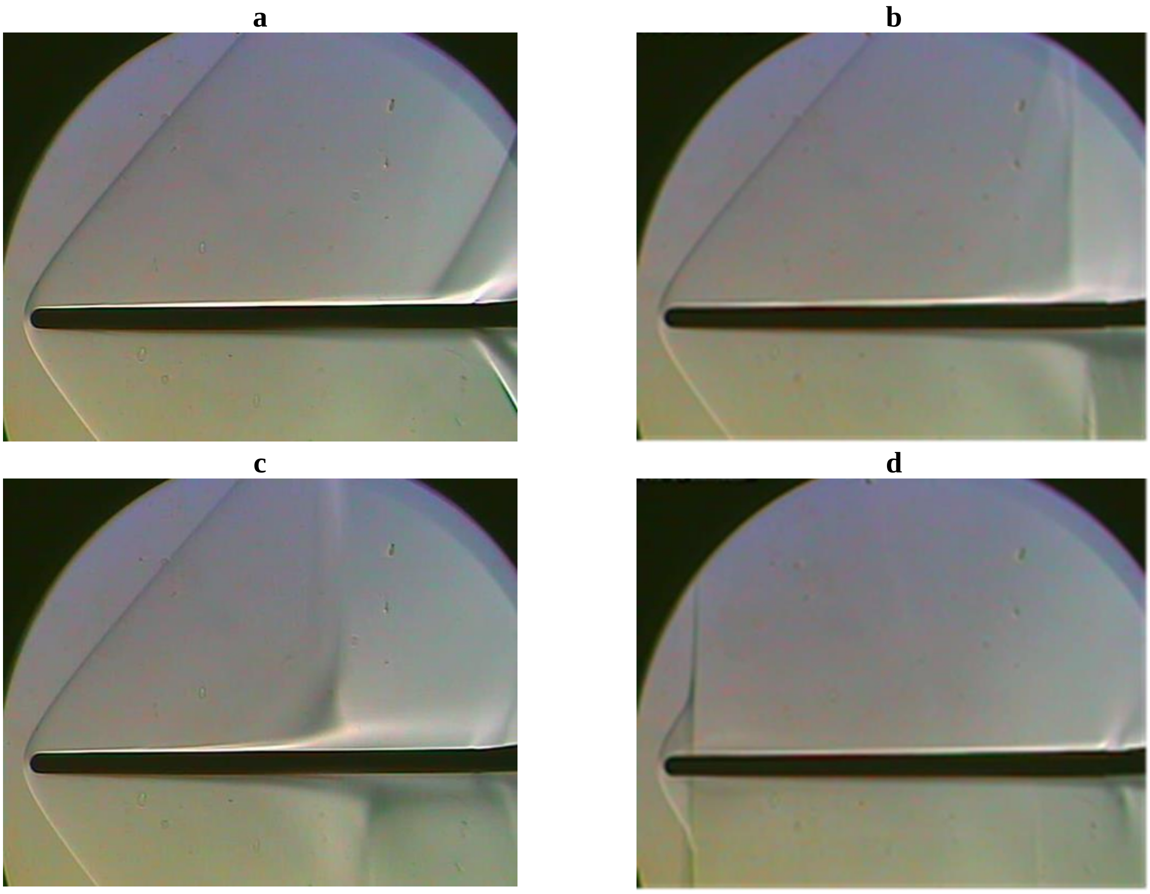


Fig. 5. Examples of shadow visualization of rotated model



e

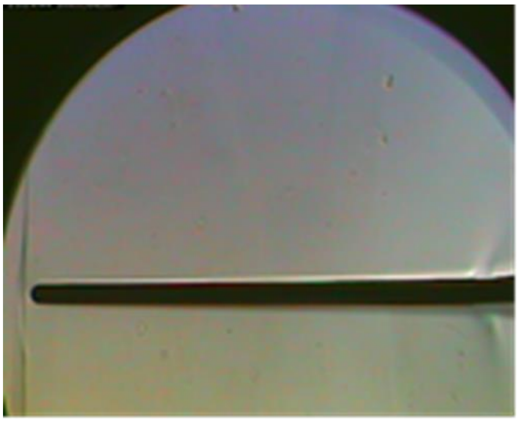


Fig. 6. Photos of shadow visualization illustrated dislocation of normal shock wave along the model in the test section of the wind tunnel with reducing of total pressure.