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SHOCK WAVE – BOUNDARY LAYER INTERACTION CONTROL BY STREAMWISE VORTICES

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Summary Control of shock wave and boundary layer interaction finds still a lot of attention. Methods of this interaction control have been especially investigated in a recent decade. This research was mostly concerned with flows without separation. However, in many applications shock waves induce separation which often leads to strong unsteady effects. In this context it is proposed to use streamwise vortices for the interaction control and the results of experiments are presented here.

Normal shock wave – boundary layer interaction (SWBLI) is still an important problem in compressible aerodynamics. It demands further investigation of the flow structure in the interaction area, but already large effort has been invested in the interaction control. It was usually considered for low Mach numbers below the range of incipient separation induced by a shock wave.

During the last decade many methods of the SWBLI control have been investigated. These concerned a local wall modification in a form of a bump and many applications of the transpiration flow through walls. Passive control, active control and mixed hybrid type of flow control by suction and blowing have been extensively studied in many research centres. These methods although not very effective, showed that new technologies are seriously considered and that application of perforated walls, slots and holes for suction or blowing is not a technological problem nowadays.

SWBLI leads to separation at Mach numbers above M = 1.3. The process of separation usually becomes unsteady and causes buffeting at airfoils and shock oscillations in the internal flows. In such cases it is important to use control methods which are able to suppress or reduce separation and to diminish the unsteady effects. Reduction of shock movement is a very important aspect, which results in dumping of the vibrations induced by unsteady forces.

It has been proposed to investigate the potential of the stream-wise vortices for the control of SWBLI. Already in 70-s it has been proven that streamwise vortices have a significant effect on the shock wave – boundary layer interaction, but so far, research has been primarily focused on the fixed vortex generators (vain type) and they predominate in the airplane technology applied today. However, it was shown that air-jet vortex generators (AJVG) are also very effective in the production of streamwise vortices. This type of vortex generators was somehow forgotten. But in the 90-s vortex generation by air jets was looked into again, especially in the aspect of the intensity of a generated vortex, at subsonic speeds.

The most useful method of the stream-wise vortex generation is by air jets because it allows a simple method of switching it on and off. The jets have to be given appropriate angle of injection in relation to the main stream. This inclination affects the intensity of the generated vortices and should be analysed for its optimisation.

This paper presents our first results of the application of this new SWBLI control technique which allows displaying very clearly the advantages of the AJVG's in the shock wave boundary layer control.

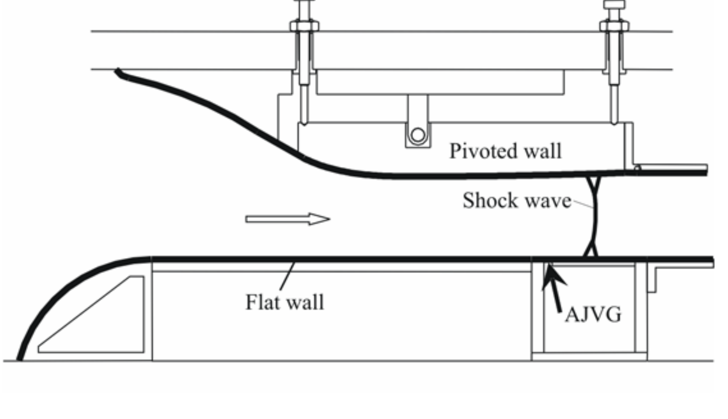

Fig.1. Test section lay-out

An experimental program has been carried out in the transonic wind tunnel of IMP PAN. In Fig.1. the test section sketch is presented. A normal shock wave was located in an appropriate place in a divergent part of the nozzle and an example is presented in Fig.2. One wall of the nozzle is flat and the interaction is studied at this wall. An opposite wall has a slight convex shape and its inclination may be varied. Thanks to this, shock waves of different intensity may be located at the same place in the nozzle. This is an important feature making measurements easier and allows maintaining the same distance of the interaction place from the AJVGs for different Mach number cases. Chosen Mach numbers are M = 1.25, 1.35, 1.45 and 1.55 which cover the range from unseparated to strongly separated flows.

One important feature of AJVG is that the intensity of generated vortices is only weakly dependent on the jet momentum. This is because the vortex is formed in the lower part of the boundary layer in the area of large gradient of velocity. High jet momentum, penetrating into the main stream, does not intensify the vortex but only increases the disturbance. For this reason we decided that the stagnation pressure of air jets is in our investigations equal to the stagnation pressure of the main stream. With this assumption our results will be applicable for all cases without any driving system for jets. Air for jets may be supplied from any leading edge or other stagnation air drainage in all studied cases. This approach enhances the applicability of our AJVG considerably.

AJVG in the test section are located in one row across the whole width of the test section. The jets are located in a distance of 9 × δ (boundary layer thicknesses) upstream of the interaction location beginning (minimum of the static pressure at the wall). The measurements consist of:

Fig.2. Shock wave and AJVG in operation, M=1.45

- static pressure distribution along the centre line of the test wall,
- boundary layer velocity profile 5 × δ upstream of the interaction area
- boundary layer velocity profile in two locations 6 × δ and 10 × δ downstream of the interaction area
- schlieren visualization of the shock wave structure, position and oscillation
- oil flow visualization at the test wall to display separation size and structure
- measurement of the hot-film sensor indications, placed in the shock vicinity to display shock's oscillations
- high speed movie visualization of the shock oscillation by a line CCD camera.

In Fig.2 flow is from the left to the right side. The oblique wave system originating at the wall is caused by the operating air jets. Further to the right side a shock system forming a λ-foot is present. Boundary layer is visible as a light zone adjacent to the wall. Shock wave causes significant increase of the boundary layer thickness.

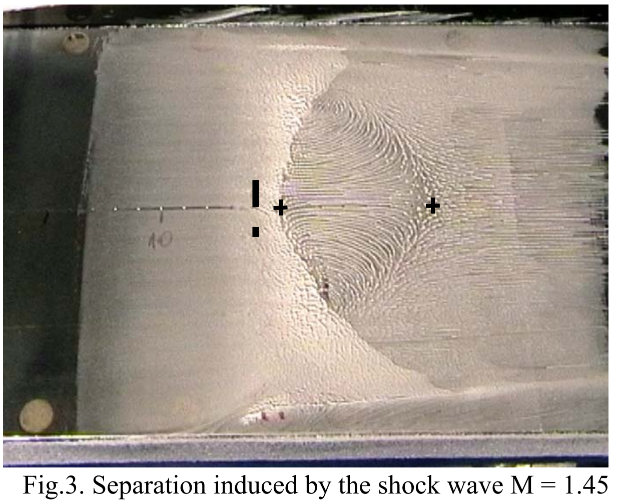

Fig.3. Separation induced by the shock wave M = 1.45

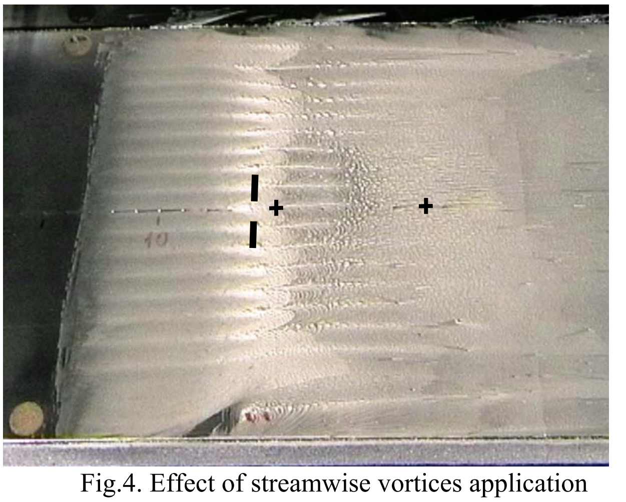

Fig.4. Effect of streamwise vortices application

The effect of streamwise vortices application is presented by Fig.3 and 4. They show surface oil flow visualisation on a flat wall in the area of SWBLI. The flow is from left to right. In Fig.3 undisturbed interaction is taking place. In the middle of the flat wall a well defined separation area is displayed by the oil streaks indicating a reverse flow area. The beginning and the end of separation are marked by small crosses. Shock location, upstream of the separation area is marked by two short lines drawn transverse to the flow direction. This is indicated by the oil accumulation. In Fig.4 the case with AJVG is presented at the same Mach number case. In the shock upstream part the traces of vortices are visible. Streamwise accumulation of oil indicates separatrices located between neighboring vortices. Two lines and two crosses are copied from Fig.3. Along the mentioned separatrices the disturbance reach far more upstream than in Fig.3. It means that it should have some effect on the shock structure causing an earlier compression. This should reduce shock losses. Another feature is that the separation line becomes a repeating span-wise 3-D structure with pitch corresponding to AJVG spacing. Simultaneously the separation looks more straight in span-wise sense.

Stream-wise vortices penetrate the separation area but are terminated at the reattachment line in the middle of the test wall. Reattachment takes place much earlier thanks to proposed flow control method.

Appearance of separation is usually connected to the induction of unsteady behavior of the interaction area. First of all this concerns instability of the separation zone caused by the vortex shedding and other 3-D vortex structure variation. Coupled with this is unsteady behavior of the shock wave. Oscillations of the shock wave are significantly reduced by the application of the AJVG. Its effect on interaction unsteadiness was measured in three locations. One is the oblique shock formation in the boundary layer which showed that the separation point fluctuations are not large but are reduced by our control method. Another location is the rear shock close to the separated zone. In this location the unsteadiness reduction is the strongest. And the last one is the unsteadiness of the main shock wave above the triple point. In this location the oscillations bear characteristics of the channel flow and are very weakly affected by our flow control means.