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EXPERIMENTAL RESEARCH OF ADIABATIC WALL TEMPERATURE INFLUENCED BY SEPARATED SUPERSONIC FLOW

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ABSTRACT

Experimental results are given for separated supersonic flow influence on adiabatic wall temperatureof the plane surface. Separated flow was generated by a falling shock wave in the 1st test and by means of a rib in front of the plane model in the 2nd test. A steel wedge with 22 degrees opening angle was used as a shock wave generator. Studied rib heights was from 2 to 8 mm. Thickness of dynamic boundary layer was about 6 mm. Studied flow Mach numbers was in the range 2÷3. Reynolds number based on the distance from the nozzle throat was over $6 \cdot 10^6$, which corresponds to turbulent flow operation regime. Experiments were conducted in supersonic wind tunnel till ascertainment of thermal balance with the use of National Instruments equipment, LabView powered automation programs, optical visualization and non-contact IR methods of temperature field capture. Field distribution of pressure, adiabatic wall temperature, Mach numbers and temperature recovery factors are presented along the experimental model. Adiabatic wall temperature and recovery factor go through local maximum (about 2% growth) in the region of shock wave boundary layer interaction. Adiabatic wall temperature in separation region is lower down to 3.5% than that for the flow around plane surface. Maximum decreasing of temperature recovery factor is over 10% in separated flow region. The research urgency is caused by the fundamental determination of heat flux in supersonic flows and by the analysis of heat transfer enhancement in supersonic channel of gas dynamic energy separation device (Leontiev tube).

KEY WORDS: Convection, Turbulent transport, Heat transfer enhancement, Shock wave, Separated flow, Gas dynamic energy separation.

1. INTRODUCTION

Separated flow is of great interest among researchers due to its highly widespread technical applications. Separation accompanies flows in air-gas channels of heat exchangers, power machines and engines. Heat transfer characteristics calculation in separated flows is necessary because of heat flux peaks in the separation region. This effect is to be suppressed in case of thermal protection design or it can be used in order to locally intensify heat transfer [1-9].

The shock wave boundary layer interaction may cause flow separation. Research in this field has been run for more than 60 years [10-15]. Nevertheless many problems are still urgent, some of which were mentioned in the review by Dolling D.S. [10]. Among them there are the location and magnitude of peak heating. According to Knight and Degrez [11] "to continue making progress in turbulence modelling for shock wave/boundary layer interaction it is essential to obtain accurate experimental data for flowfield turbulent heat flux, and wall pressure and heat transfer fluctuations".

Heat flux into the wall streamlined by supersonic flow (1) is proportional to difference between adiabatic wall temperature T_{aw} (2) and actual wall temperature T_w [16-18].

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$$q_w = h \cdot (T_{aw} - T_w) \tag{1}$$

Adiabatic wall temperature T_{aw} is defined as a function of recovery factor r (3), which indicates the dissipation of the flow kinetic energy. Numerous experimental researches for air flows [16-18] show that for a turbulent boundary layer in supersonic flow around the plate recovery factor r lies in the range 0.875…0.9. Theoretical relation for recovery factor in turbulent conditions is: $\boldsymbol{r} = \sqrt[3]{\Pr}$. This relation was experimentally proven for air in and around Pr=0.7. At the same time some researchers consider influence of recovery factor from Reynolds, Mach and Prandtl numbers, injection/suction into/from the boundary layer, shape and relief of streamlined surface [16-28]. These results show that recovery factor can significantly differ from the theoretical meaning for a plate.

$$T_{aw} = T \cdot \left(1 + r \cdot \frac{\gamma - 1}{2} \cdot \mathrm{M}^2\right) \tag{2}$$

and

$$r = \frac{T_{aw} - T}{T_0 - T} \tag{3}$$

In order to gain heat transfer coefficient authors sometimes replace adiabatic wall temperature with either its theoretical meaning for the case of smooth plate or stagnation free-stream temperature for complex separated flows [13-15]. In the 1st case recovery factor is assumed invariable, in the 2nd – adiabatic wall temperature is at all eliminated from consideration. In any case there would be considerable inaccuracy while defining heat transfer coefficient.

The problem of adiabatic wall temperature definition is also urgent due to opportunities research for higher efficiency of gas dynamic energy separation device [29-40]. Its working principle is based on interaction through the heat-conducting wall of supersonic and subsonic flows obtaining the same input stagnation temperature (Fig. 1a). In result of such interaction at output there are 2 flows with different temperature – heated supersonic air flow and cooled subsonic one.

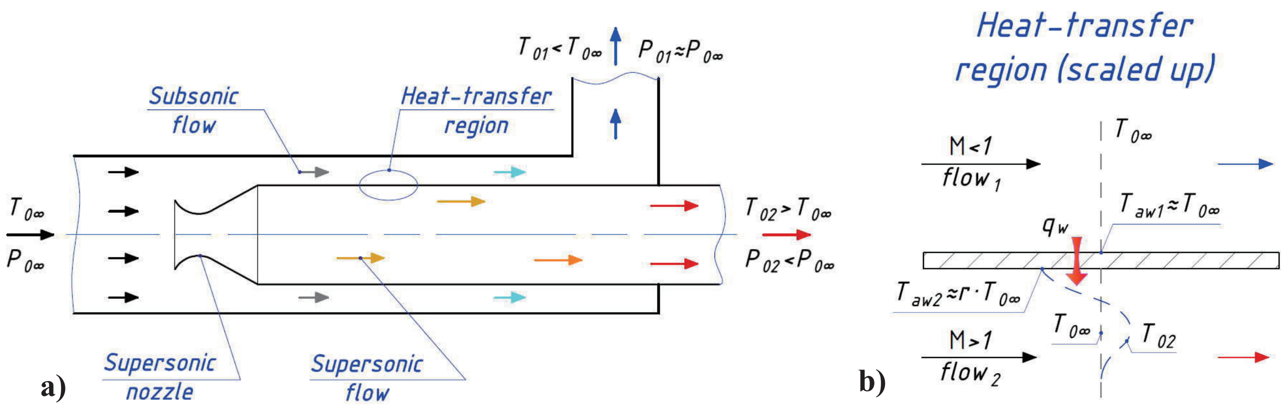


Fig. 1 Gas dynamic energy separation device operation principle

Heat flux through the division wall between 2 flows in gas dynamic energy separation device (Fig. 1b) is defined with stagnation temperature at input and adiabatic wall temperature at supersonic side of the device:

$$q_{w\_ES} = \frac{1}{\frac{1}{h_1} + \frac{\delta}{k} + \frac{1}{h_2}} \cdot (T_{aw} - T_0) \tag{4}$$

Here: T_{aw} – adiabatic wall temperature at supersonic side of the device, T_0 – stagnation temperature at subsonic side of the device (equal to the device input stagnation temperature), h_1 and h_2 are heat-transfer coefficients for cold (subsonic) and hot (supersonic) heat medium, δ is a thickness of the division wall, and k is a thermal-conductivity coefficient of the wall material.

For complete utilization of energy separation effect heat transfer should take place on a rather long distance – dozens of the tube calibers [32, 37-39]. In order to increase the heat transfer rate it is necessary to rise up the less of the two heat-transfer coefficients (4). In this case it is the one at the supersonic side of energy separation device. Turbulent boundary layer separation region cause heat transfer coefficient to get increased manifold. At the same time temperature differential in (4) is also important for energy separation process. The hypothesis of the research was that placement of some obstacle like a rib at the beginning of supersonic channel will generate a system of shock waves in the channel. This will intensify heat-transfer in energy-separation process resulting in full utilization of this physical effect on reduced length.

The goal of the work is experimental research of adiabatic wall temperature influenced by 2 factors: shock wave boundary layer interaction and separated supersonic flow behind a rib. Urgency of the problem is provided with fundamental importance of heat flux definition in supersonic flows around plane surface and for complex flows with boundary layer separation. Also the results are necessary for gas dynamic energy separation process research.

2. EXPERIMENTAL APPARATUS, INSTRUMENTATION AND TECHNIQUE

Experiments have been conducted in supersonic wind tunnel without time limitations (Fig. 2), i.e. it was hold enough till ascertainment of thermal balance. Flow velocity in the working area was changed with the use of plane variable supersonic nozzle in range of Mach numbers 2.0÷3.0. Stagnation pressure was 5÷7 atm. Stagnation temperature changed from 10 to 25°C. Maximum gas flow rate was about 10 kg/sec.

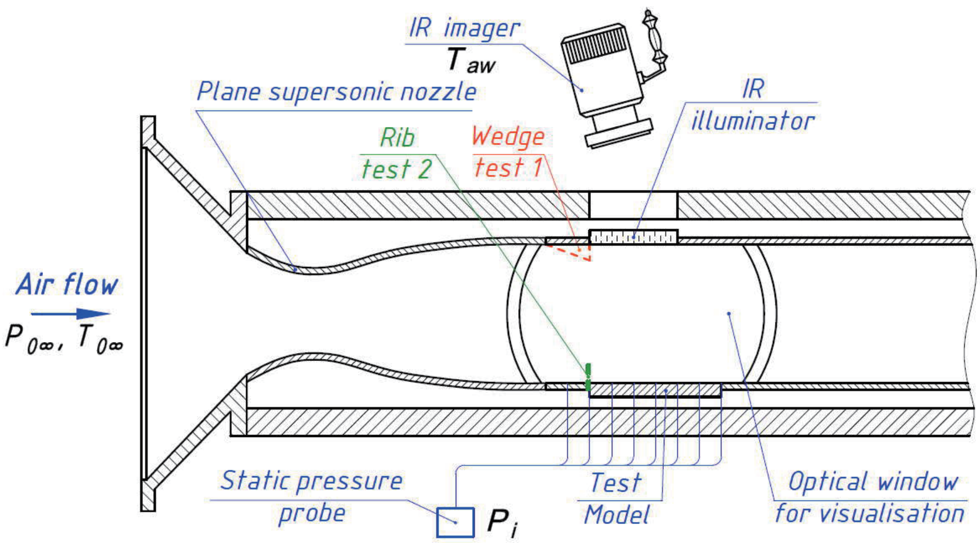


Fig. 2 Experimental apparatus and instrumentation schematic model

Working area has a rectangular cross-section 70 mm width and 90 mm height. For the purpose of experimental observation optical glass windows were installed on side walls of the working area (Fig. 3a). Flow visualization was made with schlieren photography. The upper wall of the wind tunnel – infrared illuminator – was made of KCl for the 1st test (shock wave – boundary layer interaction) and ZnSe for the 2nd test (separated flow). These materials are transparent for infra-red radiation and provide the opportunity of noncontact thermal imaging. Temperature field of the experimental model was recorded by thermal imager FLIR ThermaCam SC3000 through infrared illuminator (Fig. 3b).

As an experimental model a plate made of acrylic resin was taken. This material has a low thermal conductivity coefficient k = 0.19 W/(m·K) thus providing an assumption of heat insulated surface. The model was mounted on the lower side of the working area in parallel with the main flow (Fig. 2). Width of the model fitted the working area – 70 mm, its length was 180 mm. A steel wedge with 22 degrees opening angle was used as a shock wave generator. It was mounted on the upper wall of the working area in front of the experimental model. The shock wave falling region was preliminary defined subject to deflection angle of the air flow [8, 13].

a) b)

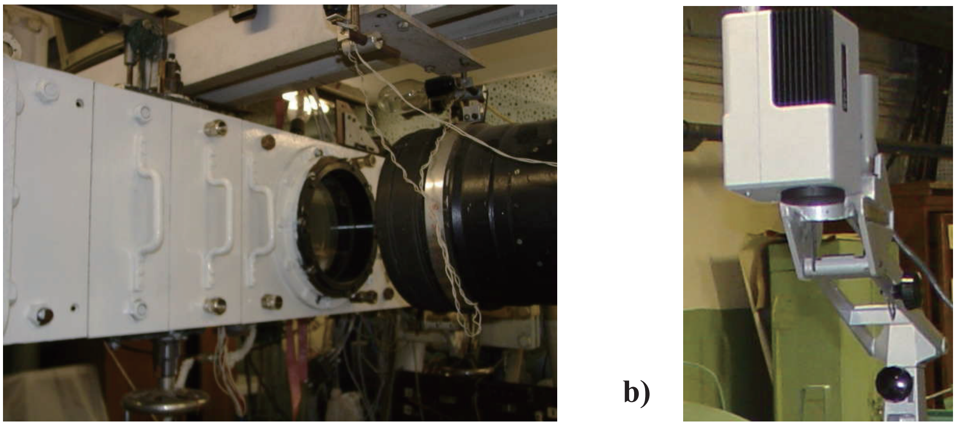


Fig. 3 Photo of supersonic wind tunnel with optical window for visualization and schlieren instrument and infra-red thermal imager

In order to measure static pressure experimental model was drained along the central line. Through this perforation air bleed was taken to the static pressure probe. Stagnation parameters were measured in settling chamber of the wind tunnel: temperature – with the use of 2 insulated chromel-koeppel thermocouples, static pressure – with the use of probes Honeywell ML-300PS2PC. Automation program was made in LabVIEW. Flow parameters are calculated in 11 points according to static pressure draining along experimental model including temperature recovery factor (5) and local Mach number (6). Local Mach number downstream the shock wave is defined according to stagnation pressure losses [8, 13]. Shock wave inclination angle was registered with the use of schlieren photography. Emissivity of the experimental model was 0.96, as the wall was painted black in advance. Infra-red illuminatior transmission factor was preliminarily defined as 0.91 for the 1st test (KCl as illuminator material) and 0.71 for the 2nd test (ZnSe).

$$r = 1 - \frac{T_0 - T_{aw}}{T_0} \cdot \left(\frac{2}{(\gamma - 1) \cdot \mathrm{M}^2} + 1 \right) \tag{5}$$

and

$$\mathrm{M} = \sqrt{\frac{2}{\gamma - 1} \cdot \left(\left(\frac{P_0}{P_s} \right)^{\frac{\gamma - 1}{\gamma}} - 1 \right)} \tag{6}$$

Temperature recovery factor is calculated by 2 methods with references: “loc” – local parameters calculation in the flow; “in” – inlet flow parameters according to Mach number upstream the experimental model. Reynolds number based on the distance from the nozzle throat was over $6 \cdot 10^6$, which corresponds to the turbulent flow operation regime.

The uncertainty of the temperature recovery factor was assumed as 3%. It depends on the uncertainties in stagnation temperature, adiabatic wall temperature and Mach number. The uncertainty of absolute adiabatic wall temperature is about 2%. The uncertainties of stagnation and static pressure are about 1.5%. The uncertainty in the local Mach number is ±1.0%.

3. RESULTS AND DISCUSSION

Correctness of measuring equipment was tested before the basic research. For this purpose temperature recovery factor was estimated for the flow around plane model without any disturbances. The results of test experiments for recovery factor are given in Fig. 6, 10 and 14 in comparison with the results of the main research. Experimental meaning of recovery factor meets interval 0.875-0.9. This meaning corresponds to numerous experimental results by various researchers [13-14].

3.1 Shock Wave Boundary Layer Interaction After testing the main experiments were held. Artificially initiated shock wave was generated by steel wedge with 22 degrees opening angle that was placed right above experimental model (Fig. 2). A series of experiments were held in range of Mach numbers 2÷3. The results for M_0 = 2.25 are given in Fig. 4-7. Flow parameters: flow with a shock wave (reference "sh") – P_0^* = $5.8{\cdot}10^5$ Pa, T_0^* = 291 K, M_0 = 2.25; flow along plane model without disturbances (reference "pl") – P_0^* = $5.1{\cdot}10^5$ Pa, T_0^* = 292 K, M_0 = 2.25.

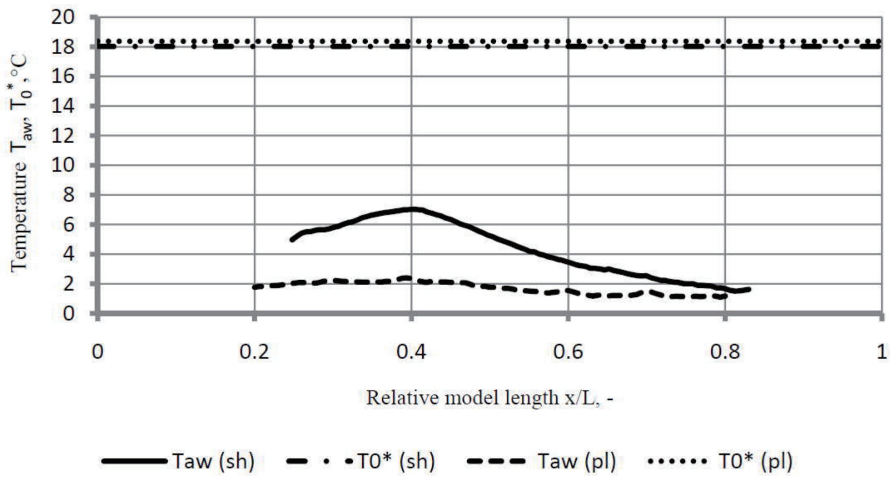


Fig. 4 Adiabatic wall temperature and stagnation temperature along dimensionless length of the experimental model for 2 test regimes: "sh" – flow with shock wave, "pl" – flow along plane model without disturbances (M_0=2.25)

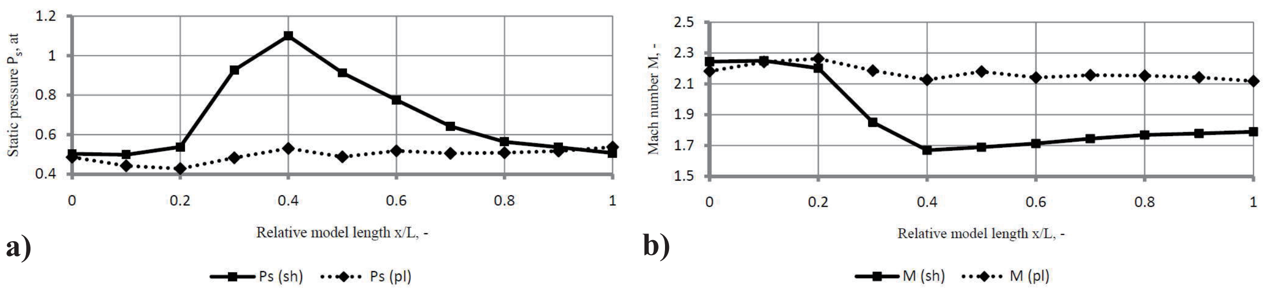
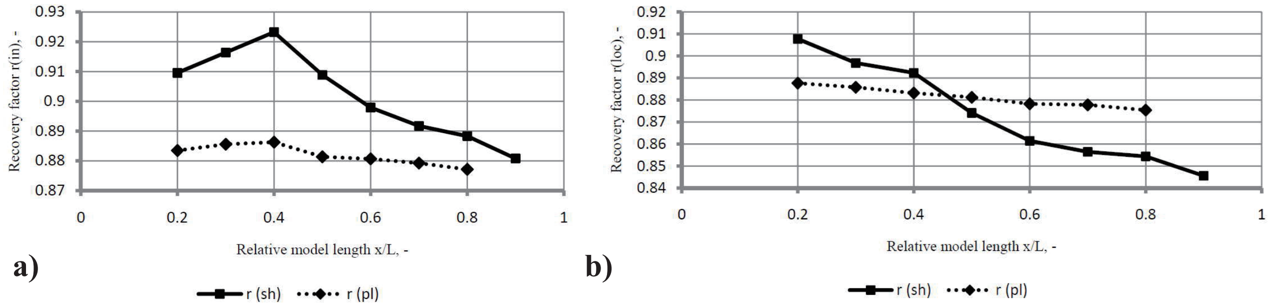
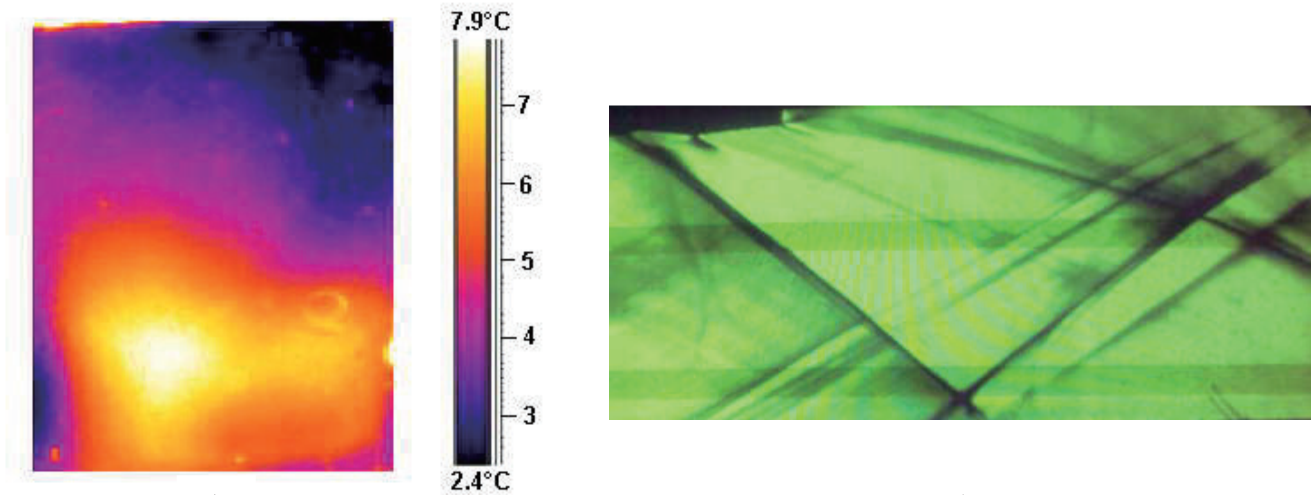


Fig. 5 Static pressure (a) and Mach number (b) along dimensionless length of the experimental model for 2 test regimes: "sh" – flow with shock wave, "pl" – flow along plane model without disturbances (M_0=2.25)



a) b)

Fig. 6 Temperature recovery factors based on the inlet (a) and local (b) parameters along dimensionless length of the experimental model for 2 test regimes: “sh” – flow with shock wave, “pl” – flow along plane model without disturbances (M_0=2.25)



a) b)

Fig. 7 Infra-red thermal imaging (a) and schlieren photography (b) for flow with a falling shock wave

As shown in Fig. 4, adiabatic wall temperature increases by 2% (in Kelvin) in a flow with a shock wave falling in comparison with a flow around a plane model without any disturbances. Static pressure goes through local maximum in the shock wave boundary layer region (Fig. 5a) and Mach number is therefore decreasing (Fig. 5b). Recovery factor is also higher in the shock wave boundary layer region than that on the plate without disturbances (Fig. 6). Opening angle of the shock wave generating wedge was chosen such a way that intensity of a falling shock wave (Fig. 5) was more than a critical pressure drop appropriate to the Mach number of the incoming flow [8, 13]. Hereupon in the shock wave falling region boundary layer separates with generation of eddy zone on the model surface. The main reason for a rise of adiabatic wall temperature and recovery factor is probably significant decrease of a flow velocity in the separation region.

Downstream the shock wave falling adiabatic wall temperature declines to that for the flow around plane model (Fig. 4). This comparison is made assuming equal inlet Mach number for 2 cases. Recovery factor decreasing in the rear part of the interaction region (Fig. 6) is about 3% lower than for typical meaning of a common flow around a plate. This decrease can be explained by the formation of a new boundary layer downstream the separation zone.

Comparison of the results for flow with a shock wave at M_0 = 3.0 and a plane model flow at M_0 = 2.0 are given in Fig. 8-10. Such comparison is useful because of flow velocity reducing after the shock wave. Thus we can analyze if the parameters of adiabatic wall change in compliance with the Mach number behavior. Flow parameters: flow with a shock wave (reference “sh”) – $P_0^* = 5.9 \cdot 10^5$ Pa, T_0^* = 291 K, M_0 = 3.0; flow along plane model without disturbances (reference “pl”) – $P_0^* = 5.0 \cdot 10^5$ Pa, T_0^* = 292 K, M_0 = 2.0.

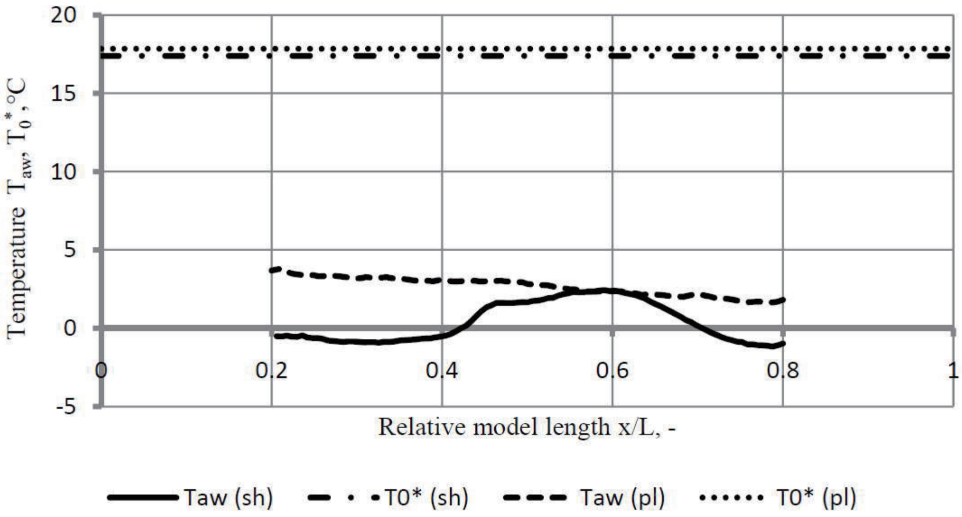


Fig. 8 Adiabatic wall temperature and stagnation temperature along dimensionless length of the experimental model for 2 working regimes: “sh” – flow with shock wave (M_0=3), “pl” – flow along plane model without disturbances (M_0=2)

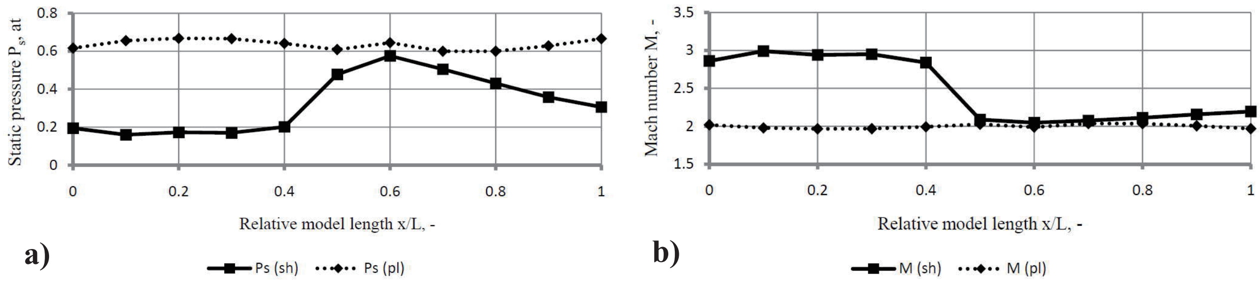


Fig. 9 Static pressure (a) and Mach number (b) along dimensionless length of the experimental model for 2 test regimes: “sh” – flow with shock wave (M_0=3), “pl” – flow along plane model without disturbances (M_0=2)

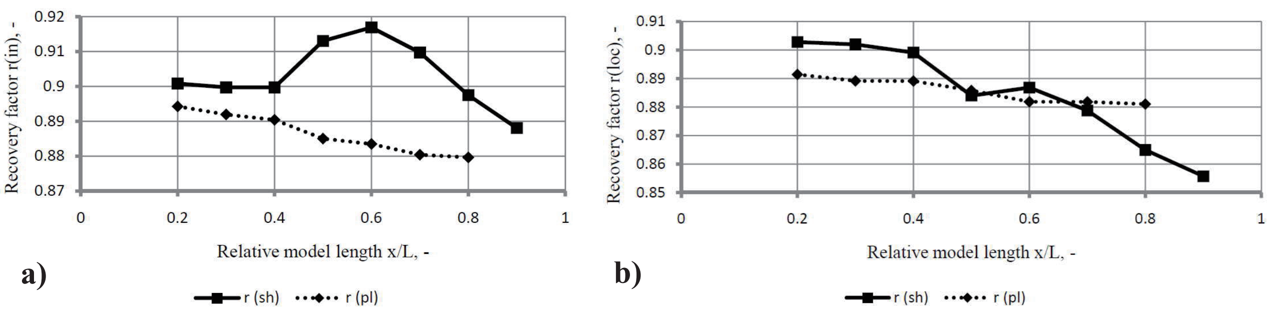


Fig. 10 Temperature recovery factors based on the inlet (a) and local (b) parameters along dimensionless length of the experimental model for 2 test regimes: “sh” – flow with shock wave (M_0=3), “pl” – flow along plane model without disturbances (M_0=2)

If we compare shock wave flow after interaction region with a plane model flow of the same local Mach number (Fig. 8-9) we can conclude that adiabatic wall temperature for the 1st flow in the separation region is almost equal to the adiabatic wall temperature for the 2nd flow. I.e. adiabatic wall temperature in the shock wave boundary layer region follows the change in the flow Mach number. But after the interaction region adiabatic wall temperature becomes lower than that for the plane surface. This result provoked the experimental research of separated flow behind the rib.

3.2 Separated Flow Behind The Rib At the 2nd stage of the research separated flow was generated by a rib placed in front of the experimental model (Fig. 2). Results are presented in Fig. 11-15. Adiabatic wall temperature for rib heights 2, 4, 6 and 8 mm is presented in Fig. 11 a-d. Boundary layer thickness for flow along plane model without disturbances was about 6 mm. Comparison of flows with separation region behind the rib (M_0=2.75) and flow around plane model (M_0=2.1) is given in Fig. 12-14. Such comparison, similar to the 1st test, is useful because of flow velocity reducing after the rib. Here each parameter from x/L=0 and further on corresponds to position behind the rib. Stagnation pressure losses and therefore Mach number reducing after the shock wave are defined with the schlieren photography (Fig. 15).

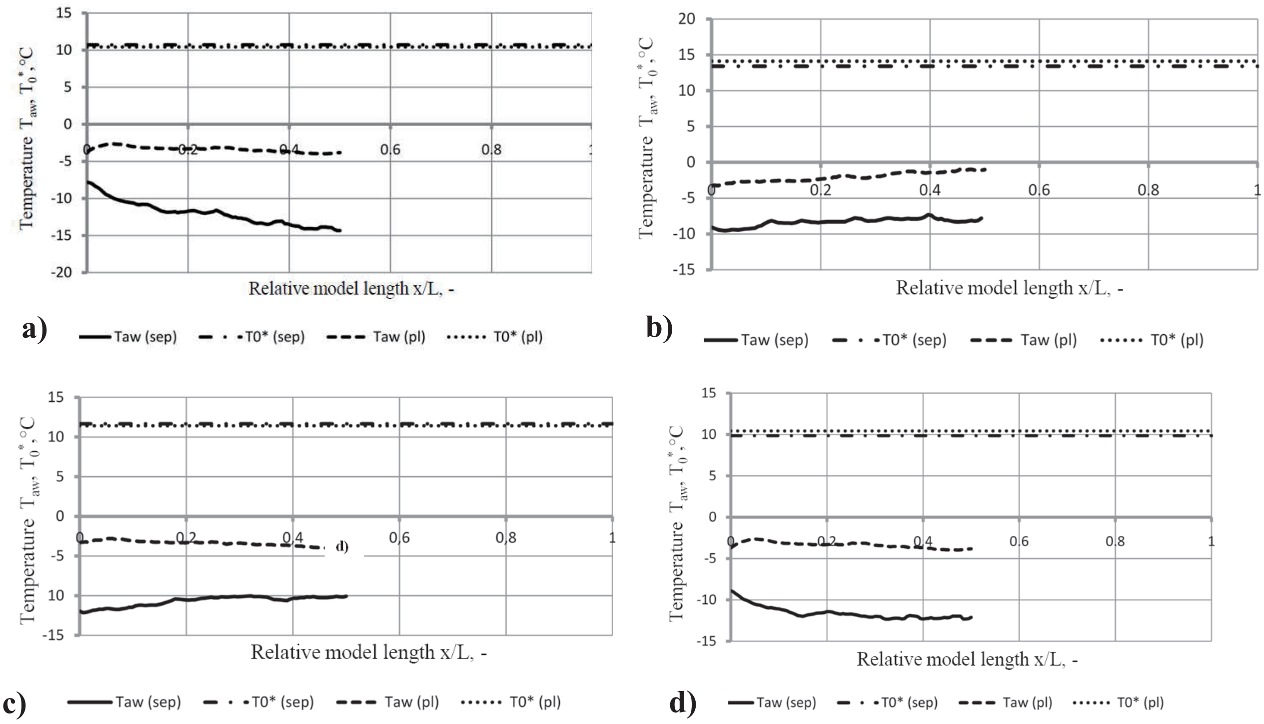


Fig. 11 Adiabatic wall temperature and stagnation temperature along dimensionless length of the experimental model for 4 heights of the rib: a) 2 mm; b) 4 mm; c) 6 mm; d) 8 mm; “sep” – separated flow behind the rib (M=2 after the rib), “pl” – flow along plane model without disturbances (M_0=2)

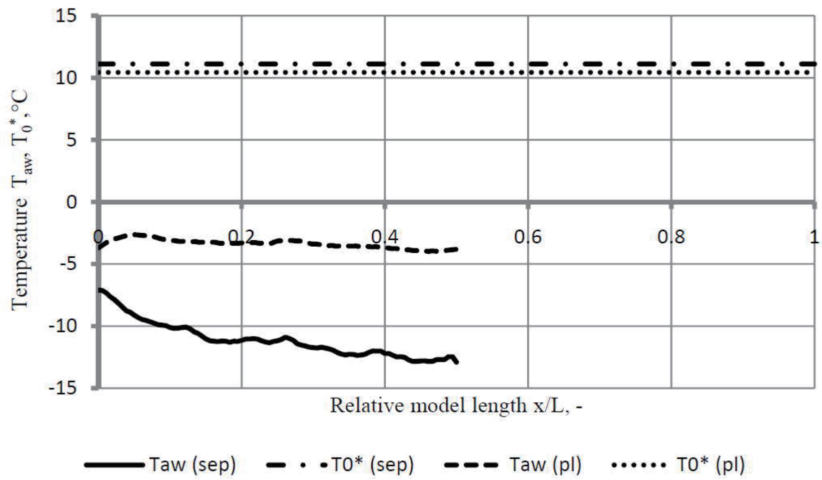


Fig. 12 Adiabatic wall temperature and stagnation temperature along dimensionless length of the experimental model for 2 working regimes: “sh” – flow with shock wave (M_0=2.75 before the rib), “pl” – flow along plane model without disturbances (M_0=2.1)

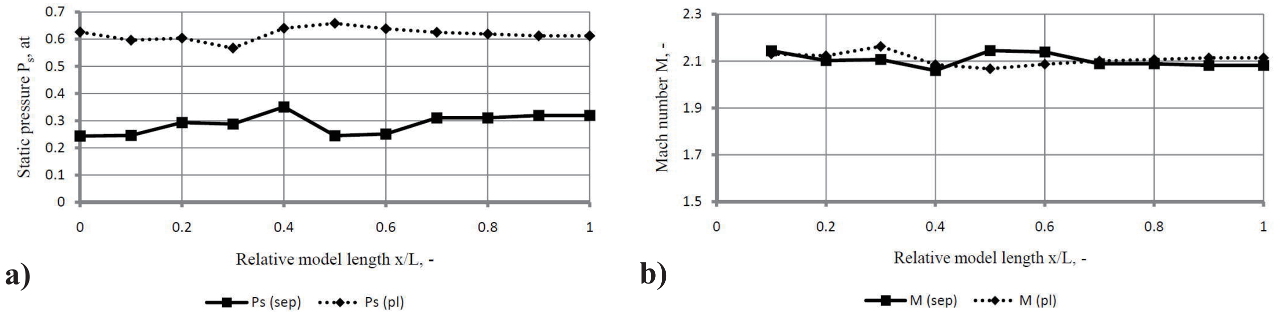


Fig. 13 Static pressure (a) and Mach number (b) along dimensionless length of the experimental model for 2 test regimes: "sh" – flow with shock wave (M_0=2.75 before the rib), "pl" – flow along plane model without disturbances (M_0=2.1)

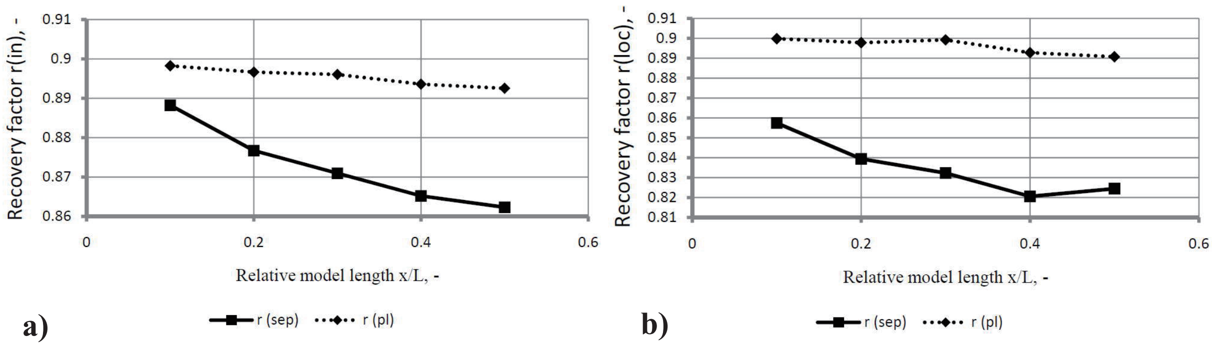


Fig. 14 Temperature recovery factors based on the inlet (a) and local (b) parameters along dimensionless length of the experimental model for 2 test regimes: "sh" – flow with shock wave (M_0=2.75), "pl" – flow along plane model without disturbances (M_0=2.1)

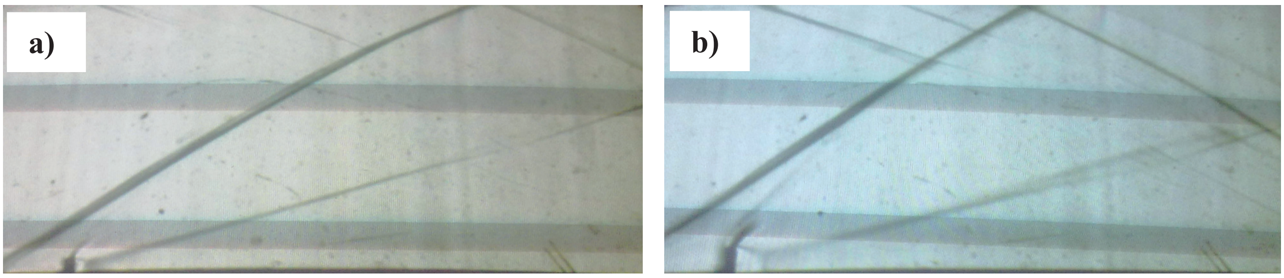


Fig. 15 **S**chlieren photography for separated flow behind the rib (M_0=2.75) with variable height: 4 mm (a) and 8 mm (b)

According to Fig. 11-12 adiabatic wall temperature in separation region is lower down to 3.5% (in Kelvin) than that for the flow around plane surface. Maximum decreasing of temperature recovery factor is over 10% in separated flow in comparison with the plane model (Fig. 14) for the whole range of explored Mach numbers.

Returning to the practical subject of the research – gas dynamic energy separation process – we can conclude that in the shock wave boundary layer interaction region temperature differential ($T_{aw} - T_0$) is lowering (Fig. 4). Downstream the shock wave falling and in the separation region after the rib this meaning increases (Fig. 8, 11). At the same time heat transfer coefficient (Stanton number) rises significantly in the separation region [8, 13, 14]. According to (4) and (5) separated flow can be used as a practical way of heat transfer enhancement in gas dynamic energy separation device. Placement of a shock wave generator on input of the tube or a series of ribs along the supersonic channel will probably intensify heat transfer and make the device shorter for the same temperature differential between subsonic and supersonic flows at output of the device.

4. CONCLUSIONS

Experimental results are presented for boundary layer separation influence on adiabatic wall temperature in supersonic flow around plane surface. Separated flow was generated by 2 ways: shock wave boundary layer interaction and by means of a rib in front of the plane model. Shock wave boundary layer interaction resulted in local increasing of adiabatic wall temperature and recovery factor by about 2%. Adiabatic wall temperature in separation region behind the rib is lower down to 3.5% than that for the flow around plane surface. Maximum decreasing of temperature recovery factor is over 10% in separated flow region behind the rib in comparison with common flow around plane model of the same local Mach number. Results indicate that local separated zones will probably intensify heat transfer in supersonic channel of gas-dynamic energy separation device.

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NOMENCLATURE

M	free-stream Mach number	(-)
T	temperature	(K)
P	pressure	(Pa)
q	heat flux	(W/m^2)
r	recovery factor	(-)
x	coordinate	(m)
L	experimental model length	(m)
Re_x	Reynolds number $\mathrm{Re}_x = \dfrac{U \cdot x \cdot \rho}{\mu}$	(-)
q_w	heat flux into the wall	(W/m^2)
δ	wall thickness	(m)
k	thermal conductivity	(W/m·K)
h	heat transfer coefficient	(W/m^{2}K)

Subscripts

0	stagnation parameter
∞	free-stream
aw	adiabatic wall
1	subsonic
2	supersonic
i	number of static pressure draining on the experimental model
ES	energy separation
(sh)	flow with shock wave
(sep)	separation flow behind the rib
(pl)	flow around plane model without disturbances
(loc)	calculation in local parameters
(in)	calculation in inlet parameters

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