Experimental research of heat transfer in supersonic separated compressible gas flow

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Abstract. The results of an experimental study of heat transfer for supersonic flow around plane surface in the wake of a rib are presented. The study was conducted on unsteady regime during the launching supersonic wind tunnel before reaching the equilibrium thermal state. The initial flow Mach number was 2.2, Reynolds number based on the length of the dynamic boundary layer from the nozzle throat was over 20 million at the nozzle exit section. The rib height was varied from 2 to 10 mm while boundary layer thickness for smooth model flow in the region of rib placement was about 6 mm. Recovery temperature and the coefficient of heat transfer enhancement for flow past the rib are presented in comparison with the regime of a smooth model flow. The research was carried out with the use of thermocouples with thermal compensation, total and static pressure probes, LabView automation programs.

1. Introduction

A specific feature of the supersonic flow around a body is the occurrence of significant aerodynamic heating due to a larger temperature gradient in the boundary layer in comparison with subsonic velocities. The temperature gradient arises due to the transition of the kinetic energy into heat under the influence of viscous friction \cite{1}.

Data on heat transfer are usually presented in the form of heat transfer coefficient (1), which is generally determined by the ratio of the specific heat flux into the wall to the temperature difference between the instantaneous temperature of the streamlined wall and some defining temperature in the flow \cite{2}:

\[
h = \frac{q_w}{T_{w\text{def}} - T_w}
\]  

(1)

The determining temperature can be the thermodynamic temperature in the flow, the total temperature, the initial temperature (at the entrance to the channel with sources or sinks of heat) or the adiabatic wall temperature. However, as shown in a number of works \cite{2–4}, only the use of the adiabatic wall temperature guarantees the accuracy of the transfer of data obtained on wind tunnels to the actual operating conditions of power machines and engines.

The ratio of the intensity of heat transfer to the full potential heat transfer of the gas flow is determined by the dimensionless heat-transfer criterion — the Stanton number \cite{5}. The heat exchange
data, presented as the Stanton number, are less sensitive to changes in the Mach number of the flow (as well as changes in the wall temperature) in comparison with the heat transfer coefficient \( h \):

\[
St = \frac{q_w}{\rho_w u_w C_p (T_{aw} - T_w)}
\]  

(2)

When calculating the Stanton number from the relation (2), one of the main problems is the determination of the adiabatic wall temperature. In the practice of engineering and scientific calculations, this temperature is determined by the temperature recovery coefficient \( r \), the total temperature in the stream \( T_0 \) and the Mach number \( M \):

\[
T_{aw} = T_0 \cdot \frac{1 + r \frac{k-1}{2} M^2}{1 + \frac{k-1}{2} M^2}
\]  

(3)

Numerous experimental studies carried out for air [5, 6] have shown that for a developed turbulent flow regime in a supersonic flow around a plate or bodies of revolution the temperature recovery factor value is in the range 0.885 ± 0.01. At the same time, the effect of reducing of the recovery factor and the corresponding decrease in the adiabatic wall temperature down to values even lower than the thermodynamic temperature for the subsonic flow across the cylinder (Eckert-Weise effect) is known [7, 8]. In [9], it was demonstrated that the effect of adiabatic wall temperature reducing propagates downstream in the wake of the cylinder. It was suggested that the vortex path formed by a streamlined body is the cause of this effect.

For supersonic flows, a decrease in the adiabatic temperature is recorded for a flow around a cylindrical wall follows a circular rib, on a flat wall behind a rib or a step, on a conical surface behind various heads in the form of a sphere, cylinder, cone and disk [10-12]. The distribution of the total temperature in the flow with cold regions in the central region and hot ones on the periphery of the wake is also observed when the flow in the channel follows the cut of the turbine blade [13].

The aim of this work is to study the effect of local reduction of the adiabatic wall temperature in the wake of a rib placed in front of the plane model streamlined by the supersonic flow. The task is caused by the need to increase the accuracy of extrapolation of experimental data on heat transfer to the actual operating conditions of power machines and engines.

2. Experimental apparatus, instrumentation and technique

The working part of the wind tunnel had a rectangular cross-section with dimensions of 70×90 mm (Figure 1). Input flow Mach number was 2.2, total temperature was 293 K. On the side walls of the working channel, for observation of the flow pattern, windows with optical protective glasses were mounted. The flow was visualized by an optical method using the Schlieren photography (Figure 2).

The experimental model used was a plate of Plexiglas, a material with a low value of 0.19 W/(m·K) for thermal conductivity coefficient. The use of such material allowed to consider the model heat-insulated [14]. The model was installed on the lower wall of the working part of the wind tunnel parallel to the main flow. The width of the model corresponded to the width of the working part of the tunnel – 70 mm, length was 200 mm. In front of the model a rib was placed at an angle of 90° to the flow direction. The rib height varied from 2 to 10 mm while boundary layer thickness for smooth model flow in the region of rib placement was about 6 mm. The Reynolds number was calculated with the use of the length of the dynamic boundary layer – distance from the nozzle throat. Its value was over 20 million, which indicates a turbulent flow regime.

The method of conducting the research was to register changes in the parameters on the model wall from the moment the wind tunnel was started for 90 seconds with a frequency of 1 Hz [15-17]. Data from all sensors of the stand was collected in NI SCXI-1303 thermocouple and pressure sensor blocks, then through SCXI-1102 (thermocouple) and SCXI-1102B amplifiers (for pressure sensors) were fed to
NI PCI-6220 analog-digital converter. The program for obtaining and processing experimental data was written in LabVIEW and reflected in the form of virtual instruments on a PC monitor.

Figure 1. Experimental apparatus and instrumentation schematic model.

The uncertainties of the main parameters were defined for 95% confidence interval [18]: Mach number ±1.2%, Reynolds number ±2.7%, wall temperature ±0.5K, flow stagnation temperature ±0.3K, stagnation pressure ±6kPa, static pressure ±0.5kPa.

3. Results and discussion

Figure 3 shows the change in the difference between the flow total temperature and adiabatic wall temperature along the model length. For a smooth model flow this difference is about 17 degrees. For flow behind the rib the temperature difference increases: from 19 degrees for 2 mm rib height to about 28 degrees for 10 mm rib height. Thus, rib placement led to an increase in the temperature difference between the total temperature and adiabatic wall temperature by up to 60%. The effect persists over a length of more than 25 calibers. This effect corresponds to the previously obtained results for the study of the adiabatic wall temperature on the steady thermal regime [19, 20].

Figure 3. Field distribution of temperature difference between total temperature $T_0$ and adiabatic wall temperature $T_{aw}$ for flow around smooth wall and for flow behind the rib of varied height along the model length relative to boundary layer thickness $\delta=6$ mm.
Figure 5 shows the ratio of the Stanton number for flow behind the rib to the value for smooth model flow $St/St_0$. Similar to the results of the research mentioned in the introduction for supersonic flow in the wake of a rib increased heat transfer coefficient is recorded. It was noticed that for a rib height equal to the thickness of the boundary layer the increase in the Stanton number was maximum.

4. Conclusions
The results of an experimental study of thermal gas dynamics parameters for a supersonic flow in the wake of a rib are presented. A decrease in the temperature difference between total temperature and adiabatic wall temperature was recorded in the wake of the rib by up to 60% in comparison with the smooth model flow. The effect of reducing of the adiabatic wall temperature is enhanced by increasing the height of the rib from 2 to 10 mm. Stanton number was increased by up to 50% for the ribbed wall in comparison with the smooth one. Reported effects persist at lengths greater than 25 calibers downstream behind the rib.

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