NUMERICAL ANALYSIS OF HEAT TRANSFER ON WINDWARD PLANE OF A BLUNT DELTA WING

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Development of novel as well as upgrading existent hypersonic flight vehicles of gliding descent is one of runaway trends in modern space technology. Such vehicles have to possess sufficient value of lift-to-drag ratio to effectively decelerates in the upper atmosphere. In that case the vehicle moves long-term time interval at high incidence that leads to intensive heating of its surface. The stagnation point region, neighborhood of edges of wings or aerodynamic controls and windward surface becomes in this situation the most heated parts of the vehicle. The simplest geometry shape that has all those elements is a delta wing with blunted nose. For this reason numerous studies in scientific literature were dedicated to both experimental and theoretical investigation of flow and heat transfer around delta wings.

In spite of relative simplicity of delta wing geometry there are diverse flow regimes in the shock layer over the wing surface even for sharp-edged wings (see, for example [1]) depending on governing parameters of the flow: Mach number – M_∞, Reynolds number – Re_∞, angle of attack – α, sweep angle – λ, wall-to-stagnation temperature ratio – $t_w = T_w/T_0$. Blunted nose and wing edges complicates the flow and heat transfer to the wing surface still more adding to above-listed a number of additional governing parameters equal to ratio of different reference lengths.

In the most cases the flow near windward plane of the wing can be subdivided on parts where it forms due to the shape of shock waves generated by nose bluntness, edges and windward surface as well as parts where flowfield is determined by interaction of those shocks. Qualitative analysis of hypersonic flow of blunt-edged delta wing with sharp nose at zero incidence was first made in [2] and asymptotic solution of that problem (using explosion analogy) was provided in [3]. The general case of 3D flow around blunt delta wings at incidence can be investigated using either experimental (for example [4],[5]) or numerical ([6],[7]) methods. In cited works the wings of either low stretch (of length about ten radii) were considered or general attention was paid to the edge heat fluxes or to the formation of vortex sheet at leeside plane. In [8] test data on heat transfer to windward plane of blunted delta wing with sweep angle $\lambda = 75^{\circ}$ obtained by the IR-thermography method are presented. Opposite to conventional sensors methods that approach allows receiving whole heat transfer field over all studied surface. Appearance of regions of elevated heat transfer near the windward symmetry plane were first described in that paper. Narrow strips of increased heat fluxes appears at some distance behind the nose (about $12 \div 15R_0$) and stretch far downstream almost parallel to the symmetry plane (at low distance from it $\sim 3\text{–}5R_0$) up to end section of the body (at $x \approx 100R_0$). The level of heat transfer in the strips was as much as two-three times greater then background level (that corresponds to the value on a flat plate at incidence). Besides, it was found that this phenomena occurs only within sufficiently restricted range of governing parameters and, in particular, there was no such elevation for the similar, but sharp-nosed wing. Explanation of this phenomena was offered in [8],[9]. It turns out that these strips are issued from the region of interaction of shock waves generated by the wing nose and edges. Rapid change of shock slope angle in that regions leads to large variation of entropy and slope of streamlines behind the shock. This divergent bundle of streamlines flowing out of the edge to the windward plane with almost constant pressure (such a flow was apt-called “inertial” one in [9]) forms a line with intensive divergent crossflow along which studied strip arises. Numerical investigation of inviscid flow for that case [10] also demonstrated existence of convergent zone with large entropy gradients near the symmetry plane.

In present work we will try to provide a detailed analysis of such flow around delta wings. Since the Reynolds number for experimental conditions was large enough ($\sim 10^6 \div 10^7$ on the wing length), then for the sake of simplicity the numerical analysis was made on the basis of Euler together with boundary layer equations solution. Current studies was undertaken to specify the range of governing parameters determining gasdynamic flow structure: nose bluntness radius, sweep and attack angle and Mach number where this phenomena takes place. Values of Reynolds number and wall temperature factors was not varied in the work and were chosen to be corresponded to the test case parameters ($Re_\infty = 10^6$, $t_w = 0.25$).

Numerical method

Inviscid flow (Euler) equation system is solved by shock-capturing cell-centered finite volume method in Cartesian coordinate system. All the flow domain is subdivided onto two parts – region of the nose bluntness with predominantly subsonic flow behind the bow shock and region near the vehicle side surface where the flow is supersonic. Within the first region the solution is found by the unsteady approach and within the second – by spatial marching procedure downstream along the body longitudinal axis. Flow parameters at the finite volume faces in the transverse sections are specified by means of third-order-in-space MUSCL interpolation [11] with TVD – limiters. Inviscid fluxes at the volume faces are found by the Roe’s flux-difference method [12].

Heat transfer parameters at the body surface are estimated using calculated inviscid flowfield by approximate solution of the boundary layer equations. They are solved by integral method of local similarity [13] with the use of axisymmetric analogy. The radius of effective axisymmetric body at a point of the surface is derived from the value of divergence of the unit velocity vector of inviscid flow at the wall. Integral equations of the boundary layer at the surface are written at the same coordinate system that was used in inviscid calculations (with except of stagnation point neighborhood where a local coordinate system with the pole at that point is used) relative to Cartesian components of velocity vector.

Such an approach is deliberately wrong at the leeward flow region where the flow is globally viscous. However, it yields quite reliable results at the windward side for the flows with sufficiently high Reynolds numbers, providing that boundary layer thickness is remarkably lower than the shock stand-off.

Results of numerical analysis of the flow at windward plane of the delta wing.

Comparison of the computed results with the test data [8] is shown in fig.1 for the cases λ=75˚, α=10,20˚, M_∞=14. Distribution of the normalized heat flux q/q_0 (q_0 – heat flux to the stagnation point of a sphere with a radius equal to the edge radius r_e) along the wing span z/R_0 (R_0 – nose bluntness radius) at the section $x/R_0 = 75$ is presented in the figure. The character of heat flux distribution corresponds to the test results [8] – there are a minimum at the symmetry plane and local maximum adjacent to it (at crossflow divergence line induced by bow shock interaction with the edge), lowering down to the level characteristic to the flat plate at incidence and further increasing at the wing edge.

A number of differences between calculation and tests should be noticed at general reasonable coincidence of the heat transfer level. First, there is a deep collapse in the value of calculated heat flux at the symmetry plane for the case $\alpha = 10˚$. This difference is explained by appearance of a streamline with quite intensive

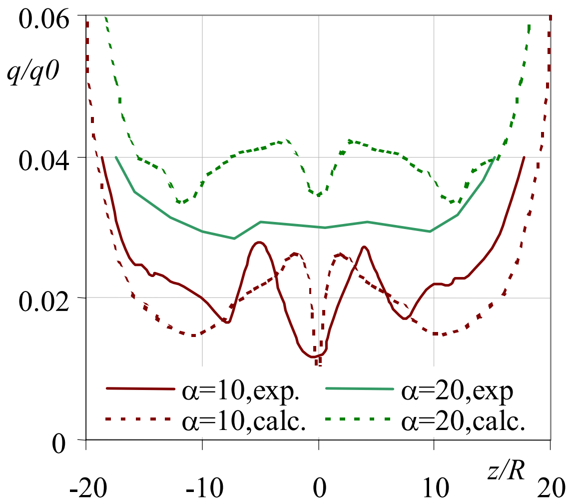

Fig. 1

crossflow divergence on the wing surface at $z = 0$ and using of method [13] for such conditions becomes rather questionable. In this case the velocity profile in that plane substantially differs from the Blasius one (that is fundamental assumption in [13]), besides, the boundary layer thickness in this region grows so considerably that account for viscous-inviscid interaction effects and viscous terms both normal and azimuth direction turns to be essential Second, location of crossflow divergence line (corresponding to the local heat flux maximum) is displaced closer to the symmetry axis compared to the tests that can be also explained by the viscous-inviscid interaction. Third, at attack angle of $\alpha = 20˚$ mentioned character of heat transfer distribution in the tests was proved to be only weakly but as concerns calculations it is quite essential, though to the lower extent than at $\alpha = 10˚$. Still another reason for some difference between numerical and experimental data is that in presented case the radius of nose bluntness in tests was $R_0 = 2r_e$ but calculations were made for $R_0 = r_e$.

A number of calculations were made with the different nose bluntness geometries to study the effect of the nose shape on heat fluxes. For this purpose the frontal part of the spherical nose (at $\varphi < 45˚$) was replaced by spherical segment of greater radius (up to $5R_0$ – practically cylindrical face). Since the bow shock shape was almost identical at some distance downstream from the nose (shock stand-offs are proportional to the square root of the drag coefficients but they are varied within the range from 1 to 2) then resulting distributions of heat transfer to the wing surface at $x/r_e > 20$ is nearly the same for all these cases. In the experiments the bluntness was pure spherical with replaceable nose parts of different radii projected over the wing plane from the leeside and accurate reproduction of the model geometry turns to be impossible in the calculations.

In addition, a number of computations were made to reveal the range of governing parameters for which the numerical analysis allows obtaining considered phenomena of elevated heat transfer.

Effect of wing sweep angle

Study of wing sweep angle the on heat transfer peaks was made by means of numerical simulation of delta wing flow at $\alpha = 10^\circ$, $M_\infty = 14$ and $\lambda = 85\text{-}45^\circ$ with a step on sweep angle $\Delta\lambda = 2.5^\circ$. As calculations shown the considered phenomena exist over the range of sweep angles of $\lambda = 82.5\text{-}60^\circ$ (some results are presented in fig.2 in coordinates q/q_0, z/z^* – z^* – wing span for the wing exit section). At greater sweep angles the crossflow divergence line occurs in the windward symmetry plane (like that in case of blunted cylinder) and wing edges are subsonic, i.e. gas flows to the edges from the windward surface. At lower sweep angles the wing edges affects the nose bow shock and there is no interaction of the shock waves at the edge that is specific for the studied phenomena. Within the range $\lambda = 85\text{-}60^\circ$ the origin of crossflow divergence line more and more approaches to the nose-edge junction point as sweep angle decreases from $x/R_0 \approx 15$ at $\lambda = 85^\circ$ to $x/R_0 \approx 3$ at $\lambda = 60^\circ$. The greatest peaks of heat transfer

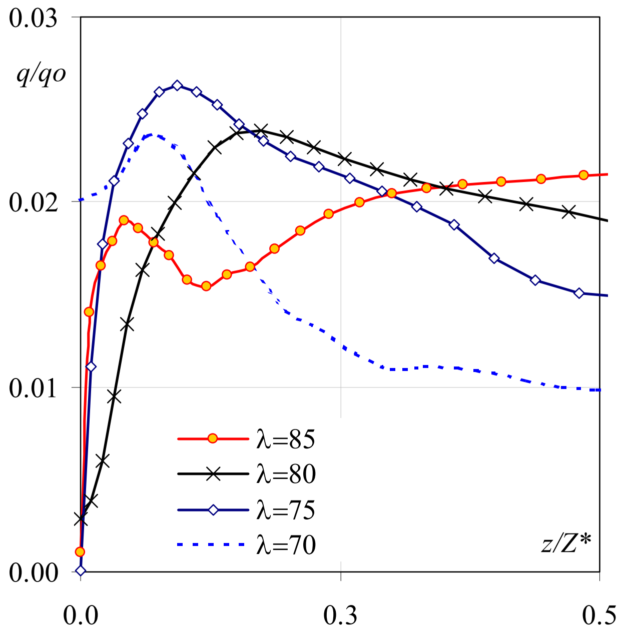

Fig. 2

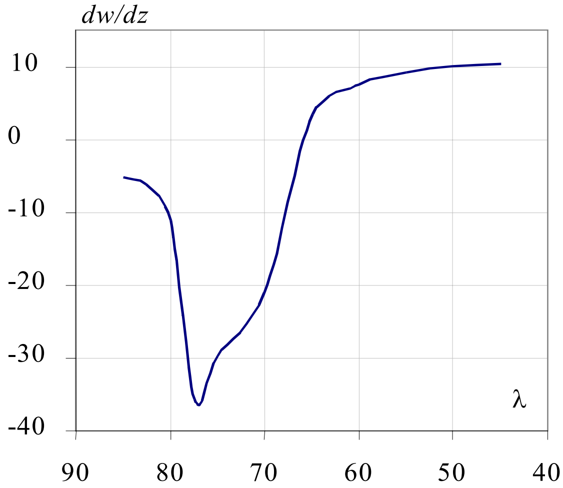

Fig. 3

were found out at $\lambda = 77.5^\circ$, in this case heat flux within the strip was about 2.5 times greater than that for the plate at incidence. The magnitude of crossflow convergence in the symmetry plane is the most intensive in this situation. The value of $\partial w/\partial z$ (w – z-component of velocity) can be considered as a measure of gas convergence/divergence in that plane, its dependence on the sweep angle is presented in fig.3.

This dependence takes complex non-monotonic character, at greater sweep angles ($\lambda \le 90^\circ$) the value of $\partial w/\partial z > 0$ as in case of blunted cylinder, however at $\lambda \sim 85^\circ$ this value becomes negative and crossflow convergence line is formed but divergence line at $z \sim 5R_0$ is still rather weak. At further decreaseof λ intensity of crossflow convergence grows sharply, simultaneously increases intensity of crossflow divergence in the region of heat transfer peak. Achieving its maximum at $\lambda = 77.5^\circ$ the intensity of convergence starts decreasing and at $\lambda \approx 65^\circ$ gas becomes to flow away the symmetry plane. However the crossflow divergence line at $z \sim 2R_0$ keeps holding up to magnitudes $\lambda < 60^\circ$.

Effect of wing attack angle

In order to specify the margins of wing attack angle influence on the phenomena of heat flux peaks on the wing windward plane

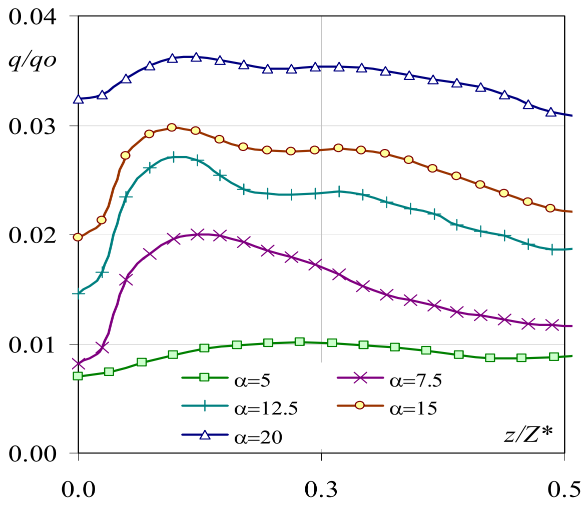

Рис. 4

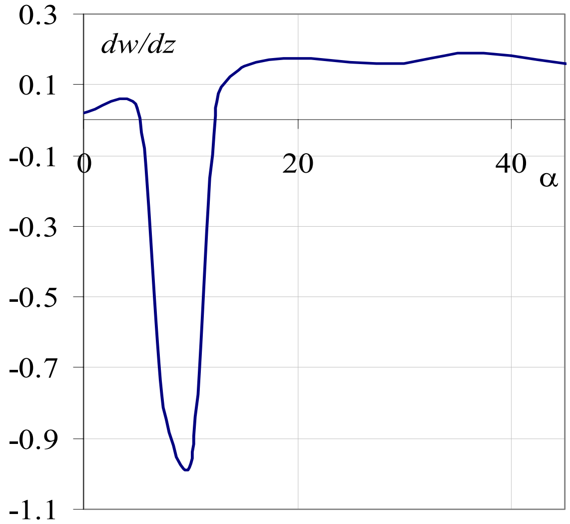

Рис. 5

several computations of the flowfield and heat transfer around the vehicle were made under conditions: $M_\infty = 14$ and $\lambda = 75^\circ$, $\alpha = 0\text{-}50^\circ$ with step on angle of attack equal to $\Delta\alpha = 2.5\text{-}5^\circ$. Numerical analysis has shown that this phenomena occurs only within the narrow range of incidence angles $\alpha = 5\text{-}20^\circ$ (see in fig.4).

In this case the heat flux peak value at extreme limits of the range $\alpha = 5^\circ$ and 20° is sufficiently low and can be hardly observed in tests. It is in a good resemblance with the results of [8] where studied phenomena was registered only within the angle range $\alpha = 5\text{-}15^\circ$. Maximum level of relative heat flux peak in computations was achieved at values $\alpha = 7.5\text{-}10^\circ$ that can be also noticed from fig.5 where dependence of crossflow derivative magnitude on the vehicle attack angle is shown. The crossflow convergence line in the symmetry plane is formed at $\alpha \approx 5^\circ$ and further its intensity grows sharply until $\alpha \approx 10^\circ$, after that it starts steep decreasing. At $\alpha \approx 12^\circ$ the crossflow convergence changes into divergence and further keeps almost constant independent on the attack angle, the flow regime with supersonic leading edges changes to the regime with subsonic ones. For this reason at greater angles of attack the considered phenomena is not already observed, besides, windward plane of the wing starts affecting the shape of the nose bow shock. At angle $\alpha \geq 30^\circ$ flow divergence streamline at the leading edge displaces to the edge-plane junction point and then, with further increase of α becomes situated at that plane.

Distribution of the heat fluxes at windward plane of the wing is shown in fig.6 for the case $\alpha \approx 7.5^\circ$, where lines of constant normalized heat flux q/q_0 and inviscid stream lines are shown. Divergent fun of the streamlines emanating from the leading edge at $x \approx 17$ мм $= 13R_0$ – in the vicinity of this point the bow shock intersects the edge, is clearly seen. Along the bisectrix of that bundle the crossflow divergence line with peak heat transfer in it is stretched down stream almost parallel to the symmetry plane. Hence, in this situation the influence of nose bluntness is

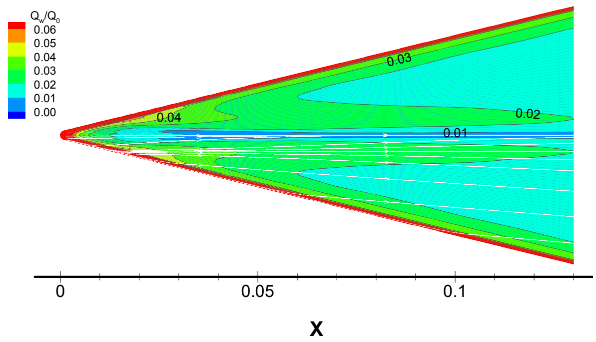

Рис. 6

non-local but spreads along all the wing surface (at least to several hundreds of radii) that is sufficiently rare situation in gasdynamic of super- and hypersonic vehicles.

Effect of inflow Mach number

Considered phenomena is only observed within the restricted range of supersonic Mach numbers. So, at $M_\infty = 6$, $\lambda = 75^\circ$, $\alpha = 10^\circ$ the bow shock shape is such that the shock does not intersect the leading edge at any value of x. But at $M_\infty = 8$ this phenomena already occurs. As expecting its intensity grows with the Mach number and distance from the symmetry plane to the heat transfer peak strip decreases up to $M_\infty \sim 10$ when hypersonic stabilization of flow begins.

Conclusions

Numerical analysis of the heat transfer to the windward plane of delta wing with blunted nose and leading edges was carrier out on the basis of solution of Euler and integral boundary equations under flow conditions at which the phenomena of formation of narrow expanded strips of heat transfer peaks was observed in the tests [8]. The conclusion of [8],[9] that origination of these strips is connected to formation of crossflow divergence streamline, issuing from the region of the bow shock interaction with the leading edge shock, is verified by the results of numerical simulation of such flow. Parametric calculations being carried out enables to determine general rules and existence margins of the phenomena under study.

Some distinction between experimental and numerical data indicated that more exact CFD modeling of this phenomena is required with account for viscous-inviscid interaction effects and azimuthal viscous terms using full Navier-Stokes equations.

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