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AERODYNAMIC FEATURES OF HYPERSONIC SPEEDS

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SUMMARY (Give summary which highlights the salient factors of narrative report. Begin narrative text on AF Form 112a unless report can be fully stated on AF Form 112. List inclosures, including number of copies)

Forwarded herewith is a translation of an article entitled "Aerodynamic Features of Hypersonic Speeds" (Osobennosti aerodinamiki giperzvukovykh skorostey), written by Docent, Candidate of Technical Sciences M. L. Gofman and published in P: Vestnik Vozdushnogo Flota (The Herald of the Air Fleet), No. 11, 1957, pp. 56-64.

The article describes some aerodynamic features of hypersonic speeds and how a hypersonic flight is investigated in wind tunnels and impact tubes by means of models.

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THE FEATURES OF HYPERSONIC SPEED AERODYNAMICS

A continuous progress of supersonic aviation presents a series of new problems associated with the flight of flying devices at high Ms. For instance, to be sure that the range of the rocket will reach 10,000 km, the speed must correspond to M = 20 (Ms are based on the sonic speed in stratosphere). For an Earth's satellite M should be about 25. It should be reminded that the meteors travel at M equal to from 30 to 100. Thus, because of the development of missiles, rockets, and the possibility of interplanetary communication the aerodynamics of hypersonic speeds acquire greater and greater importance.

A hypersonic flow possesses a series of characteristic features which appear at high Ms. To explain them, one should investigate the physical picture of the streamlines of hypersonic flow around a body. It is known that the angle of inclination of the shock and the angle of disturbances decrease with the rise of M. Furthermore, at high Ms these angles are so small that the shocks as well as the lines of disturbances strive to adjoin the surface of the body over which the flow proceeds. It is possible to say that the shock becomes almost parallel to the direction of motion. In this case the region of disturbances is very small and can be compared with the region occupied by the boundary layer.

For instance, if at M = 5 the angle of disturbances for a thin body is equal to 11.54°, then at M = 10 this angle will be equal to 5.7°. The calculations show that at M = 10 the angle of inclination of the shock for a double-wedge airfoil amounts to 8.2°.

In Fig. 1 is shown the flow over a plate at very high M. Since the increase of pressure outside the shock at high M by far exceeds the decrease of pressure in the rarefied flow, one can approximately assume that over the upper surface of the plate is a vacuum. On the lower surface of the plate an oblique shock adjoins the plate very closely. Therefore the flow shown in Fig. 1a excellently corresponds with the flow about a plane plate shown in Fig. 1b, which proceeds according to the Newton's theory of impact. Thus, it was found that the Newton's theory which yields wrong results at low speeds, fits well for very high Ms. Presently, the so-called linear theory, which is based on the assumption that bodies (wing fuselage) of small thicknesses at small angle of attack are treated, is widely used for practical problems of supersonic aerodynamics. In such case the existing shock can be replaced by a line of disturbances, and thus simplify the exact theory. For instance, according to this theory the pressure coefficient changes in direct proportion to the angle of attack.

For a hypersonic flow the linear theory becomes useless. The following explanation of this can be given: this theory can be applied only when the maximum angle at which the flow is declined due to the presence of the body is small in comparison with the angle of disturbance of the free-stream. But in a hypersonic flow the cone of disturbances is so set that their angle can be compared with the angle of inclination of the plate's nose.

This leads to a fact that the coefficient of the air pressure becomes proportional to the square of the local angle of attack. Hence, the linear theory appears to be approximately correct only for small angles of attack, up to Ms between 4 and 5. From Fig. 2 it is seen that already at M higher than 3 the linearity of the graphs, representing the coefficients of the lift force of a plane plate, is distorted. In addition, it should be born in mind that according to the linear theory the aerodynamical coefficients do not depend on the nature of the gas and the character of the thermodynamic process. However, at higher Ms the nature of the moving gas changes.

Due to a strong impact during the collision of molecules traveling at high speeds, their compound atoms start to vibrate respectively to their mean position. Additional degrees of freedom will appear which will lead to an increase of specific heat at constant volume and to a decrease of the ratio K, from K = 1.4 to K = 1.29.

As to the term "hypersonic" flow itself, the results of the linear theory allow to set the conditional limits of this flow. At M greater than 5 it can be assumed that $\sqrt{M^2 - 1} \approx M$ (for instance, at M = 5 $\sqrt{M^2 - 1} = \sqrt{25 - 1} \approx 5$).

This leads to a simplification of some known formulas of the lift force and the head resistance of a double-wedge airfoil. It can be assumed that the term "hypersonic" concerns Ms higher than 5.

The following law of similarity of hypersonic flow can be defined. If bodies having similar forms, but a different relative thickness $\bar{c}$, are placed into a flow of a different M so that the parameter $M\bar{c}$ will remain constant, then the flow will be also similar, i. e., the streamlines will be alike. This means that if a hypersonic flow is over the airfoils whose thickness and curvature are distributed identically while the angle of attack is proportional to the relative thickness, then their aerodynamic coefficients depend only on the criterion of similarity $M\bar{c}$.

To illustrate this law, an example is given.

Suppose that coefficients c_y and c_x of a symmetric airfoil with a thickness $\bar{c}_1 = 10\%$ at an angle of attack $\alpha_1 = 5^\circ$, tested at M = 5, are known. The question is: what will be the aerodynamic coefficients of a geometrically similar airfoil whose relative thickness $\bar{c}_2 = 5\%$, and to which M will they correspond. Since there is a similarity, the parameters of similarity should be equal, i. e., $M_1\bar{c}_1 = M_2\bar{c}_2$. It follows that $M_2 = 10$; in addition, the similarity of the airfoils will be at $\frac{\alpha_2}{\alpha_1} = \frac{\bar{c}_2}{\bar{c}_1}$, i. e., $\alpha = 2.5^\circ$.

From the formulas for the coefficients it follows that

$$\frac{c_{y_2}}{c_{y_1}} = \left(\frac{\bar{c}_2}{\bar{c}_1}\right)^2 = 0.25; \qquad \frac{c_{x_2}}{c_{x_1}} = \left(\frac{\bar{c}_2}{\bar{c}_1}\right)^3 = 0.125.$$

Thus, the criterion of similarity by the aid of the known coefficients of one airfoil permit quite easily to determine the coefficients for similar airfoils. For an illustration, polars of various airfoils, computed for three values of similarity parameter, are given in Fig. 3.

Although the hypersonic theory of similarity is only an approximate one, it, nevertheless, corresponds well with the experiment and allows to compute the aerodynamic characteristics of the wings and the bodies of revolution.

An important distinctive feature of the hypersonic flow is the strong effect of viscosity which leads to an essential interaction between the boundary layer and the shock wave.

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During the investigation of a supersonic flow, a general approach can be applied with regard to the analysis of viscosity. We assume that the effect of viscosity concentrates in a thin layer adjoining the surface of the body, the so-called boundary layer, and that it can be determined as a correction to a nonviscous flow. At small supersonic speeds the boundary layer and the shock wave in the forward part of the airfoil are located far from each other, so that the effect of their interaction can be neglected. It is true that, in this case, the effect of the boundary layer shows up near the trailing edge, since in this region the shock wave and the boundary layer may be the cause of the separation of the flow. This occurs because the increased pressure behind the tail shock P_2, being unable to propagate forward (at supersonic speeds the disturbances do not spread forward), penetrates into region BDC (Fig. 4) through the subsonic part of the boundary layer. The raised pressure in this region causes a separation which prevents a further expansion of the flow, and thus, the pressure in the region BDC appears to be greater than the theoretical calculations would show. This is clearly seen from the curves representing the distribution of the pressure. From Fig. 5 it is seen that the theory and the experiment coincide well in the forward part of the airfoil; they differ only in the rear part of the airfoil in the region of an increased pressure (decrease in rarefaction).

A different picture appears at hypersonic speeds. Firstly, the boundary layer behind a strong head shock wave over a very thin airfoil has a larger thickness as compared to that of the body over which the flow proceeds. The fact is that the shock wave at high supersonic speeds adjoins the surface of the airfoil so closely that the entire region between the surface and the shock wave should be considered as a region of viscous flow. Consequently, when the parameters of a shock wave are being determined, the effect of viscosity should be considered, and when the friction on the surface of the plate is being evaluated, the effect of the shock must be especially taken into account.

Thus the flow in a boundary layer strongly affects the stream in the region between the outer surface of the boundary layer and the shock wave. For instance, near the nose of the plate the boundary layer has a big curvature which, in turn, bends the head shock. Disregard of viscosity would lead to a situation where the shock would appear as rectilinear. Because of this, the region of the boundary layer and the region between the shock and the outer surface of the boundary layer cannot be investigated separately, since their interaction must be taken into account.

If, for instance, one will examine the flow around a plane plate at hypersonic speed, then the entire field of the flow, disturbed because of the presence of the plate, can be divided into three regions (Fig. 6).

Close to the plate is a boundary layer in which the effect of viscosity is strong.

From the nose of the plate, [illegible], a powerful shock wave appears; at the nose of the plate the inclination of the shock is big, afterwards it sharply decreases and, for instance, at high M the shock wave looks like drifting along the plate. A region of undisturbed stream is above the shock wave.

Finally, between the shock wave and the outer border of the boundary layer is a small region of the flow in which the effect of viscosity can be neglected; across this flow a big change in pressure and density can be observed.

Owing to that the boundary layer thickens, the shock wave at the leading edge has a curvilinear form. As a result of this, the pressure along the plate is not constant and can be higher than that of the surroundings. Furthermore, this excess of pressure depends on M and Re (Reynolds number - a dimensionless magnitude defined by the ratio of the inertia forces and the viscosity forces in the flow). For instance, at M = 5.8 and a distance of 0.63 cm from the leading edge the pressure increases by 70%. The increase of the thickness of the boundary layer can be characterized by the following: at M = 7 the thickness of the boundary layer is 10 times greater than at M = 1 at the same Re (Reynolds). At great distances from the leading edge the shock wave and the boundary layer drift apart by a considerable distance so that statically the pressure does not change any more.

It was already shown (Fig. 5) that the boundary layer at low supersonic speeds (and also at subsonic) affects the distribution of pressure, first of all, in the tail part of the body. At hypersonic speeds the thickening of the boundary layer becomes so considerable that it provokes an essential change in the distribution of pressure over the nose part of the airfoil. At the same time the usual occurences of separation in the tail part of the body may still happen.

The rise of pressure leads to a situation that the surface friction becomes higher than it would be according to the theory of the boundary layer in supersonic flow. The resistance coefficient of the laminar friction $\bar{c}_f$ decreases with the rise of M when the interaction is not taken into account, and it increases at hypersonic speeds when the interaction is taken into account (Fig. 7).

In addition, at low supersonic speeds (up to M = 3) the coefficient $\bar{c}_f$ does not depend on Re, while at hypersonic speeds it essentially depends on Re.

For characterization of the mutual interaction of the boundary layer and the shock wave the idea of "the distance of interaction" is introduced, which is expressed as a ratio $\frac{X_o}{L}$ where X_o is the distance from the nose to the point where the shock wave degenerates into disturbance wave, and L is the chord of the body (Fig. 6). From the graph in Fig. 8 it follows that for a given Re the effect of the shock propagates farther downstream when M is higher.

A special importance acquires the effect of the temperature at hypersonic speeds. A considerable increase of temperature along the head shock leads to a large alternate heat transfer to the body over which the flow proceeds. This also affects the flow close to the surface of the body.

Thus, behind the shock and along the boundary layer the temperature may become so high that the chemical features of the gas may change. This leads to a breach in the static equilibrium between different kinds of internal energy of the gas.

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Indeed, if the temperature of the gas after crossing the shock is sufficient to cause some additional degrees of freedom of molecules, then a dissociation of gas may happen, i. e., a change in the chemical compound will occur as a result of disintegration of its molecules into more simple molecules or atoms.

A still higher temperature may cause the ionization of molecules. A change in the structure of gas can be seen during the tests of models on ballistic installations or by free firing. So, for instance, by firing a ball in Xenon, at M higher than 10 one could see the ionization of Xenon. During another experiment, when a steel ball, 13.5 mm in diameter, was fired into a reservoir filled with Freon at a speed corresponding to $M = 9.27$, the disintegration of Freon was clearly seen. It shows how important is the problem associated with the investigation of the chemical features of gas near the shock wave.

The effect of additional degrees of freedom of intermolecular motion on the features of gas depends on the temperature and the pressure and varies with the altitude.

In Fig. 9 is shown the temperature of retardation for bodies moving at $M = 24$ at different altitudes. The diagram of an ideal gas (parabolic law of the temperature rise with the rise of M) does not reflect the real features of a real gas at high temperatures. So, for instance, at $M = 10$ an ideal gas shows a temperature of retardation equal to 4700° K, while a real gas depending on the altitude has considerably lower temperature of retardation.

Presently, several methods for acquiring a hypersonic flow can be recommended.

To the first method belong a ballistic installation and a free firing. Here the models are tested on ballistic installations which catapult the models into motionless air at high speeds.

Recently, the installations for free firing have been also used. As an example, a cannon which can provide a speed up to 20,000 kilometers per hour can be taken. The cannon may also fire into a tank in which the pressure of the gas or the air is variable, thus creating the conditions of necessary pressure or altitude. The firing can be also done by a high-speed cannon into a supersonic stream generated by a special pipe. In this way the cannon can fire the models in a free flight at Ms of the order of 20 and more. Applying special devices for the recording of time, it is possible to photograph the models at an exposure of one ten-millionth of a second. Although such cannons create some likeness of a real flight and real temperature, nevertheless measurement of aerodynamic forces there, like in ballistic installations, is rather difficult.

Striving to approach the real conditions, the so-called aerodynamic or rocket "vehicle" had to be developed. This is a special railroad path of 3-7 km length with some auxiliary mechanisms. Vehicles of the present designs can reach a speed up to 1500 m/sec, while the acceleration rises up to 100 g.

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Recently, investigating the hypersonic flow impact pipes were used, usually designated for the study of explosions (primarily atomic explosions), since the wind tunnels do not provide results comparable to those reached under conditions of a free flight. The most important problem associated with a wind tunnel is the sharp drop of temperature between the receiver and the working part, which is the result of the expansion of the gas at high M.

If the air is delivered at a normal temperature (for instance, at $T_o = 300^oK$), then the air temperature in the working part at M = 5 and M = 10 will decrease up to 50^oK and 14.5^oK, respectively. At an atmospheric pressure the temperature of liquefaction of the air is 78^oK. It means that in cases under consideration a condensation will occur. To prevent such phenomenon the flow must be heated.

Even having a well developed heating system it is difficult to acquire a temperature of retardation in the working part comparable to that generated in a free flight.

An impact pipe consists, essentially, of a long channel closed at both ends and divided by a diaphragm into two sections which contain gases at different pressures. During a rapture of the diaphragm a straight shock appears which propagates into that part of the pipe where the pressure is lower. The expansion of the compressed gas creates a region of a high speed flow. It is obvious that a settled motion proceeds only during a short period of time (1000 microseconds at a pressure difference equal to 2500 atm).

A steel cylinder (1) in Fig. 10, which is able to withstand a pressure up to from 150 to 200 atm, serves as a compression chamber of the pipe. It is connected with a chamber of constant expansion (3). Between the compression and expansion chambers is a plastic diaphragm (2).

Along the section of constant expansion are two observation points in which either piezoelectric transmitters or glass windows, to make photographs of the flow, are arranged. The chamber of constant expansion is connected with the divergent part of the pipe (4) in which the expansion of the flow beyond the shock is going on, thus ensuring a high M in the working part.

In the working part (5) there are also windows for both the optical study of the flow around the models, and for investigation of the pressure in the flow or over the model.

Different Ms are acquired by changing the ratio of pressures in the sections of compression and rarefaction, and also by changing the degree of the nozzle's divergence.

Hypersonic Ms can be raised in the impact pipes by using a mixture of hydrogen and helium as a working gas. The deficiency of impact pipes lies in the difficulty of measuring the aerodynamic forces which affect the model, and in the short time of action.

Presently, hypersonic wind tunnels are widely used for investigation of flow at very high Ms.

There is no essential difference between the supersonic and hypersonic tunnels (for instance, at M = 2 and M = 10); in both cases the divergent nozzle, where the necessary supersonic speed is reached, is beyond the critical section. However, there are qualitative differences.

Inability to get sufficient power and necessary cooling is the basic difficulty in dealing with the ordinary supersonic wind tunnels. The same difficulties appear also when the hypersonic speeds have to be reached.

The main peculiarity of a hypersonic tunnel is the variation of the temperature and pressure depending on M, which leads to a liquefaction of the cooled air. This creates conditions in the flow which are different from the natural ones: the liquefaction which affects the aerodynamic characteristics, the rise of the inclination angle of the shock, the change of M.

To prevent the liquefaction of the air, the compressed air must be heated before it goes to the working part. However, the compressed air cannot be heated above 250°C since the ability of the steel reservoir to withstand the pressure decreases with the rise of the temperature. Therefore, in a wind tunnel, working with air, it is not possible to obtain M higher than 10. To get higher Ms, other gases must be used: nitrogen, hydrogen, helium, freon, xenon.

A series of problems appear in connection with the low density of hypersonic flow. Low densities and temperatures increase very greatly the thickness of the boundary layer in the tunnel. Beside the difficulty in selecting the nozzle, a large thickness of the boundary layer, which distorts the posture of the shocks and angles of disturbances, makes impossible to determine M of the flow by means of the shock's inclination or by the line of disturbance.

For visual observations of the flow at high Ms the phenomenon of the luminosity of nitrogen due to its electrification (oxygen or argon can be also used but their luminosity is less bright) is used.

The new branch of aerodynamics, the aerodynamics of hypersonic speeds, presently under development forms the theoretical basis of flight of intercontinental and ballistic rockets and missiles at high supersonic speeds. By investigating their characteristics one encounters a series of new and interesting problems the majority of which require special methods.

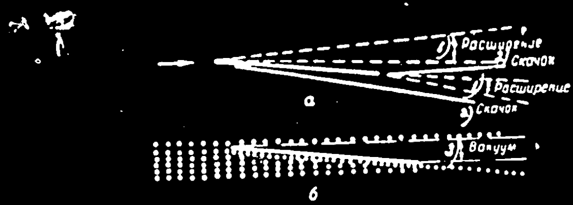

Fig. 1 - Schematic diagram of the flow around a plate at high M.

1) Expansion
2) Shock
3) Vacuum

Fig. 2 - Coefficients of the lift force of a plane plate for different Ms.

1) Linear
2) Hypersonic
3) Angle of attack
4) Uncompressible

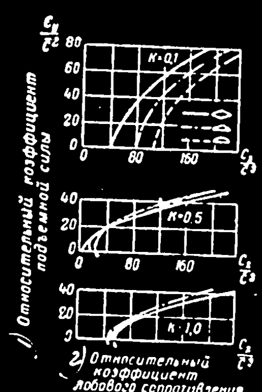

Fig. 3 - Polars for a double-wedge, triangular and convex profiles, calculated for different parameters of similarity K.

1) Relative coefficient of lift force.
2) Relative coefficient of head resistance.

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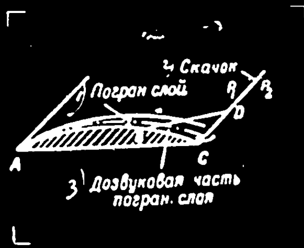

Fig. 4 - Separation of the flow at the trailing edge of a supersonic airfoil.

1) Boundary layer
2) Shock
3) Subsonic part of boundary layer

Fig. 5 - Pressure distribution over a convex airfoil whose thickness is 8.8%, the angle of attack = 0 and M = 2.13. On the diagram can be seen the increased pressure at the trailing edge of the airfoil.

1) Pressure coefficient
2) Theory
3) Experiment

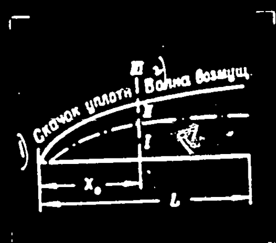

Fig. 6 - Three regions of hypersonic flow around a plane plate.

I - Boundary layer
II - Non-viscous flow
III - Free stream
1) Shock wave
2) Wave of disturbance

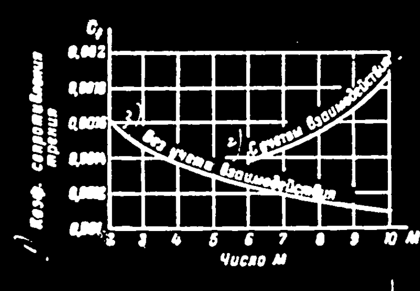

Fig. 7 - Dependence of median pressure coefficient of a plane plate on M with and without the consideration of interaction.

1) Resistance coefficient of friction
2) With consideration of interaction
3) Without consideration of interaction

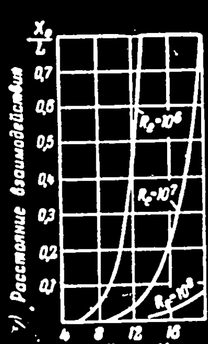

Fig. 8 Dependence of the "distance of interaction" on Re and M

1) The distance of interaction

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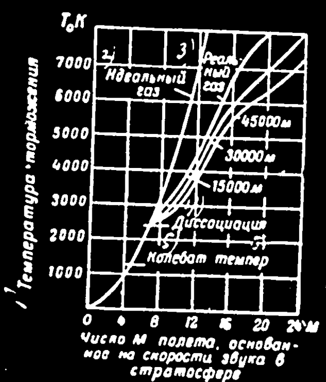

Fig. 9 - The temperature of retardation for bodies moving at different Ms and different altitudes.

1) Temperature of retardation
2) Ideal gas
3) Real gas
4) Dissociation
5) Fluctuating temperature
6) M of flight based on the speed of sound in the stratosphere

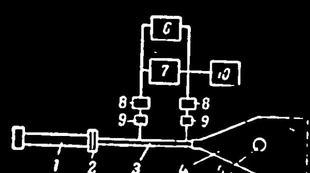

Fig. 10 - Schematic drawing of hypersonic impact pipe.

1) Pressure chamber
2) Diaphragm
3) Section of a constant expansion
4) Divergent section
5) Working part
6) Observation section
7) Counter
8) Amplifiers
9) Transmitters
10) Spark discharger

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