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WIND TUNNELS FOR HYPERSONIC RESEARCH (PROGRESS, PROBLEMS, PROSPECTS)

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The prospects for developing astronautics, the utilization of space stations and platforms, and the development of systems for communication, navigation, and environmental observations require development of a new generation of transport systems which can substantially lower the cost of putting cargo and people into orbit around the Earth and other planets. Returning dangerous and derelict structures from space will become an even more urgent problem, as they become more noticeable; in the future they can become a serious ecological problem.

Efforts to solve these problems have recently led to the planning of several space transport systems, including [1-6]:

— single-stage aerospace planes: NASP, X-20, HOTOL, STS-2000, and HOPE;

— two-stage aeroastronautic systems with horizontal takeoff and landing: ZENGER and STAR−H; and

— two-stage aeroastronautic systems with vertical takeoff and horizontal landing: ARIAN−5/HERMES, et al.

Realizing these plans depends to a large degree on successfully solving the following problems: airframe aerothermodynamics, the gas dynamics of mixing and heating in the jet engines, and design optimization of the airframe, the air intakes, and the nozzle assembly.

These transport system concepts have several problems in common, including:

— external aerodynamics of shock encounters with the laminar and turbulent boundary layers;

— aerodynamic heating, ablation, and erosion of spacecraft surfaces;

— internal gas dynamics in the engines (flow interactions in air intakes and combustion chambers, mixing and compression of the fuel under prolonged hypersonic conditions, and completeness of chemical reactions);

— modeling non-ideal gas effects related to molecular dissociation and recombination and to excitation of vibrational degrees of freedom at high static temperatures; and

— catalytic interaction of aircraft surfaces and the resultant local structural heating.

At least some of these plans envision a highly efficient hypersonic aircraft that uses atmospheric oxygen as the oxidizer during the jet boost phase and burns up the fuel at high hypersonic velocities (hypersonic-combustion ramjet).

Current studies of the physical processes that occur during hypersonic combustion in a jet engine are woefully inadequate. One reason for this is that the wind tunnels and high-enthalpy facilities used for this research do not totally model the Reynolds number, often contaminate the airflow, and do not provide steady-state conditions for the times necessary to establish equilibrium in modeling the combustion.

At the same time, numerical methods of solving these are still limited, and their development is retarded in many cases by the lack of reliable experimental data required to verify the models and methods which are selected and developed. Therefore, in spite of the constant progress in numerical algorithms and in computer capabilities, experimental research remains the basis for solving these problems and for developing numerical methods.

Flight tests are extremely expensive and do not provide enough information, not to mention the fact that the range of the results are related only to currently existing systems, not future ones. Obviously, they can be useful and effective only in the final stage of design testing. Therefore construction of ground facilities capable of handling most of the research under laboratory conditions remains an urgent problem.

Modeling the motion of objects in the Earth's atmosphere and the atmosphere of other planets requires gas-dynamic facilities capable of producing flows with Mach numbers of $M = 8\text{-}25$ at Reynolds numbers of $Re = 10^6\text{-}10^8$. Elementary estimates show that it is hopeless to try to completely reproduce all the flow parameters around an object in a ground facility, even one with "cosmic" energies (the power required to create a flow ≈ 1 m in diameter can be up to 800 MW [4]).

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Apparently almost the longest possible stationary flows with M $\leq$ 10 has been obtained in the periodic T-117 wind tunnel at the N. E. Zhukovskiĭ Central Institute of Hydrodynamics (TsAGI) [7]. The large diameter of the working section ($\approx$1 m) allows it to be used to test models of an acceptable scale, including engine models. Unfortunately the relatively low initial gas pressures ($\approx$200 atm) do not allow complete modelling of the Reynolds number. More important, however, is that in T−117, as in all arc jets, plasmatron fluctuations are generated in the stagnation temperature; after expansion occurs, they cause substantial velocity fluctuations that affect mixing processes. Also, the flow is contaminated by air dissociation products, because local arc temperatures exceeded the temperature for almost complete dissociation of the air. Add to this the contamination from electrode erosion, and the serious problems that arise in modeling combustion become obvious, especially at relatively low static temperatures, where contaminants and parameter fluctuations can seriously delay ignition.

Attempts to obtain M > 10 in this type of tunnel lead to huge energy costs. Burning fuel for heating and then adding excess oxygen (for example, at NASA Langley [4]) does not provide the necessary pressures and enthalpy at M > 10, and it contaminates the flow.

Thus, the only reasonable alternative is to use pulsed tunnels with some form of energy accumulation and to downsize the models appropriately. Here obvious requirements are maintaining constant flow parameters over a time period much longer than that required to establish the flows, make the measurements, and collect the data. Depending on the problem, this time ranges from several milliseconds to a tenth of a second and is the main determinant used in designing pulsed tunnels.

Because ground facilities cannot reproduce flight conditions completely (even in pulsed tunnels), the problem arises of adequately modeling stationary working conditions in order to investigate aerothermal effects of the external and internal flows.

Undoubtedly the most general requirement is total reproduction of the gas dynamic flow conditions in terms of the Mach and Reynolds numbers, because they define flow similitude in a viscous compressible fluid. It is especially crucial to model these parameters if there is heat transfer, mixing, attached and detached flow, or distributions of velocity, pressure, force, and momentum acting on the aircraft, etc. Thus a complete M and Re model is in fact required for all aerodynamic experiments, independent of other flight conditions.

Other very important conditions include:

— maintaining the conditions for an acceptable time period;

— reproducing the basic flow parameters (pressure and stagnation temperature) for M > 10;

— capacity for relatively large-scale test models;

— maintaining high purity of the working gas;

— modeling dissociation processes and transient effects during expansion in the nozzle;

— modeling acoustic and enthalpy perturbation distributions; and

— reproducing the static temperature and the binary reaction parameter ρL in modeling the initial combustion stages in a hypersonic-combustion ramjet, as well as modeling the tertiary reaction parameter ρ^2L in the final stages and during the expansion of the combustion products in the nozzle.

Unfortunately no one type of hypersonic wind tunnel, either operating or being designed in any country in the world can satisfy all these requirements. However, in solving actual hypersonic flight problems, one or another of the flow parameters can be unusually important. Thus, modeling stress interactions from external flow requires reproducing the pressure distribution (and not necessarily completely). From the viewpoint of the interaction of the surface flow and the effect of transients, dissociation, and recombination it is better to model the stagnation temperature (if it exceeds 2500-3000 K) and the gas density as well as possible. In modeling mixing, it is the velocity difference between the flow and the spray stream.

The necessity of modeling the stagnation temperatures for flight at high Mach numbers is rather widely discussed in the literature [1-6]. In truth, many problems, such as heat transfer, ablation, and surface erosion cannot be solved if the stagnation temperatures are not sufficiently close to reality. In studying these phenomena, the requirements on flow purity and test time can be significantly reduced, and these problems can be solved more or less successfully today by using shock tunnels or facilities with electric arc or chemical heating, although the reproduction of the Reynolds numbers in these facilities is often in question.

Diagrams of high-enthalpy facilities discussed here are shown in Fig. 1 (A — air reservoir, D — diaphragm, P — heavy piston, Φ — mixing chamber, N — aerodynamic nozzle, T — electric current generator, HPD — hydropneumatic drive, T — throttle, B — booster, V — check valve, [ACC — adiabatic compression chamber]).

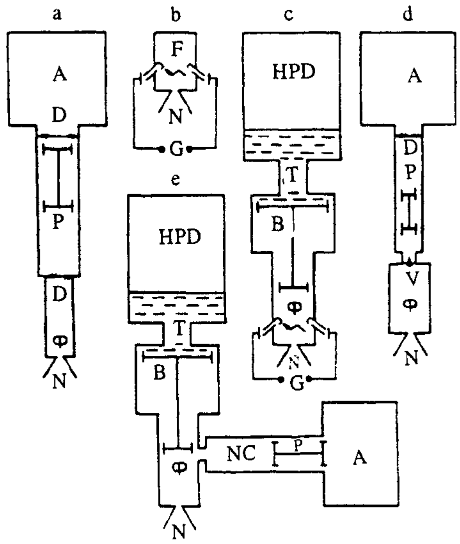

Fig. 1

In a shock tunnel, a shock wave is created by breaking a diaphragm which separates a section of propellant gas at a pressure of up to thousands of atmospheres from the section with the working gas at a pressure of several tenths of an atmosphere. Aerodynamic shock tunnels often use a gas which is compressed a second time by the shock wave that reflects from the end of the tunnel, which is fitted with an aerodynamic nozzle. A modification of an ordinary shock tunnel is the heavy-piston shock tunnel, in which the propellant gas is compressed by the kinetic energy of a piston, which [in turn] is driven in a barrel by compressed air. A diagram of this type of facility is shown in Fig. 1*a*. When helium is used as the propellant, a stagnation temperature can be obtained behind the reflected shock wave that provides a discharge flow on the order of 8.5 km/sec. A typical example of such a facility is the HEG heavy-piston shock tunnel at Göttingen [8].

Shock tunnels create acceptable conditions for modeling several phenomena which arise during flow around aircraft and their engines. However, too low a Reynolds number, the heavy contamination of the flow with air dissociation products, and too short a working time (several milliseconds) limit the problems which can be solved in such facilities. Evidently the gas behind the bow shock is almost completely dissociated [only] at the highest part of the flight trajectory. Because of these deficiencies, shock tunnels are simply not suitable for solving problems with hypersonic-combustion ramjets. Pieces of broken diaphragm create serious problems when they strike the model and damage it, especially the diagnostic transducers.

Prospects of improving shock tunnels is rather poor, because attempts to extend operating conditions by increasing the shock tunnel dimensions lead to fundamental difficulties due to the increased thickness of the boundary layer behind the shock wave, which decreases parameter values and makes the flow created in the working section unacceptable. In order to combat air dissociation, it is possible in principle to use a flow which arises behind a direct shock wave after it jettisons the boundary layer [9]. Thus, a shock wave at Mach ~7 has a static temperature behind it of ~2500 K, while the stagnation temperature reaches 4200 K. However, such conditions last for a shorter time, and the supersonic nature of the flow behind the shock wave ($M \approx 2.2$) requires a large-diameter shock tunnel due to a fundamental relationship with the diameter of the working section in this case, which makes it difficult to make the shock tunnel walls strong enough.

The same strength problem arises in attempting to increase Re in a shock tunnel. Even now the pressure behind the reflected wave reaches 2,000-3,000 atm in a 15-20 cm diameter shock tunnel. It is difficult to chose a material which can withstand multiple pulsed loads with high thermal and oxidizing reactions. Therefore, significantly increasing the initial pressure of the working gas is hardly possible; consequently, its initial density will remain on the order of half atmospheric density under normal conditions. Because the degree of compression in a shock wave does not exceed 8 (considering the change in C_p/C_V) and does not exceed 4.5 behind a reflected shock, the increase in Re due to an increase in density is limited to a 36-fold compression of the working gas from its initial state. This limit has already been reached in existing shock tunnels.

Another tunnel type, designed for flows up to Mach 20, is the pulsed tunnel (Fig. 1b) which has an arc chamber that has been used in various modifications for almost 35 years. The pulse duration of these tunnels (50-200 msec) lies between the shock tunnel (≤ 5 msec) and the Ludwig tunnels (≤ 1 sec). Because the gas in these tunnels flows from the mixing chamber under isochoric conditions, the gas flowing around the models is only quasi-stationary. The stagnation enthalpy in pulsed tunnels is somewhat less than in shock tunnels, but is sufficient for studying many flow features of a real gas [10, 11]. Even though these tunnels are simple and extremely compact, they can produce rather high Reynolds numbers.

Many researchers in the world's various aerodynamic centers have turned their attention to the possibilities of pulsed tunnels. To date results have been obtained on a wide spectrum of both fundamental and applied hypersonic aerodynamic problems [10]. This is probably the reason why the F−4 facility was put into operation at ONERA in 1991 to study flow fields around the HERMES aerospace plane [12]. The F−4 facility has the same operating principles as an ordinary pulsed tunnel. The basic difference is that it uses a 15-ton 6000 rpm flywheel to accumulate the energy for an AC generator which can supply up to 160 MW to the arc discharge. When the stator is switched to the electrodes in the mixing chamber, the arc discharge can produce a pressure up to 2000 atm and a temperature up to 8600 K [5]. When the required conditions are attained, the high-enthalpy flow starts through the nozzle to produce a 50-100 msec quasi-stationary flow around the model. Thus this tunnel also has the usual deficiencies of a pulsed tunnel:

— continuous changes in temperature, stagnation pressure, and Reynolds number during operation;

— contaminated flow from the products of air dissociation and electrode erosion; and

— nonuniform distributions of the stagnation temperature.

The fall-off of parameters in the mixing chamber of the pulsed tunnel during the test leads to a constant change in forces and moments on the model, first due to the reduction in the pressure, density, and velocity of the flow in the working section and second due to the change in the Reynolds number of the flow around the model. The allowable pressure drop-off rate is determined by the time required to establish the flow and the time required to measure and collect the data. If the total is 10 msec, for example, and 5% accuracy is required, then the pressure in the mixing chamber should change less than 5% in 10 msec when the errors in the measurement system are taken into account. The literature contains almost no discussion of stability requirements on Re. One can only assume that, when the flow around the model is close to the laminar-turbulent transition flow in the boundary layer, a twofold reduction in Re can qualitatively change the flow pattern around the model and consequently the force distribution on it.

The relative change in Re during the flow from the mixing chamber is given by

$$Re_t/Re_0 = (\rho_{\Phi t}/\rho_{\Phi 0})(a_{\Phi t}/a_{\Phi 0})(\mu_0/\mu_t),$$

where μ is the gas viscosity in the working section; ρ_Φ and a_Φ are the gas density and sound speed in the mixing chamber; and the subscripts 0 and t indicate parameters at the start and at time t.

Because the ratio of the viscosities is proportional to the square root of the ratio of the temperatures (i.e., the sound speeds) in the mixing chamber to a first approximation, it can be seen that the ratio of the Reynolds numbers is roughly proportional to the density ratio.

Adiabatic evacuation of a vessel of volume V_Φ by critical flow through an orifice of area F* is given by the formulas [13]

$$T_{\Phi t}/T_{\Phi 0} = (1 + At)^{-2},\ P_{\Phi t}/P_{\Phi 0} = (1 + At)^{-2\gamma/(\gamma-1)},$$
$$\rho_{\Phi t}/\rho_{\Phi 0} = (1 + At)^{-2/(\gamma-1)},$$

and the relative pressure drop over a time Δt is given by

$$\Delta P_{\Phi t}/P_{\Phi t} = -2\gamma A[(\gamma - 1)(1 + At)]^{-1}\Delta t.$$

Here $A = (\gamma - 1)/2t_i$, $\gamma = C_p/C_v$, and $t_i = V_\Phi/\{[2/(\gamma + 1)]^{(\gamma+1)/2(\gamma-1)}F^*a_0\}$ is the time for isobaric isothermal discharge of the gas, i.e. the time over which the gas is extruded by the piston from the mixing chamber at constant pressure with no temperature losses. For an ideal polytropic gas with $\gamma = 1.4$ and a molecular weight of $\approx$29 g/mole at M = 10 (the ratio of the areas of the working section and the nozzle throat is F/F* = 536), the following formula is valid:

$$t_i = 58{,}9V_\Phi/(d_n^2\sqrt{T_0})$$

where t_i is in msec, V_Φ is in cm^3, d_n is the throat diameter (cm), and T_0 is in K. For M = 15 (F/F* = 3755), t_i increases by a factor of 6.4, but for M = 20 (F/F* = 15,377) t_i increases by a factor of 26. During the test, the flow parameters in the pulsed tunnel change continuously according to the above formulas.

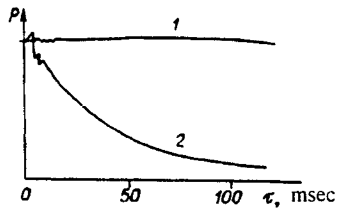

Fig. 2

The IT−302M hypersonic wind tunnel, which has operated at the Institute of Theoretical and Applied Mechanics of the Siberian Branch of the Russian Academy of Sciences (ITPM SO RAN) since 1985 [14, 15] is fundamentally different from traditional pulsed tunnels. Along with the usual elements of a pulsed tunnel, this facility is equipped with a flow-stabilization system which provides almost constant stagnation parameters over the whole test time (≈ 0.1 sec) for a relatively small mixing chamber volume (≈ 8 dm^3). For this purpose the facility is equipped with a booster, which forces the working gas from the arc chamber at a constant pressure. The operational principle of the gas source in this wind tunnel is clear from Fig. 1c. In this case the electrical current generator is a 1.75 MJ capacitor bank.

A typical pressure trace in the mixing chamber during booster operation is shown in Fig. 2 (curve 1). It can be seen that it is constant over a time $t_i \approx 0.1$ sec with an error of $\pm 1\%$. A pressure trace with the booster switched off is shown for comparison (curve 2).

During outflow from a constant volume, the pressure initially drops 2% over the time

$$\Delta t_p = 0.02 t_i/\gamma \approx 0.014\, t_i.$$

An n-fold drop in the Reynolds number occurs over a time

$$\Delta t_{Re} = 2t_i(n^{(\gamma-1)/2} - 1)/(\gamma - 1).$$

If, for example, $n = 2$, then $\Delta t_{Re} \approx 0.75\, t_i$.

According to these formulas, for $\gamma = 1.4$ and for typical values of the parameters of a facility of analogous dimensions ($V = 8$ liters, $F^* = 0.8$ cm^2, $a_0 = 1100$ m, and $A \approx 1.4$ sec^{-1}), the same initial twofold drop in Re and 2% drop in the pressure would occur over roughly 2 msec if the booster was not operating over the same 100 msec. As we see, using the booster gives the IT−302M facility convincing advantages over other pulsed tubes; however all the disadvantages related to using an arc discharge and to too low a Reynolds number remain in force. Nonetheless a series of interesting and fundamentally important investigations were conducted in this tunnel, including (the first apparently ever) hypersonic combustion of hydrogen [16].

For moderate model dimensions, the required Reynolds numbers cannot be attained without using high working-gas pressures. The high initial densities, by reproducing the boundary layer state in terms of M and Re, make possible a complete modeling of combustion in the flow in terms of the binary and tertiary parameters ρL and $\rho^2 L$, which determine the evolution rate of the chemical energy that heats the flow [17].

Diagrams of the entropy S vs. the enthaply H and of M vs. Re_1 (Re_1 is the Reynolds number per meter) computed for the gas condensation curve for specified initial temperatures and pressures [18, 19] serve as a physical basis for designing these high-enthalpy facilities. These diagrams are shown for nitrogen in Figs. 3 and 4. They show that the transition to high initial pressures leads to an additional increase in the enthalpy, which is equivalent to increasing the stagnation temperature by 1000-2000 K, the density and gas flow rate, and, as a result, M and Re. This makes it possible to model these parameters completely for moderate mixing-chamber temperatures and tunnel dimensions.

Decreasing the temperature decreases flow contamination by wall particles and by air dissociation products, which is especially important in researching flows in models of hypersonic-combustion ramjets, where contamination can change the kinetics of fuel combustion. An exact thermodynamic calculation of the chemical equilibrium of air shows that noticeable O_2 dissociation starts at 2200 K at atmospheric pressure; increasing the pressure to 3000 atm shifts this temperature by 800-1000 K. At the same time, using ultrahigh pressures — along with technical difficulties, which fortunately can be overcome — has two serious limitations. First, the boundary layer on the models become very thin, which significantly complicates the diagnostics to measure it. Second, the small linear dimensions leads to too large a difference in Re_1 between the model and

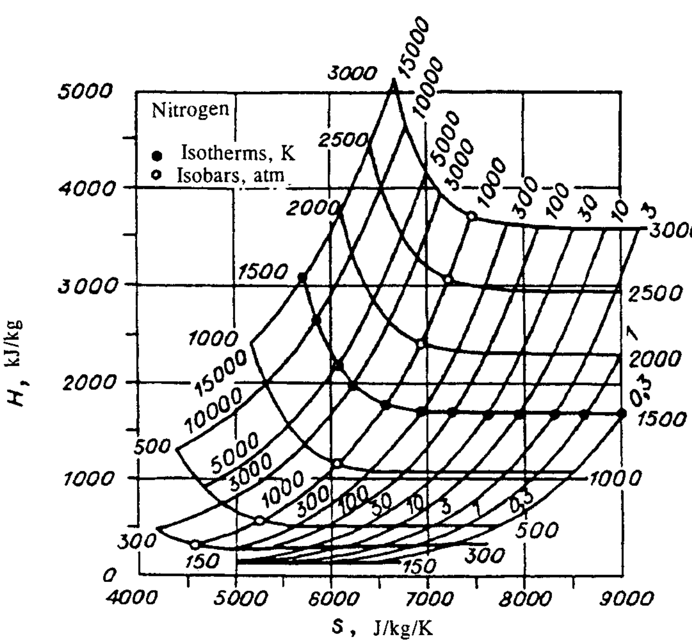

Fig. 3

reality. In this case it is difficult to model the level and scale of free-flow perturbations [20, 21]. There is reason to believe that development of optical methods for qualitatively visualizing the flows can overcome the first limitation. The second limitation can remain a fundamental one for some research.

Too high a temperature in the mixing temperature can be fatal in modeling combustion processes, because the chemical composition of the air entering the combustion chamber will not correspond to real conditions. The problem is that, in order to have a normal fuel combustion cycle, the air must have a low enough static temperature to prevent dissociation before it enters the combustion chamber.

For illustration, results of an exact thermodynamic calculation [17] of the chemical equilibrium of a mixture close to that of air ($4N_2 + O_2$) is shown in Fig. 5. Figure 5a shows the molar fraction of molecular and atomic oxygen (dashed and solid curves) as a function of temperature for various pressures. For the same variables, Fig. 5b shows the molar fraction of NO (solid curves) and the specific enthalpy for air dissociation, ratioed to an enthaply that corresponds to a velocity of 8 km/sec (dashed curves).

These diagrams show that the molecular fractions of O and NO are roughly 5% and 4% at atmospheric pressure and 3000 K. If this air is used to ignite hydrogen, these radical concentrations lead to a substantial change in the chemical reaction kinetics at low temperatures [22]. Increasing the pressure at this temperature greatly lowers the atomic oxygen concentration, but lowers the nitrogen oxide concentration hardly at all.

When the air is expanded in a nozzle its normal composition is restored as a result of recombination reactions with a third material M. Integrating the chemical kinetic equations of the type $O + O + M \rightarrow O_2 + M$ gives the recombination time as a function of the remaining degree of dissociation α [17]:

$$t = t_0\{\ln[(1 + \alpha)\alpha_0/(1 + \alpha_0)\alpha] + 1/\alpha - 1/\alpha_0\}\,..$$

where $\alpha_0 = \alpha(t = 0)$. The characteristic time t_0 is expressed in terms of the kinetic parameters as follows: $1/t_0 = 4n_0^2AT^n\cdot\exp(E_a/RT)$, where n_0 is the concentration, mole/cm^3. The usual constants for this reaction [23] ($A = 4.68\cdot10^{15}$, $n = -0.28$, and $E_a = 0$) give $t_0 = 31.6$ μsec for p = 1 atm. According to this formula, relaxation from $\alpha = 1$ to $\alpha = 1/2$ occurs over 28 μsec, but to $\alpha = 10^{-3}$ over 30 msec. At a velocity of ≈3 km/sec, gas particles cover move almost 90 m along the nozzle in this time! Increasing the pressure rapidly decreases t_0, because it is proportional to $1/p_0^2$. An analogous result, including chemical reaction kinetics, has been obtained from a numerical calculation of the gas motion in an expansion nozzle [6].

Thus, the relaxation time of the chemical mixture during expansion in a nozzle is too long, and the air composition differs significantly from real conditions. High pressure gives some improvement in the flow quality, by decreasing the oxygen

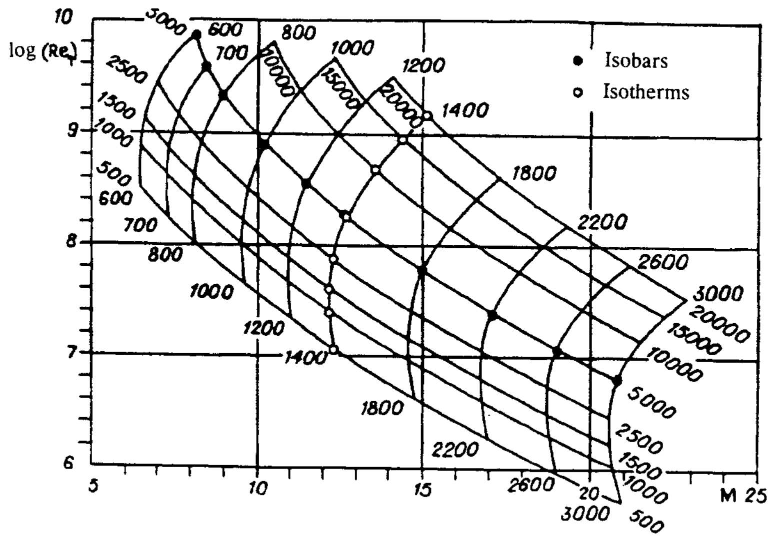

Fig. 4

dissociation in the mixing chamber, but the nitrogen oxide concentration remains too high. Thus, using too high a temperature in the mixing chamber substantially changes the molecular composition of the working gas.

On the other hand, the threshold nature of combustion reactions allows ignition retardation to be neglected if the temperature of the fuel mixture exceeds some value. Fuel in a hypersonic-combustion ramjet is ignited one way or another after gas is compressed in a shock or a system of shocks in the air collector; therefore the conditions for initiating the reaction in some sense are close to those which are observed in spin detonation of gases [24].

Actually, the temperature of the shock-compressed gas decreases from roughly 3000 K to 1100 K behind the stationary curved front of a spin detonation wave as it moves from the break along the front. At the break, where the flow velocity is almost perpendicular to the shock, the flame front coincides with the shock front, and they cannot be distinguished by optical methods that have a 0.2-0.5 mm resolution. The flames start to separate from the shock in a section where the temperature drops to 1500-1700 K. An analogous conclusion can also be drawn from direct measurements [25, 26] of the ignition retardation in hydrogen-oxygen mixtures under conditions close to those expected at the intake to the combustion chamber of a hypersonic-combustion ramjet. In this temperature range the delays are on the order of 1 μsec and die out rapidly as the temperature rises.

If the flow velocity in the combustion chamber is 1000-3000 m/sec, the gas particles are displaced by 1-3 mm in this time. Therefore, if ignition starts at these temperatures and above, the ignition retardation becomes small, and the start of combustion is controlled only by mixing processes which depend on the Mach and Reynolds numbers, and these numbers are modeled completely at high initial pressures. Then the relatively low temperature in the mixing chamber is no barrier to research.

An alternative approach that gives much higher gas density, less contamination, and better flow homogeneity than the shock tunnel and the pulsed tunnel, is to feed the wind tunnel with an adiabatic compression chamber. This tunnel has a free heavy piston, like the heavy-piston shock tunnel, but the piston compresses the working gas without an intermediate energy carrier. In the adiabatic-compression chamber, heating occurs as a result of isoentropic compression, which gives much more uniform stagnation temperature distributions than does a pulsed tunnel and can attain a higher gas density. The adiabatic compression chamber falls behind shock tunnels and pulsed tunnels in stagnation temperatures, but it gets better Reynolds numbers than all other systems, due to the high densities and pressures. One of the first facilities operating on this principle is the Long Shot system [27], which provides a nitrogen flow with an initial pressure of 4000 atm and a temperature to 2350 K.

One of the problems with a shock compression chamber with a heavy piston is that the piston rebounds after it stops at the point of maximum compression. For maximum working gas pressures on the order of 2000-4000 atm, most of the piston deceleration (to bring the pressure from half maximum to maximum) occurs in the last several tens of microseconds [28]. Piston rebound and pressure drop-off will occur over the same time; therefore either the gas must be trapped in the mixing chamber or the piston must be stopped at the point of maximum compression. Either method is technically difficult. The first case requires erosion-resistant valves with both a large cross section and rapid closure — requirements which are inherently contradictory; the second requires figuring a way to stop a very large force — $\approx$2-4 ton/cm^2 — in the piston barrel.

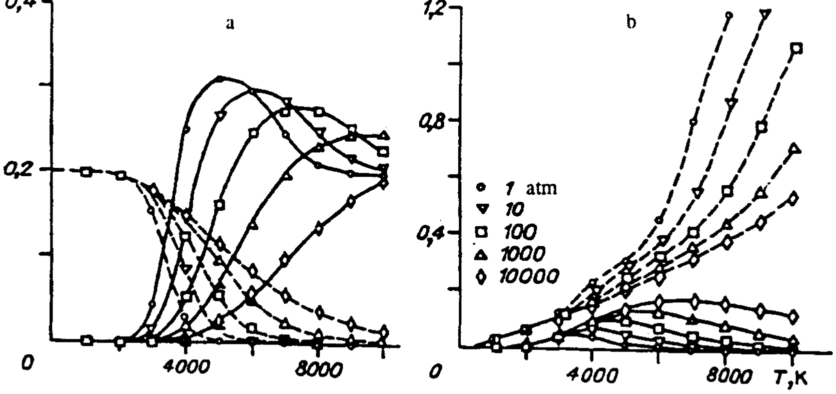

Fig. 5

In the Long Shot adiabatic compression chamber [27], the mixing chamber is equipped with a system of 48 valves which prevent reverse flow of the gas into the piston barrel. The gas flowing through the valves with high pressure, temperature, and velocity during the final compression stage rapidly erodes and destroys the valves. Therefore Long Shot operates only with nitrogen and is completely unsuited to model a hypersonic-combustion ramjet. The pressure in the working section of the tunnel (diameter 315 mm) drops by a factor of 2 in 10 msec at $M = 15$. Nonetheless it was used to solve several external-flow problems for hypersonic aircraft, including the Space Shuttle.

A much larger adiabatic compression chamber has been designed and operated at the Central Scientific Research Institute of Machinery Manufacture and Metalworking (TsNIIMASh). This facility has a piston braking system and a throttle system which stabilizes the outflow parameters [29, 30].

In 1969, work on designing high-pressure testing units with high-pressure mixing chambers was started at IGiL and KTI GIT of the Siberian Branch of the Russian Academy of Sciences. Here the problem was to provide the highest possible purity of the working gas (nitrogen or air) and constant flow parameters over 40-150 msec while completely modeling the Mach and Reynolds numbers. The approach was to divide the gas preparation into two cycles. First the gas was sprayed into the mixing chamber; then it was adiabatically compressed using a booster to the desired parameters. During the test time, the booster draws gas from the mixing chamber to provide constant pressure.

Using part of the mixing chamber volume for compressing the working gas to the desired conditions avoided having gas at maximum conditions flow rapidly over elements of the facility. The only part that was subjected to this flow is the nozzle throat; therefore erosion in facility components in contact with the hot gas is sharply reduced, as is contamination of the gas itself. Evidently this bifunctional booster was first actually used to generate dense hypersonic flows in the A–1 facility, which was designed for pressures to 10,000-15,000 atm and temperatures to 2000 K [31].

A diagram of the facility is shown in Fig. 1e. The first stage of the facility is an adiabatic compression chamber with a barrel about 2 m long and with an internal diameter of 50 mm, which has a free heavy piston to compress the gas to about 2000 atm and 1300 K. As can be seen from Fig. 3, the isotherms start to curve upwards at these pressures. The booster provides a roughly twofold volumetric compression to produce a pressure of 10,000-15,000 atm.

The piston is equipped with a wedge system which stops it at the point of maximum compression with no rebound, and then unwedges the piston and allows it to move freely after the pressure is removed in the mixing chamber [32, 33]. This avoids the use of check valves (such as those used in the Long Shot system) and provides a large direct flow cross section between the adiabatic compression chamber and the mixing chamber, which reduces heat losses and gas contamination.

After the specified pressure is attained in the first stage, the booster is automatically switched on. Initially the booster piston cuts off the adiabatic compression chamber from the mixing chamber; then it compresses the gas and expels it through the critical section of the nozzle. The gas is compressed so fast that less than 10% of the initial gas goes through the nozzle throat during the compression cycle; therefore no diaphragms or valves are required to close off the nozzle during the compression. A special damping device controls the nozzle movement and the transition from one stage to the other.

The facility is equipped with a momentum compensator which activates when the piston decelerates; therefore no foundation is required. The barrel displacement does not exceed 0.2 mm, and is within the elastic limit.

Other complex problems were solved in designing the A–1 facility besides stopping the piston and compensating the momentum: compensating the strain in the booster piston and the mixing chamber cylinder, providing reliable seals at contact pressures up to 15,000 atm and mutual surface displacements at velocities close to 1 m/sec, etc. The A–1 facility has operated

TABLE 1

Facility	p_0, atm	T_0, K	$Re_1 \cdot 10^{-6}$	V_Φ, dm^3	d_c, mm	t_i, msec	t_p, msec	t_{lit}, msec	M_{lit}	$Re_{lit} \cdot 10^{-6}$	Working gas	References
Long Shot	4000	2350	16	0,315	360	1,74	0,024	5—10	15—20	5—1	Air	[5, 26]
F-4	2000	8600[1]	< 1	—	670	—	—	17	9	0,24	Nitrogen	[5, 11]
HEG	1800	10000	< 1	≈8,3[2]	700	10[3]	0,14	1	7,5	0,5	»	[5, 7]
TsNIIMASh	2000	2600	4,5	≈25,0[2]	800	42	0,59[5]	500	10—20	50—0,5	»	[28, 29]
MT-1	3000	2500	7	6,0	300	30[4]	30	40—100	6—18	250—8	»	[33]
IT-302M	1000	3000	1	8,0	300	126	126	100	5—15	200—7	»	[14]
A-1	10000	2000	100	0,040	20	16,3[4]	16,3	40—200	8—25	700—3	Nitrogen[6]	[32]

Notes. 1) This temperature can hardly be attained at this facility. 2) Compressed-gas volume calculated from facility dimensions in the literature. 3) Expulsion time calculated from formulas presented here for an ideal gas with $\gamma = 1.4$. 4) Because these facilities overcompress the gas, the mixing chamber volume was taken to be half the value shown in the literature for computing t_i. 5) The volume is decreased by a factor of 8 to 25 if suggested [28, 29] multistage throttling is not used. 6) The facility operates on air at pressures up to 6000 atm.

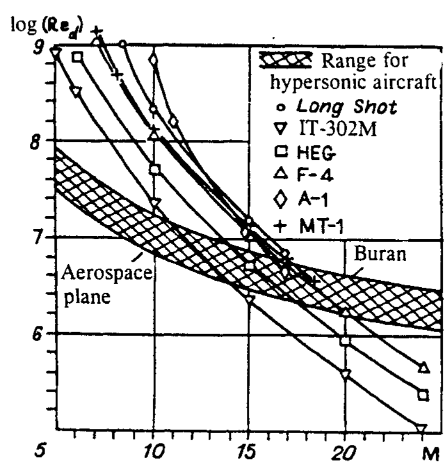

Fig. 6

up to 10,000 atm with nitrogen and up to 6,000 atm with air. When the feed to the booster is in use, the pressure can be extended to 15,000 atm.

The A – 1 facility has operated since 1975 without failure. Research programs have been conducted on it to select materials for the critical nozzle cross section and to develop methods for measuring the flow velocity directly. The latter will allow direct measurement of the heat losses to the mixing chamber walls and to the entire flow path. Toepler motion pictures show good flow uniformity. No erosion of simple bodies from contaminating particles has been observed, even after dozens of tests. Pressure traces are constant within the limits of the test time, which ranges from 20 to 250 msec, depending on the outlet diameter of the nozzle (50-20 mm). The A – 1 design can be scaled up by a factor of 5-8 [33].

The second facility, MT – 1, has an initial mixing chamber volume of close to 6 dm^3 and is designed to obtain pressures to 3000 atm at a temperature of close to 2500 K. The working gases are nitrogen, air, and helium. The working time is 40-100 msec, and the nozzle outlet diameter is 0.3-0.5 m. This facility uses two symmetric boosters which are driven by eight hydropneumatic drives. The main boosters are fed by a symmetric intersecting method which rigidly synchronizes the colliding motion of the pistons. The gas is sprayed into the mixing chamber through a Cowper heater at a pressure up to 250 atm and a temperature of 900-950 K; it is then compressed to the required conditions and exhausted through the nozzle. As in the A – 1 facility, the MT – 1 facility has a damping system that suppresses oscillations and controls the motion of the booster pistons. When the specified position is attained, a special 1-msec valve automatically opens the critical cross section of the nozzle [34].

The basic parameters of several operating high-enthalpy facilities and some that are in the process of being checked out are presented in Table 1. Besides the maximum gas pressure p_0 and temperature T_0, the table also shows the unit Reynolds number Re_1, the mixing chamber volume V_Φ (where it could be estimated from published data), the nozzle outlet diameter d_n, the extrusion time t_i calculated to include the effects of a real gas [19], the time t_p for the pressure to fall by 2% for M = 10, and also the range of Mach numbers M_{lit}, the working time t_{lit}, and the Reynolds numbers Re_{lit} presented in the literature. It should be noted that the values of t_{lit} are rather nominal, because only facilities with boosters (A – 1, IT – 300M, and MT – 1) and with shock tunnels can deliver constant flow parameters over the working time. For facilities without extrusion devices, the time for the Reynolds number to drop in half is roughly 25% less than t_i.

Figure 6 shows the regions modeled by facilities with boosters in comparison with the regions reproduced by other facilities whose parameters are shown in Table 1. Unfortunately data on Reynolds numbers in various references often do not agree. Therefore the comparison is made based on literature values of the maximum temperature and pressure in the mixing chamber and the diameter of the working section of the nozzle and the diagrams shown in Fig. 4; i.e. the Reynolds numbers are calculated from this diameter ($Re_d = Re_1 d_n$). We note that, for a given facility, the product of the working time and Re^2 is a constant as the nozzle diameter changes, while proportionately increasing the dimensions leads to a proportional change in Re and the working time. From the diagram it can be seen that an adiabatic compression chamber with a booster allows complete modeling of M and Re if the discharge parameters are constant.

In analyzing the prospects of improving the modeling of hypersonic aircraft, it should be noted that attempts to reproduce the flow velocity (stagnation temperature) unavoidably distorts the chemical composition of the working gas. Suggestions [4] to solve the high-velocity problem by using magnetohydrodynamic systems to accelerate the gas can hardly be considered promising for several interrelated reasons. In order for an accelerating magnetic field (or the current in the magnetic field) to have a strong enough effect on the flow, the magnetic Reynolds numbers must be large, which requires either using a gas with a high electrical conductivity — i.e., using very high temperatures, relatively low densities — or introducing easily ionized additives. This makes it impossible to model the combustion, let alone low values of Re.

The literature contains suggestions on increasing the total enthalpy and also the flow velocity by using chemical energy. If we stick with combustion heaters, which use hydrocarbon fuel which heavily contaminates the gas, we are looking at using nitrous oxide [29, 35]. Actually, it is tempting to use a reaction of the type $2N_2O + 2N_2 \rightarrow 4N_2 + O_2$. This reaction gives a chemical composition sufficiently close to that of air, and an additional ~1 MJ of energy per kilogram of mixture. According to Fig. 3, this increases the stagnation temperature by 1000 K; however the accompanying problems of dissociating the working gas remain unresolved. The same effect (with improvements in Re values) can be obtained by increasing the air pressure to 8000 atm, without risking the use of an explosive mixture.

Suggestions to solve hypersonic aerodynamic problems by using ballistic orbits at re-entry velocities [4] appear unrealistic. It has been impossible to attain 12-15 km/sec velocities for models ~20 cm in diameter and ~1 m long by using either light-gas cannons or electromagnet rail guns, not to mention the thought of ordnance with an accelerating section of up to 300 m and a vacuum volume 600 m long and 3-6 m in diameter. A rail gun has been able to accelerate a mass of ~1 g to a level of ~5 km/sec [36]. This limit has not been exceeded recently [37]. Light-gas cannons have given somewhat better results, but even they have only accelerated compact plastic masses of several grams to velocities up to 10 km/sec.

However, there is a fundamental limitation. Estimates show that even with the above system dimensions, accelerations on the order of several thousand g are required to attain velocities of 12-15 km/sec, which are many times higher than aerodynamic models can withstand if they have air intakes, wings, and control surfaces. If accelerations are reduced to allowable limits, the accelerator would be proportionately larger — an almost insoluble problem.

Thus, facilities which have constant parameters during the test time, and the highest flow uniformity and quality are the most promising for design problems for a hypersonic aircraft with a hypersonic-combustion ramjet. In the time range of 20-200 msec, there evidently are no suitable alternatives to using two-stage facilities with flow boosters which use part of the mixing-chamber volume to compress the gas to the maximum conditions. For pressures up to 3000 atm, it is better to use symmetric designs, such as MT−1, because they have less heat losses in compressing the gas and discharging it, when all else is equal. At higher pressures, A−1 designs can be used with automatic compensation of the deformation of the booster piston and with special seals that can operate at these pressures. The best first stage is adiabatic compression chamber with a wedged piston and momentum compensation, because it makes maximum use of the mixing chamber and gives a high-purity working gas.

Another worthwhile system would have a rather large adiabatic compression chamber with an interchangeable second stage — of the MT−1 type (for working pressures up to 3000 atm) or the A−1 type (for higher pressures). This facility would cover all M and Re ranges required to model hypersonic flight at stagnation enthalpies corresponding to temperatures up to 3500 K. Experience in designing and operating the A−1, IT−302M, and MT−1 facilities shows it is possible to use these principles to design facilities with seals sufficient to overcome the high-pressure diagnostic problems mentioned above.

Today shock tubes with a solid piston are the most acceptable for modeling external aerodynamic flows around a hypersonic aircraft, where the main requirement is to model the stagnation temperature accurately with a limited amount of flow contamination. Here problems related to valves, momentum compensation, and stopping the piston can be solved by methods used in designing the A−1 facility.

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