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16. Abstract

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The brochure is intended for a wide readership interested in the modern problems of rocket-space technology.

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The provision of heat protection for various elements of space flight apparata has great significance, particularly in the construction of manned transport vessels and orbital stations. This brochure presents a popular explanation of the methods of heat protection in rocket-space technology at the current stage as well as in perspective.

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With the beginning of space exploration, man has been /3
confronted with a space which is unknown, surprising, and often seemingly paradoxic. In actuality, does it not seem paradoxical, for example, that on Venus the optical specifics of the atmosphere, generally speaking allow an observer to see ... the back of his own head. Here is another example. If the space apparatus performing an orbital flight decelerates slightly, its speed ... increases.

Rocket-space technology is an exotic technology which has no analogies on Earth. Many new and complex problems arise during its design, which lead finally to the appearance of original designs and unusual technical solutions. One such problem consists of the necessity to ensure the given level of temperatures of various elements of rocket-space flight apparata. The success in solving this problem determines the preservation of the material part of rockets, rocket engines, and spacecraft of various classes and functions.

It would not be a great exaggeration to say that it is primarily due to the emerging necessity to solve this problem that the science of transfer of heat -- heat transfer is being developed at the modernstage, new fireproof, heat resistant and ablation materials are being developed, practical methods of removing heat from heated bodies are being improved. In this brochure we will attempt to examine the essence of heat problems arising during the design of various elements of rocket-space technology, and will show means for their solution.

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COLD OR HOT SPACE?

If a body does not have internal heat sources, then its temperature is determined by the conditions of the surrounding environment in which it is located. Therefore, we will attempt first of all to understand what the conditions are in space. /4

We know from physics that the temperature is characterized by the rate of thermal movement of body particles, and the environment (or system): the greater this rate, the higher the temperature. On earth at room temperature, the air molecules move with a speed of about 500 m/sec, experiencing up to 5 billion collisions with each other within 1 sec. As the air density is reduced, its molecules collide with each other less frequently (as specialists say, the distance of their free run increases), their speed, and consequently also their temperature, becomes ever higher.

In the Earth's atmosphere, more complex processes take place, and the temperature of its layers, as follows from the table, is not directly proportional to the density of the air (or the concentration of its particles).

Table

Change in atmospheric parameters with altitude

Altitude, km	Pressure kG/cm^2	Temperature, °C	Particle concentration, cm^3	Air composition
0	1	+15	$2.5\cdot10^{19}$	N_2, O_2, Ar
11	0,2	−56	$4.5\cdot10^{18}$	N_2, $O_2(O_3)$, Ar
20	$5\cdot10^{-2}$	−56	$2\cdot10^{18}$	То же
30	10^{-2}	−42	$4\cdot10^{17}$	»
46	10^{-3}	0	$3\cdot10^{16}$	»
64	10^{-4}	−33	10^{15}	N_2, O_2, Ar
79	10^{-5}	−85	10^{14}	То же
102	10^{-6}	−60	10^{13}	N_2, O_2, O
200	10^{-10}	+630	10^{10}	N_2, N, O, O^+
800	10^{-13}	+3040	10^{8}	O, O^+, H
6600	10^{-16}	10^{3}—10^{4}	10^{3}	H, H^+
Above 22 000	10^{-17}	10^{5}—10^{6}	10^{1}—10^{2}	H^+, He^{++}

Up to an altitude of 11 km, the temperature drops and then remains constant to altitudes of 11-25 km. This is associated with the fact that at these altitudes there is also a strong influence on the condition of the particles exerted by convection and radial equilibrium of the moving air masses. Absorption of the energy of solar radiation by the atmospheric ozone in the ultraviolet portion of the spectrum leads to a temperature increase up to an altitude on the order of 50 km. At greater altitudes (up to 80 km) /5 in connection with the reduced ozone concentration there is a certain reduction in the temperature of the air particles, and at even greater altitudes an increase in the temperature is observed due to the dissociation and ionozation of oxygen under the action of the Sun's ultraviolet rays. At an altitude of 200 km, where the air density is relatively low and the speeds of air particle movement are great, their temperature already comprises in excess of 600°C, at an altitude of 800 km -- over 3000°C.

Thus, from what has been said it would seem that we must conclude that the cosmos is "hot", and the designer consequently must take measures to protect the space apparatus against the destructive action of high temperatures. However, if we take a plate and place it in cosmic space in such a way that no heat flows are directed to it (for example, placing it away from heavenly bodies, planets, etc.), then its temperature in the course of time will be close to absolute zero and will comprise only 4 K. This experiment clearly demonstrates that the cosmos is "cold".

What, then, do we have? The temperature of the air particles in space is rather high, while the temperature of a body placed in this "hot" environment turns out to be low. The paradox is obvious, but it is a seeming paradox. This phenomenon is explained quite simply. Due to the low density of "cosmic air", its molecules collide very rarely with a body placed in their midst, and as a result, despite their high temperature, they are unable to impart to it the amount of energy which would be required for a noticeable increase in its temperature. Specialists in this matter say that heat transfer is low in space due to natural convection.

The low temperature of a body in cosmic space in no way bespeaks the fact that the designer is faced with a single thermal problem -- to protect the space apparatus against supercooling. As strange as this may seem at first glance, specialists must simultaneously solve also a second problem -- the protection of the material part against overheating. The reason for this, however, is not associated with the high kinetic temperature of the air molecules. It is conditioned by the fact that in cosmic space there are heat sources which warm the bodies placed in it. The strongest of these is our /6 star. In one hr. it sends approximately 1200 kcal to an area 1 m^2 in size, located perpendicular to its rays. The density of the solar heat flow depends on the distance from the Sun. For Mercury, for example, it comprises 8000 kcal/m^2hr, for Mars -- 525 kcal/m^2hr, for Jupiter -- 45 kcal/m^2hr, for Pluto -- 0.6 kcal/m^2hr.

The solar heat flow, reaching the Earth, is partially reflected from its surface and atmosphere: water reflects 5% of this flow, snow -- 77, sand -- 24, and structures -- 9%. The value of the reflected flow greatly depends on the time of the year. In July, when relatively slight cloudiness is characteristic for the atmosphere, 32% of the solar flow is reflected. In October, when the clouds retain a greater amount of the heat, this value increases to 52%. Specialists believe that on the average, the reflection from the Earth's surface and the clouds comprises on the order of 40% of the solar heat flow.

The remainder of this flow (approximately 60%) is absorbed by the Earth and is then irradiated by its surface into the surrounding environment. The density of the flow of the Earth's own irradiation is relatively low -- in total with the reflected flow it comprises on the average (along the Earth's surface) 35% of the solar heat flow. With increased distance from the Earth's surface, these two flows rapidly dissipate, although they may have a noticeable effect on the thermal regime of flight apparata even at relatively great altitudes.

In space flights at relatively low altitudes, the aerodynamic heating of design parts of the apparata may be noticeable. Fig. 1 presents the density of the heat flow coming to a plate moving at different distance from the Earth's surface with primary cosmic speed. It is easy to see that already at an altitude of 200 km the density of the aerodynamic heat flow becomes an order less than the density of the solar flow, and further rapidly decreases with increased altitude.

Usually located on the body of a space apparatus are various devices, instruments, mechanisms, etc., which operate under conditions of open space. Their thermal regime may be determined also by certain additional heat sources. Thus, for example, they may receive solar heat flow reflected from /7
solar batteries, from the body and other parts of the constructions, they may receive flows radiated by highly heated elements, etc.

These, in short, are the external heat sources whose action leads in the general case to heating of the construction of a space apparatus and its external elements.

The thermal regime of space apparata is determined to a significant degree also by their internal heat sources.

Various instruments, power installations, means of control and installation, etc. are located aboard these vessels which emit thermal energy during their operation which differs for each concrete case and depends on the class and function of the vessel. On the American space vessel "Gemini", for example, heat emission of the on-board instruments alone comprised on the order of 500 - 600 kcal/hr. And for manned ships, the designers must also deal with the removal of heat emitted by the cosmonaut's organism. The value of this heat fluctuates within a rather broad range, comprising approximately 230 kcal/hr during the waking period and 70 kcal/hr during the cosmonaut's sleep. With the development of cosmonautics, space apparata have acquired an ever greater number of on-board instruments, and the number of crew members has also increased. All this has led to an increase in the amount of heat emitted in the sealed compartment and this means also to a greater complication of an already complex problem of thermoregulation.

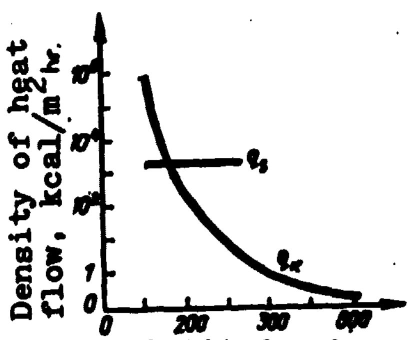

Fig. 1. The dependence of the density of aerodynamic heat flow coming to a plate, depending on the altitude above the Earth's surface.

Let us now see what happens with the internal heat flow coming onto the plate, oriented perpendicular to the direction of the sun's rays. For simplicity of discussion, we will propose that this plate is located very far from the /8 Earth and that all flows except solar radiation are small enough to overlook. The solar flow will be partially absorbed by the plate, and partially reflected from it into space. The value of the flow absorbed by the plate is determined by the average coefficient of absorption As along the entire spectrum.

The plate is in no way a heat accumulator. It does not utilize or use it -- this heat will be the means of radiation being "dumped" into space. The capacity of the plate to radiate heat is determined by the so-called degree of blackness of its surface ε. With one and the same external flow, the plate with the higher value of ε "throws off" heat which has come to it at a lower temperature. Values As and ε depend on the specifics of the material and the condition of its surface and have maximal theoretical values equal to 1.

Thus, the heat which has come to the plate, in connection with the absence of natural air convection in space (or, as specialists say, due to the insignificantly small coefficient of convective heat exchange), is transmitted by it into the surrounding environment by means of irradiation. If one side of the plate is thermoinsulated, then the temperature of this plate will be fully determined by relation As/ε, characteristic for the surface of its other side. With chemical polishing of the surface of the metal plate, coefficients As and ε are equal respectively to 0.2 and 0.1, and in this case the temperature of the plate irradiated by the solar heat flow is equal to approximately 200°C.

Such a temperature is fully realistic for the outside of a space craft turned toward the Sun. This means that

space is at the same time both "cold" (4 K without heating) and "hot" (473 K with illumination by the Sun). Consequently, the designer of a space ship must solve two directly opposite problems: protect the spacecraft both against supercooling and against overheating. The first problem may be solved, generally speaking, quite simply -- insulate the body of the apparatus with a sort of space "jacket" and, moreover, heat the individual parts of its construction (although the latter also leads to excess expenditures of the available energy resources). The second problem is more perfidious -- cooling /9
requires more serious efforts.

The conditions for removing heat into space may be improved by two methods. First, by means of reducing the relation A_s/ε, which in practice is achieved with the aid of proper treatment of the irradiating, or as it is also called, the radiation surface. The application of special paint and varnish surfacings, for example, has become very widespread. These ensure an operating value of $A_s/\varepsilon = 0.5$ (in this case the temperature will be reduced and will comprise approximately 60°C). Secondly, full thermoinsulation of one of the sides of the previously examined plate may be rejected (i.e., reducing the ratio of surfaces withstanding and irradiating the heat flow). Then the heat will come to it from one side, and be irradiated from both. As a result, the temperature of the plate will comprise, for surfaces subjected to chemical polishing -120°C, and for surfaces with paint and varnish coatings having the indicated characteristics -10°C.

In the latter case, a cylinder has a certain advantage as compared with a rectangular plate. The solar flow may fall onto the so-called solar middle of the cylinder, i.e., on the area of its section perpendicular to the sun's rays. At the same time, the irradiation of heat (without consideration of the cylinder's bases) will take place on the lateral surface of the cylinder. As shown by computations, the average temperature of the cylinder along the surface with paint and varnish coatings with $As/\varepsilon = 0.5$ is equal to approximately -20°C.

Thus, with the aid of relatively simple measures, it is possible to achieve the situation where the average temperature of the spacecraft's shell heated by the sun's rays is rather low. However, the spacecraft, as already noted, is heated not only by the Sun, but also by the heat emitted by its on-board apparatus and crew members. This excess heat may be removed by means of increasing the area of the radiation surface. By means of proper selection of the value of this area, it is possible at a given temperature to lead off rather large heat flows from the spacecraft.

In solving the problem of thermoregulation of a spacecraft, the designer seems to be in a vicious circle. In /10 actuality, during the flight time of the vessel, the plane of its orbit constantly changes its position relative to its direction toward the Sun. The flight may take place for a prolonged period only along an orbit illuminated by the Sun or along an orbit which has a portion of shade. On the solar orbit, not only will significant external heat flows fall upon the vessel, but also its instruments, operating intensively, may emit a maximum amount of heat. In the shadow of the Earth, on the other hand, the external flows, as well as the heat emissions from the routine regime of the instruments may be minimal.

Protecting the spacecraft against supercooling on the shadow side, the designer may "bundle it in an overcoat", but then on the sunny side it will be impossible to remove the excess heat and the apparatus will become overheated: the electrolyte in the storage batteries will boil, various elements of the on-board instrumentation will malfunction, etc.

What, then, is the solution to this contradiction? It is simple, although seemingly paradoxical at first glance. We will examine it in the next section.

TO HEAT ...IN ORDER TO COOL

In design bureaus engaged in the design of spacecraft, one may often overhear approximately the following conversation between the specialist responsible for the energetics of the object and the heat specialist developing its thermal protection.

Heat specialist. As indicated by precise computations, one of the instruments operating in open space is overheating. In order to cool it, we are asking for emission of an additional two watts of power for its heating.

Energy specialist. Of course, the power reserves on board the object are insignificant, but we will be able to give you two watts.

To the uninitiated listener, this conversation might seem strange: after all, the instrument under discussion is overheating. Why, then, must it be heated?

Let us attempt to understand this problem. We will assume that some instrument (for example, an optical indicator of the orientation system) is mounted on the outside of the spacecraft body. This instrument is insulated from the body /11 of the apparatus and has its own temperature, which is determined by those internal and external heat flows which we have discussed earlier. To prevent the instrument from "freezing" in the Earth's shadow, it is covered with a "jacket" which passes practically no heat. Obviously, the optical "windows" of the instrument remain open, and consequently on the solar side of the orbit they may be effected by heat flows. Moreover, internal heat emission occurs during the operation of the instrument.

All this heat must be "thrown off" into space, so that the temperature of the instrument does not exceed, let us say, +40°C. For this purpose, on one side of its surface special cutouts are made in the "jacket", i.e., radiation surfaces are created, with the application of the appropriate paint and varnish coatings to them. These surfaces should be located on that side of the instrument which is generally not illuminated by the Sun. However, if this cannot be done, which is most often the case in practice, this is no problem -- the cylinder effect examined above will aid in solving this problem. By selecting the value for area of the radiation surface necessary for casting off the excess heat, it is possible to ensure the maximal temperature of the instrument below its upper tolerance level. However, "throwing off" the heat from the radiation surface will also occur in the Earth's shadow. Here it might happen that the instrument does not operate in the shaded portion of the orbit, i.e., it does not emit heat.

As a result it will be cooled, and there is no other choice other than to heat it with the aid of an automatic heater. If it is necessary to reduce the maximal temperature of the instrument by several degrees, it is necessary to accordingly increase the area of the radiation surface, and this means also to increase the capacity of the heater. This is why in the presented conversation the energy specialist not only calmly accepted a seemingly absurd request by the heat specialist, but also confirmed it.

To "throw off" heat into space from a radiation surface is still but a single problem. The second consists of leading heat to it from a heat emission element with a tolerable temperature overfall between them. In the ideal case, the heat emitting elements should be installed on the radiation surface. However, in practice this cannot always be done. /12
If such an element is located far from the radiation surface,

then the heat flow transmitted by thermoconductivity must overcome a certain heat resistance along the path from the element to the surface. This heat resistance is greater with lesser thermoconductivity of the material, smaller area of the heat conductor cross section, and greater transmission distance of this flow.

Increasing the thermal resistance leads to the situation where the temperature of the heat emitting element will increase at the same temperature of the radiation surface. As a result it may happen that the temperature of the body of the instrument is within a tolerable range, while the heat emitting element nevertheless becomes overheated. For small instruments operating in open space, this problem is not acute, since the distance from their heat emitting elements to the radiation surface, as a rule, is relatively short. The installation of these elements on the body of the instrument, the selection of highly heat conductive material for the body, the creation of heat conductors in a number of cases -- these are the methods which make it possible to avoid undesirable occurrences in this case.

The situation with compartments of space ships and stations is different. The great distance between the individual parts of these compartments creates serious difficulties in attempts to transmit heat from the instruments to the radiation surface with the aid of heat conductivity. Moreover, the large dimensions of the body of such a space apparatus lead to a significant variation of temperatures along the radiation surface. The part which is illuminated by the Sun may be heated by tens of degrees most than the part located in the shade. In this case, the instruments located in the compartment may either become overheated due to the additional heat exchange with the "hot" part of the body surface, or, on the contrary, they may become supercooled due to heat transfer to the "cold" part.

Therefore, for large spacecraft, designers have selected a different means of maintaining the given temperature range of the instruments. A ventilator is installed in the compartment which blows the instruments with gas (the air in the compartment), "removing" the heat from them, and also evening out the temperatures along their surface. Subsequently the "removed" heat may be transmitted to the radiation surface, located directly on the body of the compartment similar to the way in which this is sometimes done for instruments operating in open space. /13

As already noted, this invariably requires expenditures of electrical energy for heating the atmosphere in the pressure chamber. In order to avoid these unsubstantiated losses, a louvre type heat protective screen may be located on the radiation surface. When the temperature of the instruments becomes comparatively high, a special indicator will signal an electric motor, which opens the louvres, and the excess heat is radiated out into space. After the instruments have cooled, this same indicator will give the command and the louvres will close, thereby stopping the removal of heat from the sealed compartment.

This system of thermoregulation is quite simple and at the same time effective. In a number of cases it is used on spacecraft. However, the use of the louvre is not the sole principle of temperature regulation in the sealed compartment. The designers of the Soviet stations "Venus-5 and -6", for example, found a rather unique solution to the problem of leading off excess heat into space. They installed ventilators near the instruments located in the sealed compartment and having the greatest heat emission. These ventilators would switch on when the instruments were turned on. When the temperature increased to a certain upper limit, a valve was opened on command of the thermoindicator, and the air was directed along a special channel to the radiation surface with the aid of yet another ventilator.

The radiation surface used was the central part (reflector) of the station's paraboloid antenna. The air gave off the excess heat to this surface, and this heat was further radiated directly into space. After cooling, the air was again directed to the sealed compartment. After the instrument temperature was reduced, upon command from the same thermoindicator, the valve closed and the heat removal stopped. It is interesting that the radiation surface in these stations was not cylindrical, but flat. The use of such a form became possible because the station had a constant orientation of its solar batteries toward the Sun, while the antenna (the radiation surface) was in the shade for the duration of the entire flight, with the exception of /14 several special regimes.

The conditions of interplanetary flight of a station are quite complicated from the thermal standpoint, since the density of the solar heat flow near Venus is almost twice as high as in an Earth orbit. Therefore, the system of thermoregulation was particularly taxed on the second half of the journey. At this time, the valve was opened constantly and the air constantly transmitted the excess heat to the radiation surface.

The temperature regulation of instruments aboard Soviet "Lunakhods" (Moonwalkers) was also accomplished with the aid of moving air. At the lunar equator, when the Sun is in its zenith, the surface temperature reaches 130°C. If we take into account the internal heat emission of the instruments, then it is easy to understand that the task of protecting the instruments against overheating is a rather complex one. However, on the other hand, during the period of lunar night, which lasts for around 15 earth days, the temperature on the Moon surface drops to minus 150-170°C. This means that it is necessary to ensure heating of the "Lunokhod" instruments in order that they do not "freeze".

But where to find the heat source? Even if such a source were found, how to keep it from additionally heating the instruments during the time of lunar day?

To solve this problem, Soviet specialists have used a double circuit system of thermoregulation. When it was "hot" on the "Lunokhod", the cooling circuit was in operation, when it was "cold" -- the heating circuit.

The excess heat was removed from the instruments with gas which moved under the pressure of a three-stage ventilator. For blowing certain particularly "hot" instruments, gas was directed to them through special air ducts. Depending on the temperature inside the "Lunokhod", the gas, with the aid of a regulatable baffle located at the entrance to the air conducting tract, was directed further either into the "hot" or into the "cold" circuit. The baffle, aside from the two extreme positions in which either the "hot" or the "cold" circuits were closed off, had four intermediate settings, which controlled the expenditure of gas through each of the circuits. The baffle was controlled by a complex automatic system which reacted upon the signals of a sensor element -- the temperature indicator.

In the "cold" circuit the gas gave off the excess head /15 to the radiation surface, for which the upper wall of the instrument compartment found in the shade of the "Lunokhod's" body was used, and which therefore had a relatively low temperature.

In the heating circuit the gas was heated from the isotopic heat source located outside the sealed compartment. This source was connected with the compartment through two pipe stems. Through one of them the gas was directed to the isotopic source, and through the other it went again to the instruments. In order to prevent the heater's heat from warming the sealed compartment during the period of lunar day, a heat protective shield was installed between it and the body of the moon walker.

Additional baffles were provided inside the pipe stems which were tied to the main baffle by a system of pull rods and which turned simultaneously with the main baffle. These baffles reduced the radial and convective heat flow from the isotopic source at those moments when the instruments did not need to be heated. Moreover, in order to reduce the heat losses from the compartment during the period of lunar night, the radiation surface was covered with the "Lunokhod's" heat insulating cover. This at the same time made it possible to protect against supercooling also the light transformers located on its internal surface, which turned the solar energy into electrical. Obviously, the body of the "Lunokhod", as usual, was thermoinsulated with a special "jacket".

Such a system of thermoregulation made it possible to maintain the instrument temperature within a range of 0-40°C with changes in the Moon surface temperature within an interval on the order of 300°C.

Widely used in space technology are systems whose operation is based on heat removal with the aid of a liquid heat vehicle. A scheme of one such system is presented in fig. 2. The entire compartment is covered with a "jacket" (6). The heat from the instruments (7) is removed by a stream of gas created by an ordinary ventilator (1), and is directed into a special heat exchanger (2), which is a set of pipes along which the heat carrier moves with the aid of a hydraulic pump (3). The gas, blowing on the pipes, transmits its heat to this heat carrier, which is directed further along the hydraulic main to the radiation surface (4) located on some part of the spacecraft, but isolated by /16 one means or another from the body of the sealed compartment.

Passing over the system of pipes onto the radiation surface, the heat carrier gives off its heat to it, and this heat is them radiated into space. If, as a result of the cooling the temperature in the compartment becomes low, the special indicator (8) gives a signal to the by-pass valve (5), which begins to direct the heatcarrier to go around the heat exchanger. When the increase in instrument temperature reaches a certain level, upon signal from the same indicator, the by-pass valve directs the heat vehicle into the heat exchanger -- and the removal of heat from the compartment begins.

Presented in fig. 3 is a scheme of the system of thermoregulation of Soviet space ships "Vostok" and "Voskhod". The elements of this system were unified into a block, which included the main and reserve ventilators (1), a heat exchanger (2) and an automatic temperature regulator system (3, 4, 5, 6). The air set into movement by the ventilator removed the heat from the instruments in the compartment and from the cosmonaut's organism, and directed it further to the heat exchanger. The cosmonaut would determine (order) the required temperature on an indicator (3), and this temperature was then maintained automatically. The sensing element (4) worked out the directing signal which "told" of the degree /17 to which the actual temperature in the sealed compartment corresponds to that ordered by the cosmonaut. This signal was transmitted to the final control element (6) which with the aid of a blind (5) changed the expenditure of air directed into the heat exchanger.

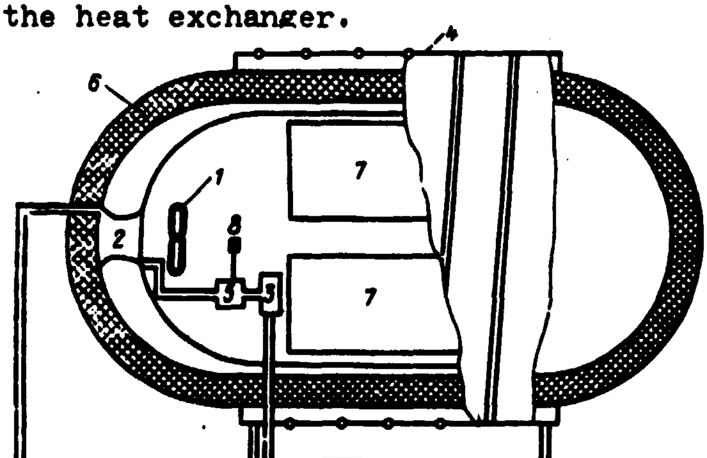

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Fig. 2. Scheme of heat removal with the aid of a heat vehicle (the by-pass contour is not shown).

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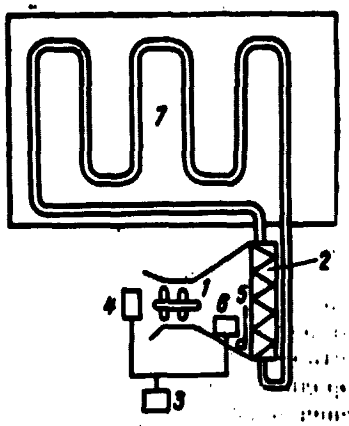

Fig. 3. Scheme of the thermoregulation system on the space ships "Vostok" and "Voskhod".

If the cabin temperature was higher than that set by the cosmonaut, more air was fed into the heat exchanger. If it was lower, less air was directed to the heat exchanger. With the aid of a spur-gear hydraulic pump, a cooling agent was pumped through the pipes of the heat exchanger, which removed the heat from the air and transferred it (i.e., the heat) to the radiation surface (7). Circulating along the pipelines located on this surface, the cooling agent gave off its heat, which was subsequently radiated into space and, having cooled off, again went to the heat exchanger. This system proved to be rather effective. It maintained the given temperature with a precision of $\pm 1.5^{\circ}C$.

Presented in fig. 4 is the principle scheme of the system of thermoregulation on the Soviet orbital station "Salyut-6". The rather large dimensions of its compartments, the considerable heat capacities produced by its apparata and crew cause significant difficulties in solving the problems of ensuring the thermal regime. For ventilation of the station, the designers had to provide several tens of ventilators on it. The heat removed by the air moved under the pressure of the ventilators is transmitted in the heat exchanger to a heat carrier circulating along the heating circuit tract (1), whose basic task consists of heating the individual elements of the station's constructions.

Thus, for example, the head from this circuit is transmitted in the heat exchangers (4) to intermediate circuits (10), which serve to heat the transport space /18 vehicles "Soyuz", "Progress" which are docked with the station. The necessity of such heating is associated with the fact that the instrumentation in these ships in the course of their joint flight with the station operates in a non-loaded, routine regime and emits little heat. The station's thermoregulation system joins with the thermoregulation system of the transport vehicle with the aid of special hydraulic locks of the docking aggregate (7) joining the tracts of the hydraulic mains of both vehicles.

The heat carrier of the heating circuit circulates also along the walls of the station (9), heating the cooled and cooling the heated sections or, in other words, equalizing their temperatures. If too much heat is emitted in the cabin of the station and its temperature increases, the heat exchanger is set into operation (11), in which the excess heat is transmitted from the heating circuit to the cooling circuit (2). Circulating along the tracts of the latter, the heat carrier transfers the heat obtained in the heat exchanger (11) to the radial surface (8), which radiates it out into space. The expenditure of the heat carrier through the heat exchanger (11) may be regulated with the aid of a special regulator valve, thereby changing the degree of cooling of the liquid in the heating circuit.

When there is no crew aboard the station and its instrumentation emits little heat, the air temperature in the compartment drops. To prevent it from dropping below a tolerable limit, an electric heater (13) is provided in the thermoregulation system.

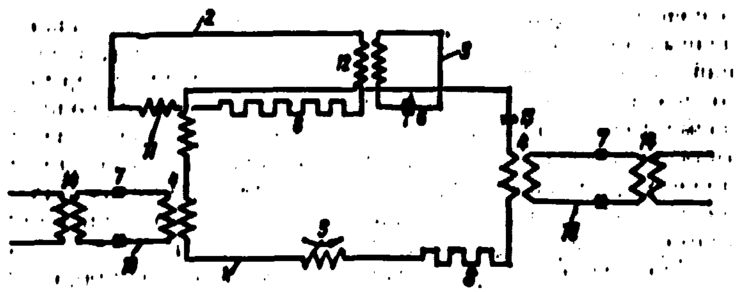

Fig. 4, Scheme of the system of thermo-regulation on the orbital station "Salyut-6".

Moisture emitted, say, during the respiration of the cosmonauts, should be removed from the station's atmosphere. Special cooling-drying instruments (6) serve to solve this problem. Moisture settles on the surfaces of these apparata cooled to a temperature on the order of 5°C, collects in a container, and is then fed to a system which regenerates water from the condensate. The cooling of these surfaces is done with the aid of a circuit (3) whose heat carrier gdves off its excess heat in the heat exchanger (12) to the circuit (2). /19

Of course, the heat carrier is pumped by hydraulic pumps along the various circuits. Since with a change in the temperature of the liquid, the volume which it occupies also changes, i.e., the pressure changes in the cooling tracts within the thermoregulating system, a volume compensator is provided.

The complex of cooling-drying aggregates (6) and heat exchanger (5) includes ventilators which direct the air between the heat exchanger tubes, and an air expenditure regulator which is a similar blind with drive as used on the space ships "Vostok" and "Voskhod". Thus, automatic regulation of the liquid temperature within the internal cooling circuit is also performed at the station. This temperature is maintained with an accuracy of $\pm 2^{\circ}C$ respectively to one of its nominal values: 5, 7 and $9^{\circ}C$, and an air temperature comprising 18-$25^{\circ}C$ within the habitable volumes of the station.

The thermoregulation system of the transport space ship "Soyuz" consists of approximately the same blocks as in the station "Salyut". Its composition includes two main liquid circuits: an internal one, intended for thermoregulation of the habitable compartments, and an external, which serves to lead off excess heat from the sealed compartment into space. The heat is removed from the heat emitting elements with the aid of air moving under pressure from ventilators and is transmitted in a gas-liquid heat exchanger to a liquid "passed" with the aid of hydraulic pumps through the hydraulic main. With the aid of the liquid, the walls of the aggregate compartment are thermostated.

The excess heat is transmitted in a liquid-liquid heat exchanger to the external circuit and is "thrown off" into space from the radiation surface. The temperature of the liquid in the internal circuit, as on the "Salyut" station, is regulated with the aid of automatic controls and regula- /20
tors. This makes it possible to maintain the walls of the cooling-drying aggregate at the necessary temperature level, and this means also the level of air humidity in the cabin. The air temperature in the ship's cabin is also regulated automatically, similar to the manner in which this is done on the "Salyut" station.

For heating the ship "Soyuz" during its flight in conjunction with the station "Salyut", it is also provided with an auxiliary circuit in the thermoregulating system.

The cabin temperature regulation system of the modern American space vessel has much in common with the thermoregulating system aboard Soviet ships. This cabin has three compartments. Located in the upper one are the main crew members: the commander of the ship, the pilot, the on-board engineer and the weight specialist. The central compartment is intended for cosmonaut-experimenters: scientific workers, engineers, etc. The life support system is located in the lower compartment. Presented in fig. 5 is a scheme of the system of thermoregulation for one of these compartments.

The ventilator (1), blowing the cabin with air, directs it through a cartridge (2) with lithium hydroxide, activated charcoal (for control of carbon dioxide concentration and deodorization) into the heat exchanger (3), where the air gives off its excess heat to water pumped with the aid of a hydraulic pump (4) along the heat exchanger tubes. Then /21 the water goes along channels (5) located on the walls of the cabin, and equalizes their temperature, as well as removing excess heat from them during the ship's flight in the earth's atmosphere.

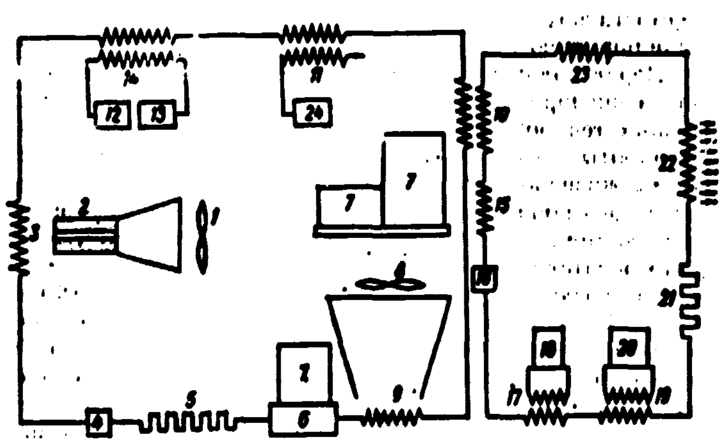

Fig. 5. Scheme of the thermoregulation system on an American space vessel.

The cabin instruments (7) are cooled in two ways: they are mounted on hollow plates cooled with water (6) and they are blown by still another ventilator (8), giving off their excess heat to the air, which transmits it finally to the water in the heat exchanger 9. In the heat exchanger 10, the water transmits the removed heat to a second heat carrier -- freon, which gives it off to the radiation surface 22. The cooled water is directed into the heat exchanger 14, where it cools the water coming from the tank 13 and directed onto the galley (12) for use by the cosmonauts. Further the water with temperature of 7°C again goes to the heat exchanger 3, "removing" here the heat from the air, and the cycle is repeated.

The freon, having obtained the heat from the water in the heat exchanger 10, is directed with the aid of a hydraulic pump 16 to the heat exchanger 15, where it takes on the excess heat "removed" in the useful load compartment of the space ship. Further in the heat exchanger (17) the freon takes on the heat from the fuel elements (18), in heat exchanger (19) is heats the heat carrier in the hydraulic system (20) intended for solving a number of problems (for example, for controlling the thrust vector of the main engine installation), thermostats the ammonia tank (21) and is directed to the radiation surface (22).

Before being fed into the fuel element chambers, hydrogen and oxygen are first heated in special heat exchangers. The water obtained as a result of the operation of these elements is directed into the life support system containers or is dumped overboard from the space ship. Special electric heaters are installed in places where water having a low temperature accumulates in order to prevent its freezing. The air temperature in the cabin is regulated within limits of 18.4 - 26.7 ± 1°C. On a particularly heat stressed section of the flight (entry into dense layers of the atmosphere), all the surfaces of the apparatus with which the crew members may come into contact do not have a temperature exceeding 45°C.

During the pre-start preparation, as well as during entry of the space ship into orbit, the heat removal from the radiation surface turns out to be insignificant. This /22 is explained by the fact that it is closed off from the surrounding space by the casing of the carrier rocket, and as a result is not cooled either by air flowing around it or by irradiation. Therefore, on all the described space ships, special elements of the thermoregulating system are provided for solving this problem. Thus, for example, on the examined American ship, a heat exchanger (23) was provided for cooling the cabin before launch, in which the heat from the ship's heat carrier is removed with the aid of a cooling agent fed into this heat exchanger by a ground installation.

During the take-off of the carrier rocket, the evaporative heat exchanger is set into operation (see position 11 on fig. 5), whose principle of operation is identical on all known space ships. As the carrier rocket attains altitude, the pressure of the surrounding air drops and a valve opens in the heat exchanger under the influence of the excess pressure, letting off air. The water found in the heat exchanger poils under the reduced pressure, absorbing the head of the phase transition. Effective vaporization begins at an altitude of approximately 30 km, with the valve in open position due to the pressure of the water vapors.

Evaporative cooling may be used also during peak heat loads during the flight of the space apparatus along an orbit and also in emergency situations during breakdown of the radiation cooler. In these cases, the valve opens upon a signal from the temperature indicator or by special command.

All the above-described systems are related to active, i.e., to those which themselves alter their operation depending on the change in the temperature regime in the sealed compartment of the space apparatus. In practice, systems are often encountered which have a pre-determined logic

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of operation which does not change during the flight process. These are called passive. With the aid of such systems, it is possible to ensure a given temperature regime, for example, for instrumentation for space probe intended for descent from an interplanetary station onto the surface of Jupiter.

From the outside, a hive construction made of fiberglass /23 may be attached to the probe's metallic body (fig. 6), and to it, in turn -- a heat protective screen: for the forward (nose cone) part from phenolic resin reinforced with coal fiber, and for the rear -- of phenolic resin reinforced with fiberglass. The entire probe, moreover, should be enclosed in a thermoinsulating sheath ("jacket"), which insulates the apparatus in the vacuum of space. The jacket consists of a large number of layers of metallicized (for example, gold plated) film, alternating with layers of low heat conductive fabric (such as glass fabric). In cosmic space, due to the fact that the individual layers of the jacket are weakly contacting with one another, heat is practically not transmitted through such insulation by means of thermal conductivity (with the exception of places where it is compressed.

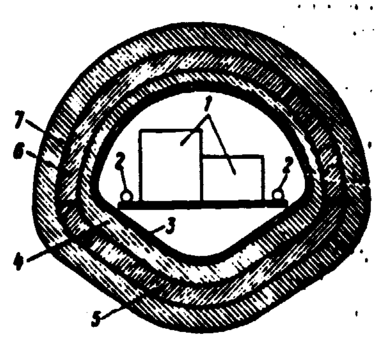


Fig. 6. Scheme of the thermal protection of a space probe to Jupiter:

1 - instruments; 2 - radiation heat sources; 3 metallic body; 4 - honeycomb construction made of fiberglass; 5 - thermoinsulating jacket of phenolic resin reinforced with coal fiber; 6 - thermal protective jacket of phenolic resin reinforced with fiberglass; 7 - thermoinsulating jacket of alternating layers of gold plated mylar film and dacron mesh.

Due to the many layers of such a "jacket", the transfer of heat by radiation is also insignificant through it, i.e., it has great heat resistance. However, no matter how ideal the probe's thermoinsulating, no matter how small the heat losses from its sealed compartment, nevertheless they may lead to supercooling of the non-operating apparatus during prolonged interplanetary flight far from the Sun. In order to prevent, radio isotope heaters may be installed inside the probe which compensate for the heat loss through thermoinsulation and which thereby maintain the necessary temperature regime of the instrumentation.

With the appearance of spacecraft, a widespread search has been conducted for new possibilities of solving the problem /24 of thermoregulation. In a number of cases the efforts of scientists have been crowned by success. New, highly effective methods of heating and cooling bodies have begun to find their application. Some of these methods will be examined in detail in the next section.

THE NEW -- THE WELL FORGOTTEN OLD

In 1834 the French scientist Jean Charles Peltier decided to determine how the temperature changes along a uniform and along a non-uniform conductors under the influence of electric current. The experiments yielded unexpected results: when current flows along a non-uniform conductor in one junction the heat is absorbed, while in another it is emitted. In other words, if the temperature of a hot junction is maintained by one means or another at a certain level, then with the aid of the cold junction it is possible to cool the body to lower temperatures. In 1838 the Russian scientist E. Kh. Lents was thus able to freeze a drop of water.

The extremely low effectiveness of the first thermoelectrical coolers based on Peltier's discoveries became the main reason why they did not find practical application for a long time. At the end of the 40's of the 20th century, Academician A. F. Ioffye became interested in this phenomenon. The new semiconductive materials which had emerged made it possible to hope for improvement in the characteristics of these coolers. Studies soon confirmed this, and widespread designing of various instruments based on this effect began in this country. Such instruments began to be used, for example, in medicine. Air conditioners and even (though still experimental) coolers are created on the basis of this effect.

In the 60-70's the Peltier effect began to be used also in space technology for solving various thermal problems. It is known that a number of elements of radio electronic instrumentation operate better at low temperatures. At the same time, installations may be present aboard the space apparatus which do not allow relatively great cooling. As a result, there is a contradiction between the required temperature regime of some elements, installations, etc., /25
and others. The Peltier effect is utilized in a number of cases to solve this contradiction. In this case, a relatively high temperature value is maintained within the entire sealed compartment, while elements requiring low temperatures are additionally cooled with the aid of this effect.

An example of cooling with the aid of thermoelectricity may be cooling of infrared receivers of the orientation system's optic indicators. In practice, it is possible to reduce the temperature of the infrared receiver by 60-65°C, but this is not the limit. By using several thermocouples which are joined together in a special manner (cascade connection), it is possible to obtain a temperature reduction of 100°C or more. A scheme of one such connection is shown in fig. 7.

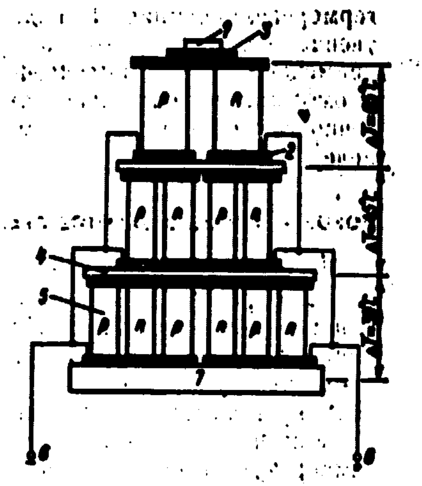


Fig. 7. Scheme of a thermoelectric cooler:

1 - receiver for radiant energy; 2 - copper;
3 - cold junction; 4 - anodized aluminum;
5 - bismuth teluride; 6 - electrical power source;
7 - hot junction.

Each thermocouple represents a small p- and n-type semiconductor plate, made for example, from bismuth telluride, joined with copper plates. The hot junction of each thermocouple is is connected with the cold junction in sequence so that there is good heat contact at the point of their juncture. During passage of electrical current along the entire system, the cold junction will have a temperature which is 140°C lower than the hot junction. If the optical receiver is properly located on the cold junction, ensuring whenever possible minimal thermal resistance between the junction and the receiver, then evidently the temperature of the latter will be significantly lower than the temperature of the hot junction. In this case, obviously, the heat must /26 be removed by one means or another from the hot junction so that its temperature (and consequently also the temperature of the cold junction) would be at the necessary level.

Unfortunately, a large increase in the number of steps in the cascade is impossible due to the sharp drop in the effectiveness of their operation. Therefore, in practice there is usually a limit of three or four cascades.

The described thermoelectrical device is sufficiently compact: it has an area on the order of 6 cm^2, consumes a current of 4A with feed voltage of 6 V. It is true, however, that due to technological difficulties in manufacturing bismuth telluride, for operation at low current loads, thermoelectrical devices produced by industry utilize a greater current (on the order of 100 A) and lower voltage (0.5 V).

In 1944 the American R. S. Gaugler was given a patent on a device for removing heat, which subsequently came to be called a heat pipe. The basic idea for it is as simple as it is original. The pipe consists of a body, on the inside of which are located longitudinal (or other) microchannels (capillary structure), with the operating body inside the pipe. If one end of this device is heated, then liquid may, obviously, turn to vapor, which fills the pipe. If at the same time the other end is cooled, then the vapor will condense on it and the drops of liquid will move along the microchannels to the heated end under the action of capillary pressure, where heating, vaporization, etc. again take place. This process is continuous and is accompanied by heat transfer from the hot to the cold end of the pipe.

The idea of the heat pipe was forgotten, and was remembered only in the mid-60's, when it was again patented in the USA by T. L. White, but this time as a method of ensuring the heat regime for elements of space vehicles. It should be noted that the application of this method in space technology is rather lucrative, since it, first of all, does not require any energy expenditures for the transfer of heat by the operating medium, secondly, is simple enough, and finally, makes it possible to ensure the heat regime of elements found in places which are inconvenient to cool with the aid of other methods. Moreover, the wide assortment of operating bodies with varying boiling points /27 makes it possible to ensure in principle any temperature of the cooled elements.

By the present time, thanks to the efforts of scientists of various countries, heat pipes have appeared for solving a wide circle of functional problems, quite varied in their construction and in their characteristics. Thus, for example, there are pipe diodes, which transmit heat only in one direction. They are particularly expedient for use in those cases when the heat source is periodically switched off, while the external background heat flows are comparable with the flow from the basic source.

If an additional gas which does not condense under operating conditions is introduced into the vapor channel of the heat pipe, this will give a pipe with variable conductivity. This non-condensing gas is displaced by the flow of vapor into the zone of condensation, where it collects without participating in the circulation. Under certain conditions, a relatively sharp division boundary is established between the vapor and the gas. In the part of the cooled pipe surface which is taken up with non-condensing gas, the heat removal will be practically absent. The length of the section blocked by the gas will be determined primarily by the temperature and pressure of the vapor.

With increased value of heat flow fed to the heated part of the pipe, the temperature, and consequently the vapor pressure increase, and this means that the length of the blocked zone becomes shorter or, in other words, the area of heat removal increases. A reduction in the heat flow, on the other hand, reduces the area of heat removal. Such a pipe may ensure a change in the heat transfer capacity by more than 15 times with insignificant changes in the temperature of the element which it cools. Moreover, its utilization also makes it possible to minimize temperature fluctuations of this element with change in the coolant temperature.

Since the late 60's, heat pipes on spacecraft have been used quite frequently. Thus, for example, on the space apparatus "GEOS" (1968), heat from the transmitter was removed by two aluminum pipes using ammonia as the working medium, and on the "OAO-1" (1972) the on-board computers were cooled with the aid of a steel pipe with variable conductivity, operating on methyl alcohol. The condensation zone in the latter case was a cooler-irradiator. The pipe maintained a device temperature at the level of $17 \pm 3^{\circ}C$. On some spacecraft the number of heat pipes is counted in tens (thus on the American orbital station "Skylab" there were 40 of them, and on the sputnik "ATS-6" -- 55). /28

In the late 60's - early 70's, studies were conducted in various countries directed toward the application of heat pipes even in such heavily loaded elements of spacecraft as liquid fuel rocket engines (LRE). The rather high temperatures characteristic for these engines, naturally, led to efforts to utilize the most heat resistant materials for the body of the pipes and high temperature working mediums. The bodies were made, as a rule, of pyrolytic graphite, and studies were conducted on the application of tungsten for this purpose. Lithium and sodium, as well as silver were used as the operating mediums.

As a result of these studies, high capacity pipes were created. They removed heat flows equal to $7 \cdot 10^6$ kcal/m^{2}hr, which generally speaking made it possible to utilize them for cooling certain space LRE. In the early 70's, the "Tiokol" company developed a system for cooling an experimental liquid fuel rocket engine with the use of heat pipes. This engine operated on flourine oxide and diborane and developed a thrust on the order of 60 kG at a combustion chamber pressure of around 7 kG/cm^2. From the opening of the supersonic nozzle, where the heat flow density is maximal, the heat was removed with the aid of 8 radially placed pipes made of pyrolytic graphite. The external (in relation to the engine) ends of the pipes were joined with a circular heat exchanger, whose internal wall served as the surface for condensation of operating medium vapors. The ends of the pipes adjoining the nozzle opening formed the evaporative surface.

Despite the applicability of utilizing heat pipes in small liquid fuel rocket engines in principle, there has not been a single known practical application of an engine with such a cooling system to the present day. No matter how effective the methods of removing heat from heated bodies examined in this section may seem, nevertheless we must acknowledge that both heat pipes and devices based on the phenomenon discovered by Peltier bear an auxiliary character in solving the problems of heat exchange in space technology and make it possible to solve a relatively small circle of problems currently facing designers. /29

HOME THROUGH PLASMA

The relatively narrow aerial strip above the earth's surface which gives life to all which lives on Earth becomes a serious barrier along the path from space, mercilessly burning up everything which comes into it. The high speeds of spacecraft (or other bodies) entering the atmosphere lead to the development of temperatures reaching 7000-8000°C in the oncoming air flow at their frontal border. There is material in nature capable of withstanding such heat loads. However, this does not mean that it is impossible to preserve the material part of a returning spacecraft. Despite such a high temperature, certain meteorites "break through" the atmosphere without fully burning up, and individual parts of the American orbital station "Skylab" which went off its orbit have reached the Earth.

Why does this occur? First of all, because the descent (or drop) in the atmosphere takes a rather short time and the heat flows coming to one body or another, while disintegrating it, nevertheless do not have enough time to finish this "job" before the descent is completed. It is specifically this effect which is utilized in the thermal protection of descending apparata. For this purpose, a special covering is applied to their body from the outside, which disintegrates during aerodynamic heating, absorbing a certain amount of heat in the process. Since the value of the heat flow affecting a unit of its area during descent of the apparatus is fully determined, it is possible to select such a thickness of the thermoprotective covering at which this flow will be absorbed by it, while the main body of the apparatus remains undamaged.

The method of heat protection based on the pre-determined process of material disintegration, absorbing heat flow in the process, is called ablation cooling. The possibility of /30
of its application is generally determined by the existence of materials capable of absorbing during their disintegration a significant amount of heat and at the same time of having a comparatively low specific density and satisfactory strength.

The history of such materials is relatively short. Already in the 20's of the XX century, plastics were in existence which represented various resins (polyester, malamine, phenolic, etc.) reinforced with cotton and cord fabrics. In the early 40's, fiberglass began to be used instead of these fabrics, as a result of which a new material emerged which possessed high strength and low specific density.

Thus, for example, if the specific density of mild steel comprises approximately 7.8 g/cm^3, while the ultimate tensile strength is 4200 kG/cm^2, then for plastic based on polyester resin these parameters turned out to be equal to 1.85 g/cm^3 and 6265 kG/cm^2, respectively.

In the 40's, piston aviation was at the limits of its development, when the gain in every kilogram of airplane weight, every kilometer per hour of its speed was obtained at the price of great difficulties, and therefore it was not by accident that the new material attracted the attention of aviation specialists, who began using it to make various elements. Beginning with the mid-50's, when rocket technology specialists became faced with the question of heat protection of returning heat parts of rockets, special plastics based on phenol-phormaldehyde resins were developed which had good heat absorption properties. In the early 60's, new materials based on epoxy resins were also developed which, although they did not exhibit good ablation properties, nevertheless possessed good mechanical and technological characteristics. Aside from fiberglass, asbestos, coal, quartz, graphite, and certain other types of fibers are currently finding application.

Technological methods of making parts from plastics have also been developed. In the 50's, parts were made by means of direct pressing in molds. However, with the beginning of the application of reinforced plastics in rocket-space technology, this method no longer satisfied the increased requirements, since the manufacture of large scale /31 parts with the aid of this method required equipment (for example, presses) of large dimensions. The search for new methods led to the development of the so-called winding method in the early 60's: a fiber held together with the aid of a resin is wound onto a frame having one form or another.

The next step on the road to improving the technology of making parts from reinforced plastics became the development of the lamination method, in which the fibers are not wound, but laminated onto the frame. The indicated three methods of making parts (direct pressing, winding and lamination) are applied equally at the present time in rocket and space technology.

Reinforced plastics are widely used for making heat protective screens for returnable spacecraft. Despite the low specific density of plastics, the weight of these screens turns out to be significant. Therefore, to reduce it, it is preferable to select a form for the descent segment having the least area subject to great thermal loads. A semisphere is sufficiently suitable for this purpose, and is often used in practice.

Here, for example, is how it is possible to provide heat protection of the previously described probe descending from a spacecraft to the surface of Jupiter. Upon entry into the planet's atmosphere, the "space jacket" insulating the apparatus in open space burns off, and the thermal loads from friction with the atmospheric gases will be borne by the heat protective screens. Disintegrating, they will absorb heat in such a way that the basic construction remains undamaged. After the speed becomes subsonic, these screens will not only be useless, but also harmful. Heated to high temperatures, they will themselves become heat sources. A protection against the latter may be honeycomb thermoinsulation made of fiberglass.

As the probe descents through the dense layers of the atmosphere, the surface of the heat protective screens will be rapidly cooled, with the intensity of this cooling, with all other conditions being equal, 20-40 times greater than during descent into the Earth's atmosphere. After landing of the probe, its instrumentation will begin to operate, emitting around 150 kcal/hr of thermal energy. Since the probe is thermoinsulated, under the effect of this energy the instruments will begin to heat up, but because of the short computed time of the probe's operation on the surface of Jupiter, which comprises only 15-30 min., the temperature will not have sufficient time to increase above the allowable upper limit. /32

It is approximately according to this principle that the thermoprotection of a descending apparatus (probe) from Soviet stations of the "Venera" type is provided. This apparatus has a spherical form and is equipped with several layers of heat protective covering, part of which disintegrates during aerodynamic deceleration, while the remaining part protects the probe instrumentation against the action of the high temperatures on Venus, which reach 280° C on its surface. From the thermal standpoint, it is significantly more complex to ensure the safety of the material part of apparata descending to the surface of other planets as compared with descent from an Earth orbit. This is explained by the fact that the "other planetary" apparata enter the planet's atmosphere from a higher, second cosmic speed. The pioneers in the solution of this problem were the designers of the Soviet station "Venera-4", who ensured the safety of its material part during its landing on the surface of Venus.

To solve the problem of heat protection of spacecraft during their descent into the atmosphere of a planet, it is necessary also to consider certain ballistic specifics of flight. For example, the examined probe should be directed along a gently sloping trajectory for the descent into Jupiter's atmosphere, so that the point of entry is near the planet's equator, and the probe moves along the direction of its rotation. This makes it possible to reduce the speed of motion of the craft relative to the planet's atmosphere, and therefore also to reduce the heating of its construction. The configuration of the probe is selected in such a way that it begins to decelerate at the greatest possible altitudes, where the atmosphere still has significant rarification.

There are numerous ballistic specifics associated with the heating of spacecraft during their descent, and the selection of the optimal flight trajectory may rightly be considered as one of the methods of heat protection.

The problem of heat protection is particularly complex for spacecraft with repeat flight capabilities. Their developed surfaces lead to a rather large weight of ablation /33 heat protective covering. Moreover, the requirement of repeated use presents, generally speaking, the problem of developing materials which are capable of withstanding the occurring heat loads without disintegration. The maximal temperatures on the surface of the body of the American space ship comprise 1260 - 1454°C. The operating temperature of the aluminum alloy from which this body is made must be maintained at no higher than 180°C. But even this value is unsatisfactory for the crew and the instruments aboard the craft. Its further reduction requires the application of additional measures: internal cabin thermoinsulation, heat removal with the aid of the thermoregulation system.

The entire surface of the examined craft is divided by temperature level into four zones, each of which utilizes its own covering. In places where the temperature does not exceed 371°C, a repeated use flexible heat protective surfacing (a) is used, which is a felt made of special fibers. This covering in the form of 0.9 x 1.2 m sheets is applied to the body with an adhesive-hermetic sealant. To give the surfacing moisture resistance and the necessary optical properties, a silicon organic elastomer film is applied to its surface before installation. To protect the flexible joints, the external layer of the surfacing is made of a special ceramic fiber. The protected units may simply be wrapped with this covering.

In areas where the surface temperature comprises 371-649°C, a repeated use covering (b) is also used, consisting of amorphous quartz fiber of 99.7% purity, to which a binding agent is added -- colloidal silicon dioxide. The covering is made in the form of plates 203 x 203 mm in size with thickness of from 5 to 25.4 mm. Borosilicate glass containing a special pigment (a white surfacing based on silicon oxide and shiny aluminum oxide) is applied to the outside surface of the plates to obtain a low absorption coefficient of solar radiation (As) and high radiation coefficient (ε).

The thermal protection of the part of the body with temperature of 649 - 1260°C is provided with the aid of repeated use insulation (c) similar to that which we have just described. The difference is in the size of the plate /34 (152 x 152 mm with a thickness within the range of 19-64 mm), as well as in the greater value of As and ε. Used on the nose cone and wing tips of the craft, where the temperature exceeds 1260°C is a material made of carbon, reinforced with carbon fiber (d). In the process of returning the craft to Earth, this material is disintegrated, and must be replaced before each subsequent flight.

Fig. 8 shows the placement of various types of surfacings on a repeated use ship. Surfacing a takes up 20% of its surface, with the remaining part covered with other surfacings. The number of plates of surfacing c comprises 24,100, and surfacing b - 6800. The overall weight of the thermo-insulation is equal to 7.2 tons.

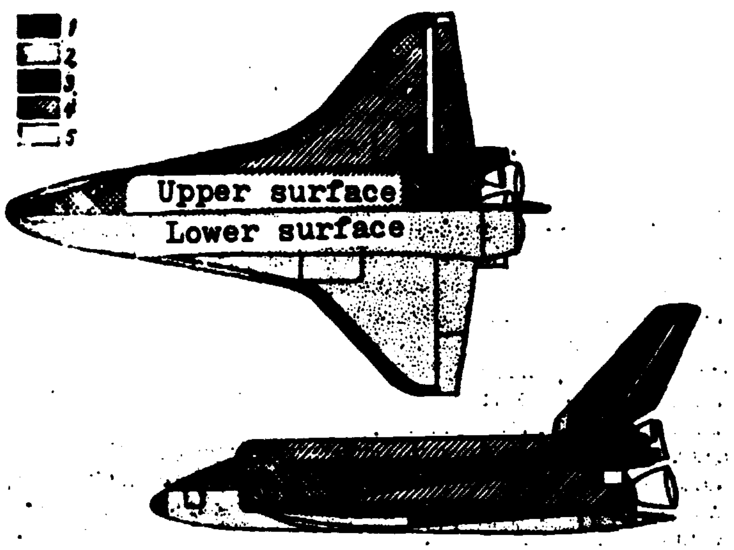


Fig. 8. The location of various types of surfacings on the body of a repeated use spacecraft:

1 - carbon, reinforced with carbon fiber (d); 2 - high temperature heat protective surfacing of multi-use application (c); 3 - low temperature heat protective surfacing of multi-use application (b); 4 - flexible heat protective surfacing of multi-use application (a); 5 - metal or glass.

It should be noted that the requirements for heat protective surfacings of multi-use craft are quite varied and /35
very complex. Thus, for example, these surfacings must have fully determined optical properties, which is necessary for maintaining their temperature regime during orbital flight and in the segment of descent. They must withstand great dynamic loads during the craft's entry into the dense layers of the atmosphere. To solve this problem in the examined case, the material is made porous -- the hollows occupy 90% of the plate volume. As a result, the pressure in the plates is always equal to the pressure of the surrounding environment. Therefore, all the aerodynamic loads are transmitted to the facing of the ship's basic construction.

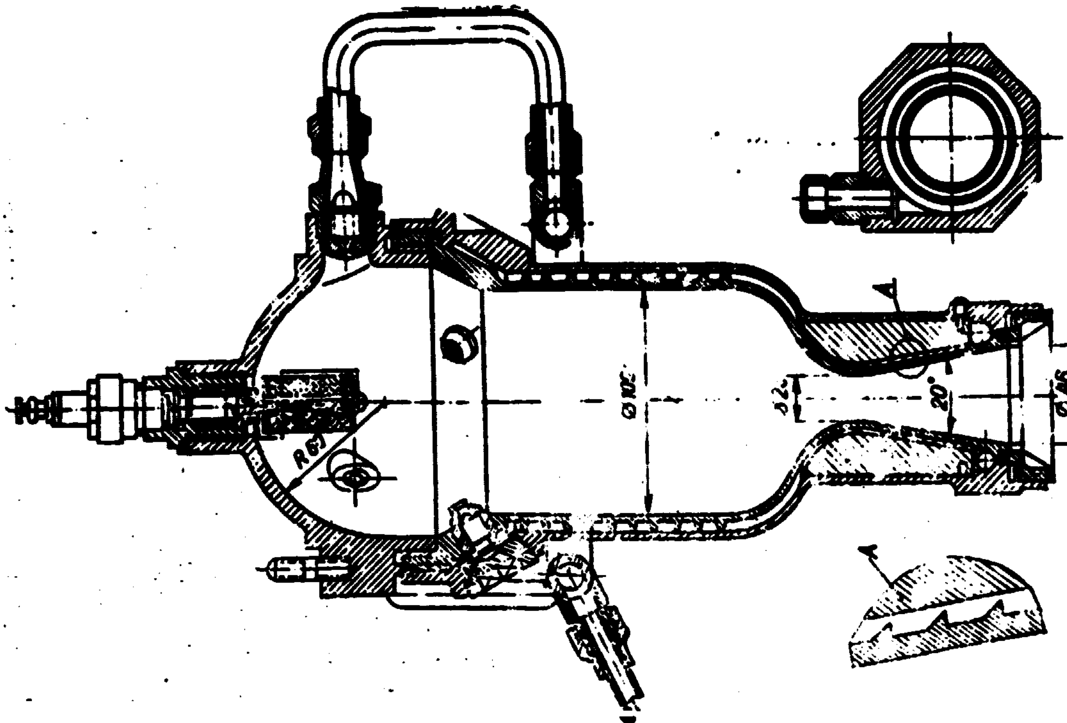
The described multi-use spacecraft is equipped with liquid fuel rocket engines, the chamber temperature exceeds 4000°C, i.e., it is only a third less than the temperature on the surface of the Sun, the pressure of the combustion products exceeds 200 kG/cm^2, while the rate of movement of the gas flow reaches 4500 m/sec. In 1 hr, an astronomical amount of heat is transmitted to 1 m^2 of the wall of such an engine, which is equal, for example, in the critical section of the nozzle to $150 \cdot 10^6$ kcal. The presented figures clearly characterize the complexities which designers must overcome in solving the problems of heat protecting which stand as a serious barrier in the path of development of space liquid fuel rocket engines.

The first to ponder the problem of cooling the liquid fuel rocket engine was K. E. Tsiolkovskiy. In 1903 in his famous article "A Study of Outer Space using Rocket Devices" he wrote: "The hydrogen and oxygen in liquid form, before coming to the gun (the combustion chamber -- G. S.), pass along a special jacket along its surface, cooling it, becoming heated themselves and only then coming into the chamber and exploding". An example of a rocket engine with such cooling, which received the name of external regenerative, may be the liquid fuel rocket engine ORM-65, created in the second half of the 30's by V. P. Glushko. The chamber of this engine (fig. 9) was surrounded by a jacket having a screw ribbed cooling tract. One of the fuel components (nitric acid) was introduced at the nozzle edge, passed along this tract, cooling the chamber walls, and was then fed into the chamber, where it burned together with the kerosene. The head of the engine was cooled from the inside by the fuel components.

In developing an engine for the A-4 rocket, German specialists using flow-through cooling were not able to protect against the burn through of the narrowest part of the nozzle, its neck, or in other words, its critical section.

ORIGINAL PAGE IS OF POOR QUALITY

Attempting to solve this problem, German specialists in 1938 created a chamber having only internal cooling with the aid of several bands of fuel screens. Evidently,



/37

Fig. 9. Scheme of the ORM-65 rocket engine (in cross-section).

in the course of combustion testing they realized that internal cooling alone was not very effective, and on subsequent variants they augmented it with external cooling using alcohol. The solution to the problem of cooling was facilitated by adding water to the alcohol (a 75% aqueous solution of alcohol was used), which on the one hand reduced the combustion temperature and increased the cooling capacity of the alcohol, but on the other -- led to certain losses in the specific impulse. Thus was developed the cooling system used in the engine for the A-4 rocket. /38

A significant fuel expenditure for internal cooling led to a noticeable reduction in the economy of this liquid fuel rocket engine, which was a serious disadvantage of the selected cooling system. Soviet specialists were able to overcome this problem. In 1946, A. M. Isayev proposed, with the aid of additional injectors located along the periphery of the chamber head, to create in the pre-wall layer combustion products an excess of one of the fuel components, which would reduce the heat flow from the gases to the wall. This idea is widely used on all presently known space liquid fuel rocket engines. However, one solved problem regularly led to the necessity of solving another, no less complex, problem.

Its essence consisted of the following. In forcing the engine parameters (increasing the pressure in the chamber, utilization of more caloric fuel), which invariably led to an increase in the density of the heat flow coming from the combustion products to the chamber wall, a situation arose in which the wall could only be protected against burning through by making it thinner. But in order to ensure its durability, it was necessary on the contrary to increase its thickness (with increased pressure) or to leave it as before (with the use of more caloric fuel).

It turned out that this problem can be solved by connecting the internal and external walls of the chamber with frequent connections in the form of ribs on the inside wall or with intermediate corrugates. The expediency of this solution was at that time far from evident. Logic dictated that such a construction must necessarily disintegrate due to the varying thermal expansion of the internal "hot" and the external "cold" walls of the engine.

However, practice has shown that this construction works /39
well. It turned out that the outside wall withstands part of the load from the pressure of the cooling agent on the inside wall, which as a result could be made sufficiently thin. The chambers in which the internal and external walls are joined together by one means or another are called "joined".

A great input into their development was given by the collective directed by A. M. Isayev. They received intensive development in out country thanks to the work conducted at the GDL-OKB [Gazodinamicheskaya laboratoriya - Opytno konstruktorskoye byuro; Gas dynamic laboratory - Experimental design bureau]. In this organization, the so-called solder-welded chambers were first created and, most important, they were first used for solving problems in increasing the pressure of combustion products.

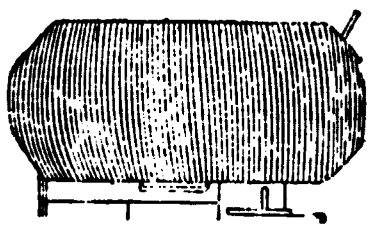


Fig. 10. Engine chamber by E. Zenger of tubular construction.

In the USA the problem of increasing pressure was solved by a somewhat different means. As early as the 30's, the German scientist E. Zenger created a number of engines with chambers made of one (or two) spirally wound tubes, along which the cooling agent passed during the process of the engine's operation (fig. 10), thus cooling the chamber. In the post-war years, Zenger worked in the USA and American specialists evidently became acquainted with his idea of the tubular chamber. However, they were suitable only for small liquid fuel rocket engines and were definitely unsuitable for engines with thrust of several tens of ton forces. With increased engine thrust, it became necessary at the same time to also increase the number of tubes from which the chamber was made.

The ultimate case of this increase became a construction made of a large number of longitudinally placed tubes. In order to keep this chamber from losing its stability under the effect of pressure from combustion products, it is reinforced on the outside with special bands (for example, wrapped with wire). Since the chamber and the nozzle have a rather complex configuration, the area of the tube cross-section varies along the length. The tubular construction chamber was first used in the USA in 1953, and since that /40 time it is used on all large American space engines. With the creation of "joined" chambers, the appearance of the cooling system for space liquid fuel rocket engines was generally determined. Here, for example, is how our country's RD-107 engine used on the first space rocket was cooled. Located at its head along 10 concentric circles were over 300 injectors for feeding fuel. At the periphery of the head were injectors which fed combustibles (kerosene) to the pre-wall layer of combustion products (i.e., for internal cooling). External cooling was provided by the flow of kerosene along solder-welded tract (jacket) from the nozzle to the head.

At the point where the area of the chamber's cross section was minimal (the critical section of the nozzle), while the heat flow to the wall, on the contrary, was maximal (reaching $14 \cdot 10^6$ kcal/m^{2}hr), the speed of the cooling agent comprised approximately 20 m/sec. Such cooling ensured preservation of the material part of the engine, despite the rather high temperature of fuel combustion (3250°C). The engine operated at a chamber pressure of 60 kG/cm^2, which made it possible to obtain a rate of gas emission of 2950 m/sec out of the nozzle.

This engine significantly surpassed in its characteristics all foreign made liquid fuel rocket engines of that time. It is enough to say that its specific impulse (the parameter characterizing the engine's economy, representing the ratio of thrust to fuel expenditure per second) was 30 sec. higher than for the same class of engine N-1, which was used as of 1966 at the first stage of the booster rocket "Saturn-1B".

The development of cosmonautics required the creation of more powerful and more economical liquid fuel rocket engines. The road toward this goal lie, in particular, through further increase in the chamber pressure, which automatically led to an increased heat flow to its wall and, consequently, also to the complication of the problem of heat protection. It should be noted that all the engines created in the 50's operated on the "non-closed" scheme: part of the fuel was burned in the gas generator, the obtained gas was directed to the turbine which drove the pumps, and afterwards was ejected into the surrounding environment. With a chamber pressure not exceeding 90 kG/cm^2 and engine thrust comprising approximately 150 ts, these /41
fuel losses were insignificant and equalled 1.5 - 2%. However, with further increase in pressure, they became evident, rapidly growing to unacceptable values.

The losses could be avoided by means of after-burning the gas coming out of the turbine in the main combustion chamber. With this scheme, the possibility arose for significantly increasing the pressure in the chamber, but the rather complex problem of cooling, among others, presented a hindrance. Despite all the difficulties, Soviet specialists in the mid-60's created a liquid fuel rocket engine operating according to such a scheme and used since 1965 in the booster rocket "Proton".

The pressure of combustion products in the chamber of this liquid fuel rocket engine is 2.5 times greater than in the RD-107 engine. The difficulties in solving the problem of its thermal protection were overcome partially due to the achievements in the sphere of ceramic refractory materials. This material was used to face the inside of this engine's chamber. The refractory surface was cooled by an internal gas-liquid film, created by means of feeding one of the fuel components from the cooling tract through a system of special openings in the wall. Moreover, the chamber was also cooled from the outside by the fuel component which flowed through its jacket. The channels for flow of the cooling agent were created with the aid of milling the inside wall of the chamber. The rips formed in this manner took an active part in transmitting heat from the wall to the cooling agent, intensifying this process.

In the USA the development of a "closed" scheme of liquid fuel rocket engine with chamber pressure on the order of 200 kG/cm^2 was delayed for many years. Having begun in the 60's, it was completed essentially only in 1981, when the first such liquid fuel rocket engine set off into space. The solution to the problem of cooling turned out to be particularly difficult for the specialists in creating the chamber, and this, of course, was not by accident. The density of the heat flow coming to the chamber wall on such an engine, operating on hydrogen-oxygen fuel, reached $140 \cdot 10^6$ $kcal/m^2hr$. Therefore, the very first preliminary works showed that it is necessary to seek more effective methods of cooling than those which had been used previously.

In 1963 at the "Pratt-Whitney" company, an engine was /42 designed whose small-scale models (with thrust of 5 ts) were used to develop the so-called transpirational cooling of the combustion chamber and the nozzle opening in combination with flow-through cooling os the super-critical (expanding) part of the nozzle. The notion of transpiration cooling belongs to the American engineer R. Hoddard, who conducted combustion testing of a liquid fuel rocket engine in 1923. The combustion chamber and nozzle of this engine were made of a ceramic porous material. Liquid oxygen was fed into the cooling jacket, which was "pushed through" due to the excess pressure through the porous wall into the chamber, so that during the engine's operation there was cold oxygen near the wall.

Work on this type of cooling was discontinued, and was resumed in the USA only in the late 40's, when the "Aerojet" company began the development of a hydrogen-oxygen liquid fuel rocket engine for a booster rocket. However, these developments were also discontinued.

In 1966, the "Pratt-Whitney" company began the development of a transpiration cooled hydrogen-oxygen liquid fuel rocket engine, the XLR-129, with thrust of 115 ts, which was viewed for some time as an engine for a prospective transport apparatus.

However, this engine did not find practical application, and was unable to compete with the liquid fuel rocket engine produced by the "Rocketdyne" company. All the latest scientific technical achievements were utilized on this company's engine in solving the cooling problem. First of all it turned out that the tubular construction of the cooling tract which had been used up to this time on practically all American liquid fuel rocket engines was unsuitable in engines with high heat flows to the wall. This is explained by the circumstance that the tubes have a form close to oval, and consequently in their utilization it is impossible to make a smooth inside chamber surface. As a result, this surface turns out to be overdeveloped, having an excessively large heat accepting area. This disadvantage is not present in chamber designs used on Soviet engines. Presented in fig. 11 is a comparison of the wall temperatures of a tubular and milled cooling tracts, from which it is clearly evident that with all other conditions being equal, the superiority of the latter is noticeable.

As early as the beginning of the 30's, results were /43 published on the study of effects of rough tube surfaces on the speed distribution in the boundary cooling agent layer, i.e., essentially the effects of friction. These studies at the same time indicated that on a rough surface the heat exchange to the cooling agent will be different than in the case of a smooth surface. A detailed study of this question convincingly showed that the creation of a certain rough texture in the channels intensifies the process of heat transfer to the cooling agent. On modern liquid fuel rocket engines this effect is often considered in solving the cooling problem.

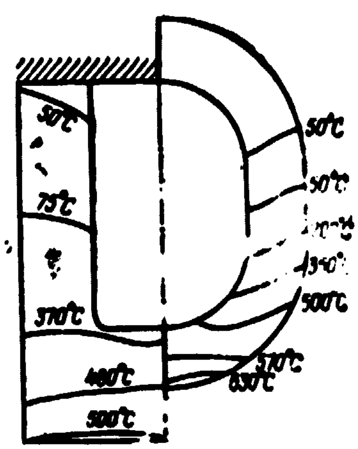


Fig. 11. A comparison of the temperatures of a tubular and milled cooling tracts.

If the tube is bent in a circular fashion and the cooling agent passed through it, then the intensity of the heat transfer will be greater on that side toward which the centrifugal force from the movement of this cooling agent is directed. Studies have shown that with the presence of proper curvature of the cooling tract, the heat transfer to the cooling agent may increase by approximately 50% as compared with that which occurs in rectilinear channels. This phenomenon also found its application in the liquid fuel rocket engine. For this purpose, the direction of movement of the cooling agent is selected running from the nozzle to the head. In the region of the nozzle's critical section, where the density of the heat flow to the wall is maximal, the heat transfer is significantly increased due to the curvature of the cooling tract, and the zone of increase extends to several tens of millimeters above this section.

We have already examined some of the early works in the introduction of internal cooling of chambers in liquid fuel rocket engines. The appearance of internal cooling had an inestimable input into rocket engine construction, facilitating the solution of rather complex problems in the heat protection of liquid fuel rocket engines. At the same time, hydrogen-oxygen engines, generally speaking, /44 may not even have such cooling. This, of course, is a sort of return to the past, already utilized earlier, technical solution. However, this return was already on a higher technological level. In order to understand this, let us briefly examine certain specifics of intra-chamber processes.

We know that the combusion of 1 kg of hydrogen requires 8 kg of oxygen. In this case, the temperature in the chamber of the liquid fuel rocket engine, and consequently also the heat flow (from the products of combustion) to its wall are maximal. This, however, does not mean that the maximal rate of gas emission will be achieved. The fact is that this rate depends not only on the temperature, but also on the molecular weight of the combustion products and at a given temperature will be greater with lower weight. Hydrogen-oxygen fuel is expediently used with a certain hydrogen excess.

In this case the combustion temperature is somewhat reduced, while the rate of emission, on the other hand, increases due to the reduction in the molecular weight of the combustion products. This effect may expediently be utilized on the liquid fuel rocket engine: along the entire cross-section of the chamber it is possible to create the same optimal (i.e., with the proper excess of hydrogen, giving maximal emission rate) ratio of fuel components, so that on the wall of the chamber there will not be a clearly expressed pre-wall "cold" layer, although there will be a excess of combustible on it.

In solving the problem of cooling, great attention is given to the selection of the form and dimensions of the chamber and the nozzle. This selection is done in such a way as to ensure minimal hydraulic losses in the cooling tract, minimal engine weight and a heat flow from the combustion products to its wall. We will note that with movement of the gas flow, a so-called marginal layer is formed neat the wall, in which all the flow parameters undergo a change. There are changes, for example, in the temperature and speed from the values at the wall to the corresponding values in the flow nucleus or, as is also said, on the outside of the marginal layer.

This layer has a certain heat resistance: the thicker it is, the greater its resistance. This means that the density of the heat flow to the wall in this case will be /45 less. The marginal layer evens out gradually along the length of the chamber -- its thickness turns out to equal to zero somewhere in the region of the head, and increases with proximity to the nozzle. Since thickening of the marginal layer leads to a reduction in the density of the heat flow to the wall, it is expedient from the standpoint of heat transfer to make the chamber longer so that the density of this flow will be less in the region of the nozzle's critical section.

However, with increase in this length, there is a simultaneous increase in the total heat flow to the wall, as well as the length of the cooling tract and, consequently, the hydraulic losses. As a result, designers are faced with the necessity of a rational compromise between the reduction and density of the heat flow in the critical section of the nozzle and the hydraulic losses in the cooling jacket.

The ratio of the chamber's cross-section area to the area of critical section of the nozzle, and the angle of narrowing of the supercritical (narrowing) part of the nozzle also effect the thickness of the marginal layer, and consequently also the density of the heat flow to the wall of the nozzle. Moreover, the nozzle dimensions effect the total heat flow to this wall: with a lesser area this flow will be less, with a greater area -- it will be correspondingly greater. The selection of values for all these parameters which are favorable from the standpoint of heat transfer is an important step in the path toward solving the problem of heat protection of liquid fuel rocket engines.

Thus, the solution to the problem of cooling liquid fuel rocket engines in no way is reduced to merely the selection of the necessary value for rate of flow of cooling agent along the cooling tract. It leaves its imprint on practically all aspects of combustion chamber design (selection of their length, form, etc.).

LARGE REQUIREMENTS FOR SMALL LIQUID FUEL ROCKET ENGINES

With the appearance of large spacecraft, the necessity has arisen for having small engines intended for controlling the motion of these craft along trajectories and, in particular, for their orientation and stabilization, for correcting flight speed and trajectories, for performing maneuvers in orbit, etc. The requirements for such space engines are /46 in many ways different from those which are presented for the previously examined liquid fuel engines of booster rockets. In this case there is the necessity for repeated starts under conditions of weightlessness and space vacuum, the impulse regime of operation, the prolonged presence in space flight, etc.

These requirements finally have led to the situation where, instead of the flow-through cooling used in the large liquid fuel rocket engines, the small space engines began to use other, simpler methods. In the late 50's - early 60's, practical work was begun on the application of radiation cooling in space liquid fuel rocket engines. This method essentially consisted of the following. By the selection of low caloric fuel, organization of combustion with a large excess of one of its components and with a relatively low pressure in the chamber, low temperature of the combustion products is achieved. The combustion chamber of the liquid fuel rocket engine is made of heat resistant material capable of withstanding the pressure of the combustion products at a temperature on the order of 1800°C. At such a temperature the heat flow coming to the engine wall is radiated by it into space in such a way that a thermal balance ensues: as much heat as comes to the wall is irradiated into space. In the 30's, the pioneers of rocket technology dreamed of the possibility of utilizing such a method. However, at that time there were no materials in production capable of working at such high temperatures and such cooling could not find application then.

In the 40's, under the influence of the needs of atomic energy, aviation, chemical machine building, radio electronics, and certain other areas, extensive work was begun on obtaining heat resistant alloys of tantalum, niobium, and molybdenum. By the late 50's, i.e., essentially by the time that the need for the use of these new materials arose in space technology, the latter were at a sufficiently high level of their development. In the early 60's, liquid fuel rocket engine chambers made of tantalum allow 90Ta-10-W began to be tested, as well as those made of molybdenum alloys with an anti-oxidation surfacing.

The assortment of construction materials used in space liquid fuel rocket engines is constantly encreasing, which makes it possible for designers to improve the qualitative characteristics of these engines. Thus, for example, in the late 60's - early 70's, specialists from the "Rocketdyne" company decided to attempt to use beryllium which, aside from its high fusion temperature, also has high thermal conductivity. On the inside wall of the liquid fuel rocket engine chamber there was created a combustible film for internal cooling, and only radiation cooling was proposed for cooling the nozzle. The experiments conducted clearly showed that in such a construction a noticeable part of the heat is transmitted by heat conductivity along the beryllium from the hottest part of the engine (the critical section of the nozzle) to its "cooler" parts, i.e., to the part of the chamber cooled by the internal method. As a result, the wall temperature in the region of the nozzle's critical section turned out to be lower than the expected value, and this made it possible to improve the economy of the engine. /47

Of great significance in solving the problem of heat protection for liquid fuel rocket engines were the new ceramic refractory materials. Work on the discovery of these had begun in our country as early as the late 30's. The main breakthroughs in this area were achieved beginning in the 50's.

The emergence of new refractory materials turned the attention of space technology designers to such widespread but unsuccessfully applied heat production methods used in the 30's as heat absorption and thermoinsulation. In the first case the heat coming from the combustion products is accumulated by the wall and the engine only works until its temperature reaches the ultimate margin for the given material. In the second case, the refractory material insulates the wall, having an allowable temperature higher than the temperature of the combustion products. In heat absorption, materials having a high heat capacity and heat conductivity are required, and in thermoinsulation, on the contrary -- materials with relatively low heat conductivity.

It should be noted that the method of heat absorption is presently used only on certain experimental liquid fuel rocket engines. As concerns the method of insulation, it is widely used, and often in combination with flow-through cooling.

The continuous search for new means of cooling space liquid fuel rocket engines has led in a number of cases to the emergence of rather original technical solutions.

As early as 1933, Zenger in his "Program of Experiments with Rocket Engines with Constant Pressure in the Combustion Chamber" proposed adding an "appropriate substance" to the fuel, which precipitates on the wall during combustion so that its regeneration occurs. He proposed using iron carbonyl or asphalt as this substance. /48

This method did not find practical application for a long time. In the first post-war years, studies of this method of heat protection began to be conducted by specialists at the "General Electric" company, who analyzed two variants: the natural formation of a heat protective carbon layer on the wall during combustion of hydrocarbon fuels, and the formation of this layer with the addition of special silicone organic substances to other types of fuel (for example, alcohol and liquid oxygen).

The results of the experiments, in particular, showed that at pressures of up to 70 kG/cm^2 in the combustion chamber, it is possible to achieve a 30% reduction in the heat flow to the wall (with the addition of 1% by weight of silicone substances to the fuel) with practically unchanged specific impulse. This method was used by specialists at the "Bell" company in the 70's for the correction engine installation of the "Agena" rocket stage.

At the initial stage of development of liquid fuel rocket engines, the pioneers of rocket technology practically did not conduct scientific research aimed at studying the specifics of intra-chamber processes. In the 30's, for example, it was not even known that with emergence of high frequency instability in the chamber, the value of the heat flow to the engine wall increases. Not knowing this specific characteristic often lead to the situation where, when the engine wall burned through the researchers began to improve its cooling, rather than the combustion process, which was the true "culprit" for the burn-through.

However, the gap between theory and practice in rocket engine building could not continue for long -- it became ever more difficult, and in a number of cases simply impossible, to develop engines on the basis of intuition or common sense of the researchers alone. As a result, in the 40's, theoretical and experimental research studies in the area of liquid fuel rocket engines began in various countries, /49 and particularly expanded in the years after the second world war.

New knowledge concerning the specifics of processes taking place in liquid fuel rocket engines also facilitated the solution of the problem of heat protection for designers. Thus, for example, in the early 60's during the study of small engines, experiments clearly showed a significant reduction in the heat flow to the wall with the application of a relatively great angle of constriction of the supercritical part of the nozzle. As a result, yet another method facilitating the solution to the problem of liquid fuel rocket engine heat protection was found.

It should be noted that in developing liquid fuel rocket engines for booster rockets, many of the methods of heat protection are used which aid in protecting the material part of the liquid fuel rocket engines of the spacecraft themselves. In the 60's - 70's the expanding parts (fittings) of the nozzles of large engines were often made without external flow-through cooling. Thus, for example, the super-critical (expanding) part of the nozzle of the American liquid fuel rocket engine F-1, starting from the area where the degree of expansion equalled 10, was made of a heat resistant alloy and was cooled only by an internal pre-wall layer of gas directed to the wall after output from the turbine pump assembly.

The nozzle fitting of the LR-81-BA-9 (model 8096) engine was made of a titanium alloy and had an internal covering made of refractory materials. On the engine for the second stage booster rocket of "Titan-2", the expanding part of the nozzle was made of ablation material based on epoxy and phenolic resins. On the liquid fuel rocket engine AJ-10-104 of the rocket stage "Able Star", the nozzle fitting was made of a titanium alloy and had radiation cooling.

The example for the practical application of heat protection by deposit (i.e., the method based on the precipitation of a constantly renewing surfacing on the wall) in small liquid fuel rocket engines has already been given. This method is also applicable for engines intended for booster rockets. In the 70's it was used to implement the heat protection for the main engine of the "Agena" rocket stage, which made it possible to switch to the use of a denser oxidant on it and to increase its specific impulse by almost 2 sec (19.6 m/sec).

It has already been noted that the heat protection of /50 descending spacecraft is implemented with the aid of ablation cooling. This method is widely used also in space engines. In recent years, rocket engines operating on solid fuel have become widespread in space technology. In particular, they are used as the acceleration stages of booster rockets and have a relatively low heat intensity and a short time for continuous operation. Their heat protection is implemented with the aid of the same methods as the heat protection of liquid fuel rocket engines on spacecraft: insulation, ablation cooling, heat absorption noted earlier, etc.

However, in a number of cases, certain original methods are also used on solid fuel engines. Here, for example, is one of them. The wall in the region of the nozzle's critical section is made of a heat resistant porous material (tungsten, graphite, etc.), impregnated with some filler (copper, teflon, silver, etc.). During the engine's operation, the temperature of the wall will not exceed a certain maximal value, since the heat will be absorbed due to the heating, fusion and vaporization of the fillter which also, having vaporized into a gas, will come out of the pores of the wall, creating a "cold" pre-wall layer.

The search for new methods of cooling rocket engines is being conducted along a broad front, leading from time to time to success. Every year in various countries, a large number of inventions appear in this area, in which rather original technical solutions are often proposed. At the same time, despite the abundance of new proposals, only a few of them find practical application. Evidently the time has come when all which is expedient has already been discovered, tested, and introduced. Specialists have only to improve upon that which is already known, and each year there are fewer hopes for the discovery of a principally new method of cooling. However, there are no boundaries to discovery and may time and effort will still bear their fruit, as has been the case many times in the history of science and technology.

SPACE BEGINS ON EARTH /51

In order for booster rockets, as well as spacecraft, to satisfy the requirements of reliability, durability and operability, they are thoroughly and comprehensively tested on Earth under conditions which maximally correspond to those during launch and in the course of orbital flight. Specialists are particularly interested in answers to the question of how the "sensitive" instruments will withstand vibration and overloads, what is the degree of air tightness of the spacecraft, how does the system of single feed operate, how various materials behave in deep vacuum at extremal temperatures, and numerous other questions.

An important place in the earthbound development of spacecraft belongs to the study of their thermal regime. Rigid weight and energy limitations force designers to create systems of thermoregulation without significant reserves of cold- and heat productivity. Under these conditions, even insignificant errors in the thermal computations may lead to the situation where the thermal regime of the spacecraft will not be maintained within the given limits and will cause a breakdown of the on-board instrumentation elements.

As illustrated in fig. 12, the specifics of the thermal regime (even within the framework of the allowable range) greatly effect the reliability of the spacecraft. The slightest number of malfunctions is observed at normal, room temperature. With its reduction, the number of malfunctions increases, becoming significant at reduced temperatures and particularly great at increased temperatures.

To conduct thermal studies on Earth in special pressure chambers (fig. 13), certain conditions of space are recreated, primarily pressure, temperature and solar radiation. The imitation of these conditions in full measure is complex. Therefore, in practice, in practice usually one or another degree of approximation is accepted. Thus, for example, already at a pressure of 10^{-8} - 10^{-10} kG/cm^2 the thermal conductivity of the gases becomes small enough to overlook, and it may be discounted. Heat removal from the spacecraft to the surrounding environment in this case will take place in the same way as into space -- only by irradiation.

The temperature of cosmic space is imitated with an /52
accuracy acceptable for practical purposes by means of cooling the internal walls (screens) or the pressure chambers with liquid nitrogen (77 K). From the inside these screens are covered with special surfacings which ensure a degree of blackness close to one. This is done so that the heat flow irradiated by the object is absorbed by the wall and not reflected back to the object.

Generally speaking, a number of specialists believe that the imitation of actual space conditions may possibly never be achieved, but any degree of appromation to them is worth the expended effort. However, in practice, a rational degree of approximation is always selected, determined by a certain intersection of the technical feasibility and the economic expediency of creating the experimental installations on earth.

At the initial stage of development of cosmonautics, the earth-bound experimental base was relatively weak. In the USA, for example, the first installation intended for conducting thermal testing of spacecraft was built only in 1958 and was rather primitive. It consisted of a chamber

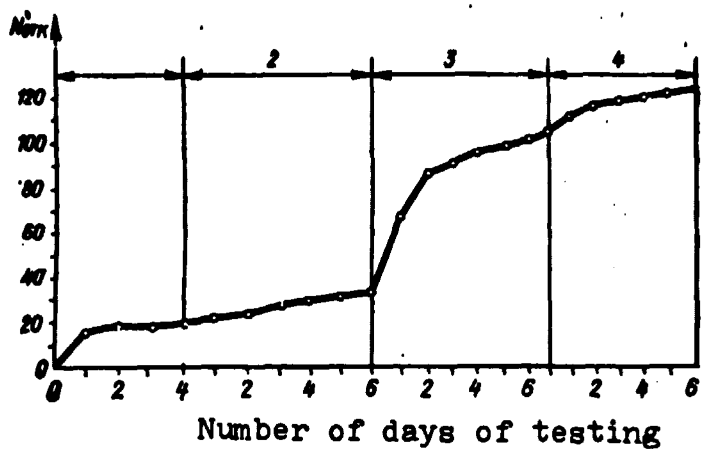


Fig. 12. Dependence of the number of instrument malfunctions on 11 spacecraft on the surrounding temperature:

1 - room temperature; 2 - transition phase from maximally allowable to minimally allowable instrument temperature; 3 - minimally allowable instrument temperature; 4 - maximally allowable instrument temperature.

2.4 m in diameter and 4.6 m long. Its screens were cooled /53 by liquid nitrogen flowing over them. With the aid of three mechanical and one diffusion pump, it was possible to maintain a pressure on the order of 10^{-11} - 10^{-12} kg/cm^2 within it. The imitation of the external heat flow coming to the craft placed in the chamber was done with the aid of infrared heaters which quantitatively reproduced the flows obtained by analytical computation.

Practical experience, however, soon showed that the experimental technology must be more developed primarily in the aspect of imitating the external heat flows. As a result, in the 60's, work on the creation of solar radiation imitators was begun in various countries. The application of these solar radiation imitators, aside from a more comprehensive study of the heat regimes of spacecraft, made it possible also to solve a wide range of other rather important problems: to test optical devices of the orientation system and solar batteries supplying the craft with electrical energy, to study the effect of the Sun's radiation on the properties of materials, etc.

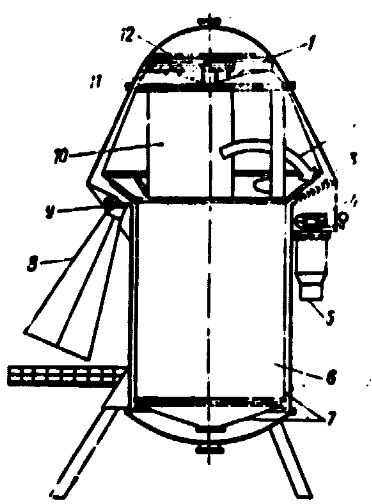


Fig. 13. Scheme of a pressure chamber (right half is conditionally turned at a 45° angle to show the diffusion pump):

1 - beam construction to which collimation mirrors are suspended 11; 2 - direction of air flow during vacuum formation; 3 - diffusion pump; 6 - working part of chamber; 7 - wall cooled by liquid nitrogen; 8 - installation for imitating solar radiation; 9 - mosaic lens system; 10 - wall cooled by liquid nitrogen; 11 - collimating mirrors; 12 - service platform.

First used as irradiation sources were carbon-arc lamps, which imitated the spectral distribution of the Sun's energy rather well in the entire range of wave lengths except the /54 ultraviolet sphere (0.2 - 0.4 μm), where the power insufficiency could be compensated with the aid of additional sources. However, this type of lamp had a number of significant disadvantages. In particular, due to the rapid burning of the positive electrode (for an arc with capacity of 10 kW the speed of burning comprised 0.5 m/hr), it was constantly necessary to replace it with a new one. Moreover, the mechanism of electrode feed was complex and it was necessary to protect the elements of the optical system (mirrors reflecting and refracting the light, lenses, etc.) against pollution by the combustion products.

Therefore, gas filled lamps began to be used (xenon, mercury-xenon), which combined the brightness of the carbon arc with ease of operation. At the same time, even these lamps had disadvantages. Thus, for example, they imitated the solar spectrum worse than the carbon-arc lamps. In the process of development, both types of lamps were improved: a non-expendable negative electrode was developed in the carbon-arc lamp, the imitation of the solar spectrum was improved due to the introduction of additional gases into xenon and mercury-xenon lamps, etc.

To create the necessary intensity of heat flow, a large number of lamps was used, located in such a way that the test object had no shade or heavily heated spots on it due to the mutual overlap of rays from individual illuminators. The light rays were directed to the chamber with the aid of special optical systems distinguished by their wide variability of design. Here, for example, is the design of an optical system of one of the model installations (fig. 14).

The flow of rays is created by the solar imitator, located in a 10-meter addition above the cylindrical pressure chamber with height on the order of 14 m and diameter of 8 m. The source of radiation comes from mercury-xenon lamps (9) with capacity of 2.5 kW each, located outside the chamber on a panel 1. There are over 130 such lamps, and the rays from them are collected by a paraboloid mirror (2) and directed to a convex mirror 3. The latter consists of 19 hexagons with the diameter of their circumferences equal to 0.165 m. By this mirror the beam of rays is directed through a lens (4) and comes to the dispersion mirror (5) /55 which in turn sends these rays to the main paraboloid mirror (6) which forms the flow in the working zone of the chamber (7).

Each lamp illuminates its own sector in the working zone and is equipped with a device for regulation from the main control panel. The mirror (5) has a diameter of 0.765 m and consists of 1150 individual water-cooled paraboloid mirrors made of stainless steel. Since mirror (5) creates a shadow area in the working zone, there is an auxiliary optical system which illuminates this sector (not shown in fig. 14).

The increased dimensions of spacecraft reasonably necessitated the creation of larger volume chambers. In the early 60's, chambers with volume of over 50 m^3, and even over 500 m^3 began to appear. By the end of the 60's, in the USA, for example, there were 14 chambers with volume exceeding 1000 m^3 (the chamber intended for testing the space ship "Apollo" had a volume of 11,233 m^3).

The methods of imitating temperature and external heat flows as well as the methods of creating a space vacuum in such installations underwent significant changes. In actuality, in the larger installations it is necessary to have,

for example, higher rates of pumping off gases, since the internal surfaces of the chamber walls emit vapors and gases in a vacuum whose amount with all other conditions equal are directly proportional to the size of the chamber. Moreover, /56
in large installations, as a rule, there is a significant seal length through which aid penetrates into the pressure chamber. Finally, the amount of emitted vapors and gases is affected by the auxiliary equipment and the dimensions of the test objects, having in most cases materials with high gas separation (all organic materials, rubber, etc.).

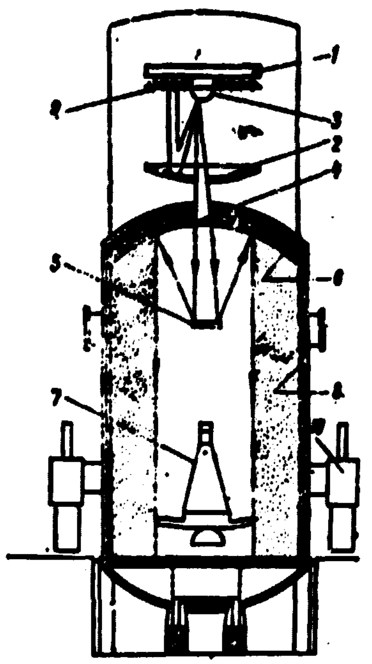


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Fig. 14. The optical system of an experimental vacuum installation.

However, maintaining the necessary level of vacuum in large chambers by means of increasing the rate of evacuation with the aid of pumps becomes technically complex, and therefore the solution to this problem proceeded along a different route -- with the aid of cryogenic evacuation. For this purpose, sections (cryogenic panels) were provided in the chamber which were cooled by liquid hydrogen (the boiling point at normal pressure comprises 20 K) or gaseous helium (11 K). The molecules of the residual gas, falling on these panels, are "frozen", which leads to a reduction of pressure in the chamber. The cryogenic panels are located in the space between another screen, which is cooled by liquid nitrogen.

Cooling all the screens of the chamber entirely with helium is technically complex and economically inexpedient, since in this case in particular, the initial cost of the device and the operational expenditures become rather great. Using cryogenic panels makes it possible to solve the problem of maintaining the necessary vacuum in the chambers with minimal expenditures. Diffusion pumps are most often used as the primary, although in a number of cases other forms are also used: ion-sorption, titanium sublimation, turbomolecular, etc.

Since the cost of spacecraft is rather high, the heat tests are sometimes conducted on special thermal mock-ups, on which imitators are used instead of instrumentation. On such a mock-up, temperature indicators are located at various places so that by their indications a full understanding of the spacecraft's thermal regime may be compiled. The thermal mock-up is placed in the pressure chamber, the gases are evacuated from the chamber and its screens cooled. Then the solar imitators are switched on and the tests begun.

With the aid of a special device, the mock-up is rotated, imitating the change in the craft's position relative to the /57
Sun in the course of its space flight. The instrumentation imitators operate according to given programs, reproducing the heat emission of the instruments at various regimes of the "flight". The indicator readings are automatically

recorded on special instruments. If it is determined in the course of testing that the thermoregulation system operates unsatisfactorily, the appropriate changes are introduced into its design, and if necessary the tests are repeated to determine the effectiveness of these changes.

Also conducted under earth conditions is the imitation of the thermal regime of spacecraft on flight sectors within the atmosphere of planets. In this case, two main parameters are reproduced in the temperature and pressure chambers: pressure ("altitude") of the surrounding medium and the surface temperature of the spacecraft. Heating of the surface of the tested objects is done by special heaters, for example infrared tungsten-quartz radiation heaters. Such heaters consist of tubes made of quartz powder, with a tungsten filament woven around them. They are mounted in ceramic reflectors which reflect the heat flows. The existing heaters have a filament temperature of 3000°C. Quartz lamps, graphite casings and other devices are also used as heaters.

A particularly complex problem in conducting this type of test consists of imitating the change in the surrounding pressure. During the launch of the spacecraft, the reduction in air pressure from atmospheric to 10^{-8} kg/cm^2 occurs in several minutes. Changing so rapidly the pressure in the vacuum chamber within a real time scale is rather complex, since most space imitators are not equipped with vacuum pump systems ensuring such evacuation rates. Therefore, in practice this problem is solved by various approximated methods based, as a rule, on changing the time scale.

An experimental study of aerodynamic heating in a number of cases led to obtaining rather valuable results. Thus, for example, American specialists were able with the aid of its imitation to find the answer to the question of why in the process of flight one rocket systematically suffered /58 destruction of the honeycomb construction lining made of fiberglass, impregnated with phenolic resin. The results of the experiments clearly showed that the reason for this consisted in a sharp increase in the pressure of gases inside the lining due to weak gas conductivity of its material.

Ground-based development also plays a great role in ensuring the heat regime of the rocket engines.

Firing tests are conducted in special test complexes, which for large liquid fuel rocket engines are rather complex engineering structures which include a warehouse for fuel components and shops for the production of compressed gas, systems for provision of electrical energy and firing stands, research laboratories and production buildings, etc.

Firing tests of engines in vacuum installations imitating the conditions in cosmic space are conducted for solving a number of problems. Thus, for example, studies of the heat regime of engines with radiation cooling are conducted in such installations, the behavior of ablation surfaces of chambers are studied, the actual temperatures of fuel tanks are determined, etc. Ground-based development makes it possible to find and correct design errors and thereby to increase the reliability of fulfilling expensive space programs.

Thus, concluding the examination of the methods of heat protection, we will note that the development of rocket space technology is still closer to its beginning than to its end. There are still many new and complex problems ahead whose solutions will inevitably lead to the emergence of original technical solutions, including those in the sphere of thermal protection.

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