On the issue of reducing the temperature heating of the airframe structure of a supersonic aircraft

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Abstract. In this paper, the influence of heating of the structure on its strength is considered in three aspects: a decrease in the mechanical characteristics of structural materials at high temperature, heating of the structure can lead to temperature stresses in its elements and elements of aircraft systems and the occurrence of the phenomenon of creep of structural materials - continuous plastic deformation of structural elements under constant load. There are several options for reducing heat flows due to modification of the basic aerodynamic layout: increasing the sweep angles of the bearing surfaces and the radius of the wing tip and the front edges of the inflows, reducing the area of the washed surface due to the "flattening" of the initial aerodynamic contours. Each of the considered existing and promising methods of thermal protection separately will not allow to fully develop thermal protection of promising reusable supersonic aircraft. A comprehensive approach is needed. At the same time, it is necessary to consider both the replacement of structural materials of panels and cladding with titanium or steel alloys, as well as design solutions for the use of fuel lines.

1 Introduction

To date, the operation of supersonic commercial aircraft, it would seem, is the most advanced and in-demand industry in aviation [1], but for a number of reasons, the use of this type of transport in the modern world has been completely suspended.

First of all, for passenger flights, such criteria as comfort, speed, flight safety are important [2,3], and the cost of the flight is not a little important factor. As part of a study on the efficiency and convenience of supersonic aircraft of the first generation, it was noticed that this class of transportation is less in demand among a large circle of people. Also, technical aspects were a serious problem, which did not allow to fully realize the potential of these aircraft.
The current level of technology development may lead to a revision of the attitude towards supersonic transport in the long term. Thus, a number of social studies were conducted, which showed that the speed of transportation at the present time is a key criterion for choosing transport for long-distance trips. Based on the data on transportation in the world, a list of promising routes for supersonic transport has been determined, and the dynamics of the corresponding transportation in the long term has been modeled.

Leading design bureaus are faced with the task of designing a second-generation supersonic passenger aircraft capable of overcoming the problems of the last generation of this class of aircraft. The problem of thermal heating of the structure is significant and, in this paper, we will consider options for its solution.

2 Problem statement

When aviation reached high supersonic speeds, the task arose – to ensure the strength of the structure under heating conditions. When flying at such speeds, the temperature of the air flowing around the surface of the aircraft increases, which leads to heating of the structure itself (the so-called "aerodynamic heating"). Air heating occurs in the shock waves formed around the aircraft, as well as in the boundary layer, due to the friction of the surface against the air. In addition, the heating of the structure comes from solar and atmospheric radiation, from engines and equipment.

At flight altitudes $H \leq 50$ km, the heating from solar and atmospheric radiation is negligible compared to the heat flow from the boundary layer, therefore, for aircraft flying in dense layers of the atmosphere, the main external source of heating is the boundary layer of air. The heat released at the surface partially enters the airframe structure, partially is transferred to the surrounding air mass, the temperature of which is equal to the air temperature at a given flight altitude ($T_A$). As is known, heat exchange is carried out in three main types Fig. 1:

![Fig. 1. Types of heat exchange processes.](image)

The temperature of the aircraft skin ($T_s$) can be determined from the heat balance equation, which is based on the result of summation of various heat fluxes.

With an approximate assessment of the skin temperature, it is assumed that its temperature regime is determined by the following heat flows Fig. 2: $q_c$ – convective flow from the boundary layer to the skin; $q_r$ – heat flow due to the radiation of the skin directed from the skin into the surrounding space; $q_f$ – heat flow entering the airframe structure (mainly determined by thermal conductivity).

The considered heat flows $q_f$ characterize the inflow (departure) of heat through a unit of surface area per unit of time and are called "heat flux density", W/m². Thus, the expression has the form:

$$q_f = q_c - q_r$$

(1)
The considered heat flows $q_f$, $q_c$, and $q_r$ have the form of surface area per unit of time and are called "heat flux density", $W/m^2$. Thus, the main source of heating is the boundary layer (mainly determined by thermal conductivity of the skin into the surrounding space; from the boundary layer to the skin; temperature regime is determined by the following heat fluxes).

Accordingly, the problem of thermal heating of the structure is significant for leading design bureaus for supersonic passenger aircraft capable of overcoming the problems of the last generation of supersonic transport. The current level of technology development may lead to a significant increase in the number of social studies related to supersonic transport in the long term. Thus, a number of social studies were conducted, which showed that the speed of transportation at the present time is a key criterion for choosing transport for long distance trips. Based on the data on transportation structures for a supersonic transport can be determined, and for the chosen criterion for choosing transport for long distance trips.

Fig. 1. Three main types of heat exchange processes.

The convective flow density is determined by the formula (2)

$$q_c = \alpha (T_r - T_s)$$  \hspace{1cm} (2)

where: $\alpha$ – heat transfer coefficient from the boundary layer to the skin;
$T_s$ – skin temperature;
$T_r$ – the temperature of the boundary layer near the skin surface is determined by the formula (3).

$$T_r = T_A(1 + r \frac{k+1}{2} M^2)$$  \hspace{1cm} (3)

where: $r$ – temperature recovery coefficient depending on the state of the boundary layer (laminar or turbulent);
$k$ – adiabatic index (for air $\kappa = 1.4$);
$T_A$ – ambient air temperature.

For a turbulent boundary layer, the skin temperature is determined by the formula (4).

$$T_r = T_A(1 + 0.18 M^2)$$  \hspace{1cm} (4)

The density of the heat flux of radiation from the surface of the aircraft is determined by the Stefan-Boltzmann law according to the formula (5).

$$q_r = \varepsilon \cdot \sigma \cdot T_s^4$$  \hspace{1cm} (5)

where: $\varepsilon$ – the radiation coefficient, which depends on the material, the state of the surface and the temperature of the skin (its value lies in the range $0 \leq \varepsilon \leq 1$).

As can be seen from the formula for $q_r$, the radiation flux into the surrounding space depends on the radiating ability of the surface and the temperature of the skin. Up to the numbers $M = 2.0 \ldots 2.3$ and up to the flight heights $H = 10$ km, the magnitude of the radiation fluxes is insignificant. As the flight speed increases, the radiation flux increases. Accordingly, $\Delta T = T_r - T_s$ grows. Therefore, to reduce the heating of the structure, even at large Mach numbers, the flight is carried out at $H = 30$ km.

Fig. 2. Heat flows.

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The density of the heat flux that removes heat from the surface into the structure is determined by Fourier's law. If we assume that a noticeable change in temperature occurs only along the thickness of the skin (in the direction of the x axis), and in the other two directions (y and z) the temperature does not change, then the flow into the structure can be expressed by the formula (6).

\[ q_f = -\lambda \frac{dT}{dx} \quad (6) \]

where: 
\( \lambda \) – coefficient of thermal conductivity of the material; 
\( \frac{dT}{dx} \) – temperature gradient characterizing the temperature change along the wall thickness.

Minus means that the heat flow goes from a zone with a high temperature to a zone with a lower temperature.

Thus, the expression (1) can be represented in the following

\[ \alpha(T_r - T_s) - \varepsilon \cdot \sigma \cdot T_s^4 = -\lambda \frac{dT}{dx} \quad (7) \]

The differential equation of thermal conductivity for the accepted one-dimensional process is found by the formula (8).

\[ \frac{dT_s}{d\tau} = \frac{\lambda}{\rho C} \frac{d^2T_s}{dx^2} \quad (8) \]

where: 
\( \frac{dT_s}{d\tau} \) – changing of the skin temperature over time (\( \tau \)); 
\( \rho \) и \( C \) – accordingly, the density and specific heat capacity of the cladding material.

The main term of the heat balance equation (7), which determines the temperature of the structure (skin) in a wide range of speeds and altitudes of flight, is the convective heat flow \( q_c \). The heat transfer coefficient \( \alpha \) depends on the flight altitude \((\rho H)\), the number \( M \) of flight, the sweep angle of the leading edge of the wing.

If we assume that the covering is insulated from the rest of the structure, after heating, when the temperature on its thickness will be the same \((T_{s1} = T_{s2} = T_{s3})\), the maximum steady-state temperature can be determined from equation (9).

\[ \alpha(T_r - T_s) - \varepsilon \cdot \sigma \cdot T_s^4 = 0 \quad (9) \]

During flight, heating occurs as a result of adiabatic compression of the air in front of the flying aircraft. In this case, the full braking temperature in front of the leading edges will be equal (for clarity, the temperature will be expressed in degrees Celsius).

\[ t_b = T_A(1 + 0.2M^2) - 273.2 \quad (10) \]

where \( T_A \) – absolute air temperature at flight altitude, K.

For flying in the stratosphere \((H \geq 11000 \text{ m})\),

\[ t_b = 216.7 + 43.3M^2 - 273.2 = 43.3M^2 - 56.5 \quad (11) \]

For a more accurate understanding of the process pattern, we give an example of \( t_b \) on the leading edge of the wing and the skin temperature \((t_b \approx t_w)\) located at a distance of 1 m \((H \geq 11000 \text{ m})\).
During flight, heating occurs as a result of adiabatic compression of the air in front of the leading edge of the wing and the skin temperature. The expression (1) can be represented in the following form:

\[ t_0 = t_b - \delta + \frac{\epsilon}{\lambda t_b} \]

The heat transfer coefficient \( \alpha \) depends on the flight altitude (state temperature can be determined from equation (9)).

\[ \alpha = \frac{1}{\frac{t_b}{\rho C}} \]

The effect of heating the structure on its strength is considered in three aspects. Firstly, there is a decrease in the mechanical characteristics of structural materials at high temperature. Secondly, heating of the structure can lead to the occurrence of temperature stresses in its elements and elements of aircraft systems [4]. Thirdly, the creep phenomenon may occur, that is, continuous plastic deformation of structural elements under constant load [5].

### 3 Approaches to solving the problem of thermal heating of supersonic passenger aircraft of the second

Application in the construction of heat-resistant materials.

Each material, starting from a certain temperature, has a significant decrease in its mechanical characteristics, including \( \sigma_{0.2}; \sigma_v; E \). Figure 4 shows the curves of changes in the tensile strength (\( \sigma_v \)) and elastic modulus E of some materials (1 – duralumin; 2 – titanium alloy; 3 – stainless steel) in temperature endences related to the values \( \sigma_{B_0} \) and \( E_0 \) at a temperature of +20 °C.

![Figure 4: Mechanical properties of materials](image)

**Table 1.** Braking temperature at the leading edge.

<table>
<thead>
<tr>
<th>Number M</th>
<th>1.0</th>
<th>1.3</th>
<th>1.5</th>
<th>1.7</th>
<th>2.0</th>
<th>2.2</th>
<th>2.5</th>
<th>2.7</th>
<th>3.0</th>
</tr>
</thead>
<tbody>
<tr>
<td>( t_0, ^\circ C )</td>
<td>-13.3</td>
<td>16.7</td>
<td>41.0</td>
<td>68.5</td>
<td>116.5</td>
<td>169.0</td>
<td>214.0</td>
<td>259.0</td>
<td>336.0</td>
</tr>
</tbody>
</table>

**Table 2.** Surface temperature at a distance of 1 m from the leading edge.

<table>
<thead>
<tr>
<th>Number M</th>
<th>1.0</th>
<th>1.3</th>
<th>1.5</th>
<th>1.7</th>
<th>2.0</th>
<th>2.2</th>
<th>2.5</th>
<th>2.7</th>
<th>3.0</th>
</tr>
</thead>
<tbody>
<tr>
<td>( t_{wL}, ^\circ C )</td>
<td>-17.5</td>
<td>8.0</td>
<td>30.0</td>
<td>52.0</td>
<td>94.0</td>
<td>124.0</td>
<td>174.0</td>
<td>210.0</td>
<td>271.0</td>
</tr>
</tbody>
</table>
One of the ways to solve the problem of heating the structure is the use of heat-resistant materials. So, in a number of designs with a moderate degree of heating, instead of the D16 material, D19 is used in the airframe design, which has a higher strength in the temperature range of 120 ... 190 °C. At temperatures above 200 °C, aluminum alloys are no longer used, and "hot" structures are beginning to be made of titanium alloys or steels. For clarity, the temperature limits of the use of materials are shown in Fig. 5. It shows that the temperature effect is significant, therefore, these alloys are used, as a rule, only up to the lower limits of the limits.

![Fig. 5. Temperature limits of materials.](image)

So, when modeling a flight at an altitude of more than 21,000 m and at a speed of M = 3.5, the temperature of the inlet edges of the air intake, nose fairing and front edges of the wing is 350 – 37°C, and the rest of the skin is 200 – 250°C.

Therefore, titanium and steel alloys are used in the design of such a high-speed aircraft, which are able to withstand intense heating of the structure [6].

![Fig. 6. Temperature summary of modern designed aircraft.](image)

By replacing the most common aluminum alloys in aviation, such as D16AT and B95p with titanium, we will achieve a higher tensile strength and heat resistance with sufficiently good plasticity.

Do not forget about steel alloys, which perfectly cope with the high temperature generated by the engines. In compartments of engine nacelles and highly loaded aircraft components. Such alloys retain good mechanical properties at high temperatures typical for flights with a large number of M.


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Table 3. Properties of materials used in the construction of aircraft.

<table>
<thead>
<tr>
<th>Alloy</th>
<th>σ₀s</th>
<th>σ₀.2</th>
<th>E, GPa</th>
<th>δ, %</th>
<th>ρ, kg/m³</th>
<th>t, °C</th>
</tr>
</thead>
<tbody>
<tr>
<td>D16AT</td>
<td>400-430</td>
<td>270-300</td>
<td>72</td>
<td>8-10</td>
<td>2.77×10³</td>
<td>to 190</td>
</tr>
<tr>
<td>B95p</td>
<td>500-560</td>
<td>430-480</td>
<td>72</td>
<td>7-8</td>
<td>2.78×10³</td>
<td>to 200</td>
</tr>
<tr>
<td>BT20</td>
<td>930-1200</td>
<td>850-1050</td>
<td>112</td>
<td>8-10</td>
<td>4.45×10³</td>
<td>to 400</td>
</tr>
<tr>
<td>BT23</td>
<td>1080</td>
<td>1010</td>
<td>105</td>
<td>6-9</td>
<td>4.57×10³</td>
<td>to 425</td>
</tr>
<tr>
<td>BT18ch</td>
<td>1000-1200</td>
<td>950-1150</td>
<td>120</td>
<td>10</td>
<td>4.54×10³</td>
<td>to 600</td>
</tr>
<tr>
<td>BKC-10</td>
<td>1300</td>
<td>–</td>
<td>–</td>
<td>–</td>
<td>–</td>
<td>to 450</td>
</tr>
<tr>
<td>BKC-6</td>
<td>1550</td>
<td>–</td>
<td>–</td>
<td>–</td>
<td>–</td>
<td>to 500</td>
</tr>
</tbody>
</table>

Use of fuel heat capacity

No less common in aviation is the method using the cooling resource of fuel. I.e., the organization of active cooling of the most heated sections of the airframe and power plant by means of heat removal by fuel Fig. 7.

Fig. 7. Wing compartment.

Under certain flight conditions, stationary thermal fields with large temperature gradients are established. Such examples can serve as parts of the structure, from different parts of which heat is removed with different intensities. Figure 7 shows the wing fuel compartment partially filled with fuel. At the fuel level, there is a sharp jump in the temperature of the tank wall, as a result of which the most heated part of the wall, which is not washed by fuel, experiences compression deformation. Local temperature gradients also occur in the panel skin at its junctions with stringers, where there is an increased outflow of heat from the skin.

In other words, the fuel "takes" part of the temperature from the wing surface, which helps to reduce the temperature stresses on the aircraft structure. Thus, it prevents such frequent and abrupt occurrence of "heating-cooling" cycles and does not lead to fatigue destruction of aggregates and assemblies.

4 Improvement of the aerodynamic layout

Replacing the construction materials with more modern ones, changing the location of tanks and fuel grades, we must remember that most of the heating occurs at the leading edges and it is necessary to deal with this first of all there.

There are several options for reducing heat flows due to modification of the basic aerodynamic layout: increasing the sweep angles of the bearing surfaces and the radius of
the wing tip and the front edges of the inflows, reducing the area of the washed surface due to "flattening" the initial aerodynamic contours [7].

The type of experimental setup and dimensionless dependences of the heat flow for the front points of the console are shown in Fig. 8 and 9. Wind tunnel purges were performed at the sweep angles of the console 45°, 55° and 65°. It can be seen that an increase in the sweep of the wing by 20° reduces the heat flow along the front points by about 3-3.5 times, and on the surface of the plate – by 2 times.

Do not forget about the important fact that with an increase in the radius of blunting of the leading edge, there is a significant decrease in the heat flow, which strongly affects the heating of the structure.

There is also a significant increase in temperature on certain parts of the surface of the aircraft. An important task is to seal the gaps, for example, between the wing and the slat, between the keel and the rudder, as well as the flaps of the hatches and exit openings of various onboard aircraft systems. Vortices arising in these zones can cause intense local heating of the structure.

Fig. 8. The appearance of the experimental model for studying the flow around the console.

Fig. 9. The dependence of the heat flow on the longitudinal distance L along the leading edge and the sweep angle.

Despite the improvement in the conditions of thermal loading, the change in the parameters of the basic aerodynamic layout significantly affects the flight characteristics implemented in the entire range of heights and flight speeds of the aircraft. Therefore, this method cannot be considered the main one when creating such a class of aircraft.
5 Replacement of more critical parts of the skin with three-layer panels

From the point of view of the structural design of the cladding of the most critical areas, preference is given to three-layer panels with filler of various configurations. Not so long ago, Chinese scientists showed the possibility of using multilayer panels with a three-dimensional bulk filler (panel materials are titanium, zirconium alloys and carbon-silicon class KM) in the design of a high-speed aircraft airframe. One of the variants of such a panel with additional felt insulation is shown in Fig. 10. For use in the design of the front parts of the fuselage and the edges of the bearing surfaces, solid-formed or composite parts made of heat-resistant composite materials are considered.

Fig. 10. Panel with felt insulation.

However, despite this, this variant of the perception of thermal loading cannot be considered a completely optimal solution, especially for reusable high-speed aircraft that do not go into outer space. This is primarily due to the need to minimize the weight of the airframe [8-13]. Nevertheless, internal thermal insulation remains the only option to protect electronic equipment and other important aircraft components from overheating at high supersonic flight speeds.


Perhaps this is the most modern and promising method of combating thermal heating. The essence of this method is the replacement of well-known construction materials with more advanced and advanced ones. Back in the 70s of the last centuries, the first experiments were carried out using beryllium in the design of aircraft, as well as composite materials based on boron or carbon fibers. These materials still have a high cost, but at the same time they are characterized by low density, crack resistance, high strength and rigidity, as well as significant heat resistance.

In order to manufacture the most heat-loaded elements of the hull, parts of the hot path of engines and structural elements of radio engineering with operating temperatures of more than 1500 °C for new generation aircraft, it is necessary to use structural ceramic and glass-ceramic composite materials with low weight, high values of strength, hardness, crack resistance, corrosion and erosion resistance in combination with a long-life cycle under conditions of high-temperature oxidation.

It follows from the data in Table 4 that the list of materials with a melting point of 3000 °C has only a limited number of non-oxide compounds. Moreover, a high melting point is only one of the requirements for composites based on ultra-high temperature ceramics.
Table 4. Composite materials with high melting points will be useful.

<table>
<thead>
<tr>
<th>Material</th>
<th>Crystal structure</th>
<th>Density, g/cm³</th>
<th>Melting point, °C</th>
</tr>
</thead>
<tbody>
<tr>
<td>HfB₂</td>
<td>Hexagonal</td>
<td>11.2</td>
<td>3380</td>
</tr>
<tr>
<td>HfC</td>
<td>Face - centered cubic</td>
<td>12.76</td>
<td>3900</td>
</tr>
<tr>
<td>HfN</td>
<td>Face - centered cubic</td>
<td>13.9</td>
<td>3385</td>
</tr>
<tr>
<td>ZrB₂</td>
<td>Hexagonal</td>
<td>6.1</td>
<td>3245</td>
</tr>
<tr>
<td>ZrC</td>
<td>Face - centered cubic</td>
<td>6.56</td>
<td>3400</td>
</tr>
<tr>
<td>ZrN</td>
<td>Face - centered cubic</td>
<td>7.29</td>
<td>2950</td>
</tr>
<tr>
<td>TiB₂</td>
<td>Hexagonal</td>
<td>4.52</td>
<td>3225</td>
</tr>
<tr>
<td>TiC</td>
<td>Cubic</td>
<td>4.94</td>
<td>3100</td>
</tr>
<tr>
<td>TiN</td>
<td>Face - centered cubic</td>
<td>5.39</td>
<td>2950</td>
</tr>
<tr>
<td>TaB₂</td>
<td>Hexagonal</td>
<td>12.54</td>
<td>3040</td>
</tr>
<tr>
<td>TaC</td>
<td>Cubic</td>
<td>14.50</td>
<td>3800</td>
</tr>
<tr>
<td>TaN</td>
<td>Cubic</td>
<td>14.30</td>
<td>2700</td>
</tr>
<tr>
<td>SiC</td>
<td>Polymorphie</td>
<td>3.21</td>
<td>Dissociation at 2545 °C</td>
</tr>
</tbody>
</table>

To create promising structures capable of perceiving large temperature and strength loads, it is necessary to develop polymer materials that will use ultra-high-temperature ceramics based on diborides, carbides, nitrides as matrix components. To meet all the criteria, they must include elements such as Hf, Zr, Ti, Ta, as well as silicon carbide, which, according to Table 4, have the highest melting point values (Tm).

6 Conclusion

Each of the existing and promising methods of thermal protection considered in this paper, separately, will not allow to fully develop protection against heating for supersonic passenger aircraft of the second generation.

The implementation of each of these methods requires evaluation, mass and financial costs for its implementation, taking into account technical and technological excellence. Thus, the options for reducing the impact of this phenomenon on the work of structural elements are diverse. From the replacement of structural materials of panels and cladding with titanium or steel alloys, to design solutions for the use of fuel lines.

At the design stage, it is necessary to consider the formation of an integrated combination of methods that allows achieving a more rational configuration of the design of aircraft units for the projected supersonic aircraft in terms of efficiency and weight. The development of polymer composite materials used in the construction of airframe units of a high-speed aircraft should be carried out taking into account the thermal heating of the structure.

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International Conference Management of large-scale system development (MLSD), Moscow, Russian Federation, pp. 1-5 (2022). doi: 10.1109/MLSD55143.2022.9934185


