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Thermal Management for a Mach 5 Cruise Aircraft Using Endothermic Fuel

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A thermal management concept for a Mach 5 cruise aircraft using noncryogenic fuel is presented. Catalytic endothermic reaction of petroleum fuel is used as the heat sink for engine cooling. A secondary closed-loop coolant circuit removes heat from the engine and transfers this heat to the catalytic reactor. Insulation on the engine flow path surfaces reduces the cooling requirements. A high temperature insulation system, which is capable of a surface temperature of 4000°F (2200°C), is used for the combustor and nozzle. A complete closed-loop cooling system design is shown in detail. Main features of this system include a fuel preheater, a catalytic fuel reactor, and engine wall cooling panels. A silicone-based liquid polymer, designed for extended use at 750°F (400°C), is used as the coolant. The preheater and reactor design are based on the results of recent experimental work. The cooling panels are designed using a thermal fluid analysis computer program that was originally developed for the National Aero-Space Plane (NASP). Major components are analyzed structurally as well as thermally and weights are presented.

Introduction

IN 1989, in a cooperative effort by NASA and the U.S. Navy, a conceptual design study was made of a Mach 5 cruise aircraft.[1] Design specifications for this aircraft included a range in excess of 1200 n.mi. (1900 km) and a cruise altitude above 90,000 ft (27,000 m). In was also required that the aircraft be capable of carrier operations. Those specifications led to some distinct design restrictions in terms of aircraft size, weight, takeoff and landing characteristics, safety, and logistics. Top and front views of the aircraft resulting from this study are shown in Fig. 1. The requirement for hypersonic flight, coupled with the safety and logistics restrictions, created a unique challenge in the areas of fuels and thermal management. This paper presents a detailed thermal management concept for such an aircraft.

Propulsion

The propulsion system concept incorporated in this aircraft was derived from a Mach 5 penetrator study done in the mid-1980s as a joint effort between NASA, Lockheed California, and Pratt & Whitney.[2-4] That system consisted of an over/under turbo/ramjet design that utilized a F-100 class turbofan, integrated with a two-dimensional ramjet. That system was designed for liquid methane fuel. The system designed for the current concept is similar to the Lockheed/P&W concept in that both use a two-dimensional mixed compression inlet design with a variable geometry ramp system scheduled to position the shocks for efficient operation. Because of the extreme temperature of the exhaust gases, the current concept uses a thermal choke in the ramjet nozzle. This airplane uses two turbo/ramjet units.

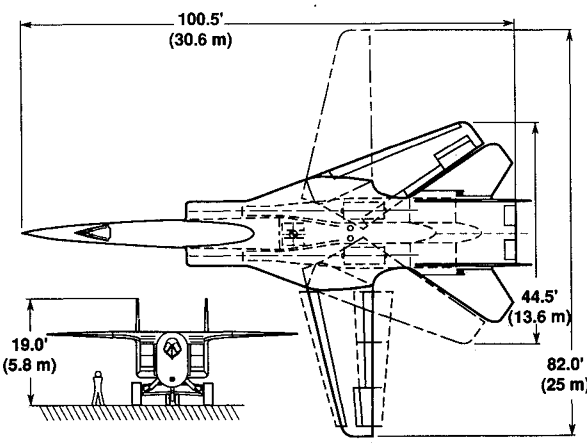

Fig. 1 Mach 5, carrier-based hypersonic aircraft.

The modes of operation for the current concept are shown in Fig. 2. Subsonic propulsion is provided by the turbojet with afterburner, whereas the ramjet is cold flowing. The ramjet is ignited transonically, and the system operates in dual mode until the turbojet begins to spool down near Mach 2.5. The ramjet has been shown to be of assistance transonically, primarily for drag reduction, even though its low speed efficiency is rather poor. As the aircraft accelerates from the speed of sound to Mach 3, the ramjet becomes more efficient and provides increasing thrust. At the same time, aerodynamic heating becomes an increasing problem for the turbojet. The turbojet remains in full afterburner until it is finally shut down at Mach 3. The ramjet then provides all of the thrust up to Mach 5.

Fuels

In the high Mach number region of the flight envelope ($M >$ 4), aerodynamic heating is too great for conventional structural materials to survive without active cooling. The fuel not only must have a good heat-of-combustion but also must pro-

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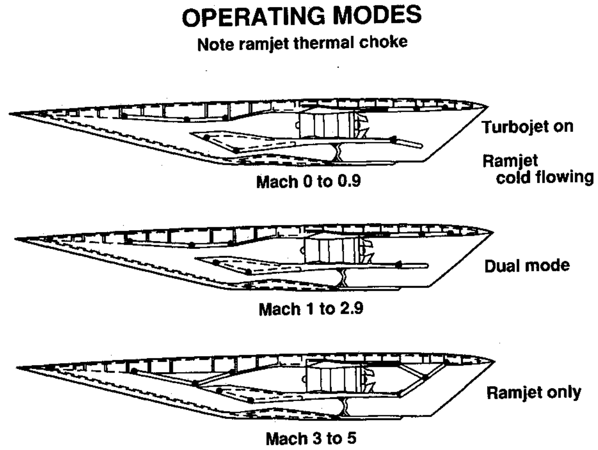

Fig. 2 Operating modes for turbo/ramjet engine.

vide the necessary heat sink for the cooling system. Three different fuels are compared in Table 1. Both liquid methane (CH_4) and liquid hydrogen (H_2) are cryogenic. The heat sink and heat-of-combustion figures for the fuels are based on heating to a given temperature before injection, labeled "when heated to" in Table 1. The cryogenic fuels would be heated directly in a cooling system, whereas the noncryogenic fuel would be used with a secondary coolant loop. The given cost figures do not include the cost of energy necessary to maintain the fuel at cryogenic temperatures. Because of the extreme cold involved, cryogenic fuels pose many design, logistical, and safety problems. In the cooling system design, extremely low temperatures cause high thermal gradients that lead to thermal stress problems. Materials tend to lose toughness at low temperatures and use of H_2 can cause problems with hydrogen embrittlement of the coolant passages. The insulation required for cryogenic fuel tanks is extra weight for the aircraft. In this particular design, U.S. Navy specifications do not allow the use of cryogenic fuels in aircraft carrier operations.

An alternative to cryogenics is provided by the chemical heat sink available in an endothermic petroleum fuel such as methylcyclohexane (MCH) ($C_6H_1CH_3$). Methylcyclohexane is a cycloalkane formed by exothermic catalytic hydrogenation of high purity toluene ($C_6H_5CH_3$). MCH is a clear liquid with approximately the same density and heat-of-combustion as JP-7. The viscosity is less than that of JP-4. On a volume basis, it has more than four times the heat-of-combustion of liquid H_2. The flashpoint for MCH (25°F, −4°C) is higher than that for JP-4, making MCH safer to handle.[5]

The heat sink is provided by a two-stage preheater-reactor system. A fuel pump pressurizes the fuel beyond the critical point to avoid boiling. The preheater heats the fuel to the proper reaction temperature while removing heat from a secondary coolant. To maximize efficiency, the preheater is designed as a counterflow heat exchanger. Experiments have shown that the preheater provides about one-third of the total heat sink of the system.[6] After preheating, the fuel passes through the catalytic heat exchanger/reactor (HE/R). The endothermic catalytic reaction of the MCH is basically the reverse of the process by which it was formed from toluene and hydrogen. At 100% efficiency the reaction will absorb 1000 BTU/lbm of MCH at 925°F (496°C) and 500 PSIA (3.45 MPa). In this design, the endothermic reaction efficiency is estimated to be 58%, with the reaction absorbing 580 BTU/lbm. Catalytic efficiency decreases with temperature and is limited by the operating temperature of the silicone-based coolant (Fig. 3). Preheating of the fuel before reaction absorbs another 313 BTU/lbm from the coolant.

The fuel is chilled to achieve maximum range at Mach 5 speeds and to keep the fuel below the flashpoint while on deck. Before takeoff, the fuel is chilled to 20°F (−6.7°C). By the time the aircraft has accelerated to Mach 5, the fuel will have been heated to 70°F (21°C) due to aerothermal heating. If unchilled fuel is used, the cruise airspeed or range would be reduced. The fuel tank insulation could be increased to limit the fuel tank temperature and preserve the available heat sink; this would cause an increase in aircraft weight and a reduction in fuel tank volume, thus reducing the range. Alternatively, the fuel could be allowed to heat up, which

Table 1 Liquid fuels for hypersonic aircraft

	MCH, indirect cooling	CH4, direct cooling	H2, direct cooling
Cryogenic	no	yes	yes
Heat of Combustion			
BTU/lb	19,804	21,465	52,870
BTU/gal	127,090	83,475	30,884
When heated to, °F	803	1340	1340
Heat sink, BTU/lb			
Chemical (58%)	580	0	0
Physical	313	1427	6057
Total	893	1427	6057
Boiling point, °F	213	−259	−423
Cost, $/gal	0.90	2 to 4	1.30

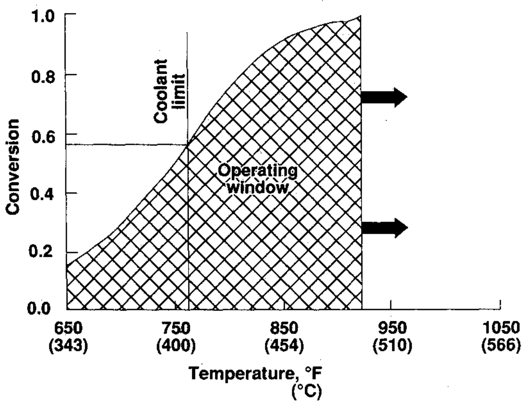

Fig. 3 Catalytic endothermic process operating window.

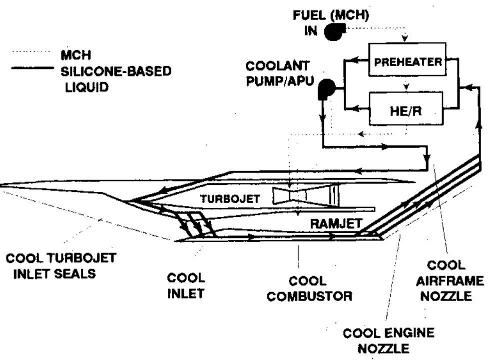

Fig. 4 Coolant routing diagram.

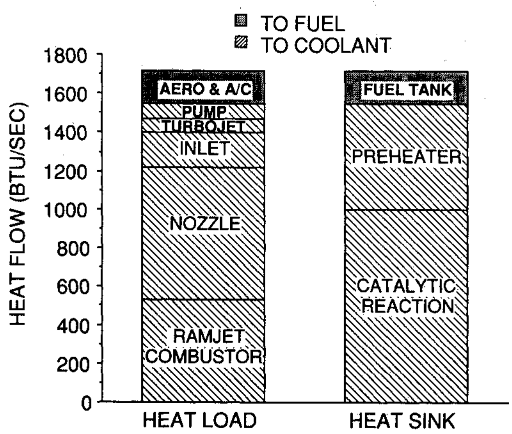

Fig. 5 Heat load and available heat sink at Mach 5 for one engine.

Table 2 Coolants for closed loop

	Syltherm 800[R]	He @1050 psi	NaK, 78%K
State	liquid	gas	liquid
Highly flammable	no	no	yes
Toxic	no	no	yes
Max. use temp., °F	925	none	1400
Density, lb/ft³	30.5	0.21	53.1
Density × Spec. heat (BTU/ft³ °F)	17.0	0.26	11.9
Thermal Conductivity (BTU/h ft °F)	0.034	0.16	13.8
Freezing point, °F	−40	−459	14

Table 3 Effect of coolant on system design for one engine

	Syltherm 800[R]	He	NaK, 78%K
Max. pressure, psia	424	1050	35
Pump power (HP) 80% efficient	143	628	1.4
Mass flow, lb/s	77	4	11.2
Coolant weight, lb	1415	2	1700
Expansion tank	yes	no	yes

would reduce the available heat sink, thus limiting the cruise airspeed. An increase in fuel temperature at Mach 5 to 110°F (43°C) would decrease the available heat sink by about 25 BTU/lb (2% of the total). Launch delays for an aircraft with chilled fuel are not a problem. Because the thermal mass of the fuel is so great and the tank insulation is very effective, the fuel bulk temperature in the tank only increases at a rate of 1.6°F/h (0.9°C/h) under the most stringent ambient weather conditions. This means fully fueled aircraft could be left on deck for long periods before having the fuel rechilled. The air conditioning system for the pilot and avionics uses the fuel directly as a heat sink. Refrigerant coils in the fuel tank transfer A/C heat to the fuel in a conventional vapor-compression refrigeration cycle.

Only the surfaces in and near the ramjet and nozzle are actively cooled. A basic coolant routing diagram is provided in Fig. 4. Coming out of the coolant pump near the rear of the airplane, the coolant is used to cool the airframe nozzle area and is then piped forward to the turbojet inlet close-off doors, where it is used to cool the seals. The coolant is then pumped down the ramjet inlet sidewalls and then back down through cooling panels that surround the ramjet, forming a cooling jacket. The coolant is then returned to the heat sink.

Heat load and heat sink are shown in Fig. 5. The fuel tankage is allowed to absorb heat both from the airframe hot structure and from the air conditioning system. The ramjet area heat loads are as follows: engine nozzle (46%), ramjet combustor (35%), ramjet inlet (13%), airframe nozzle (2.6%), and turbojet inlet close-off door seals (2.6%). Approximately $\frac{1}{3}$ of the heat sink for active cooling comes from the preheater, the rest from the reactor.

Survey of Coolants

The chosen coolant is a commercial heat transfer liquid "Syltherm 800," a product of Dow Corning Corporation of Midland, Michigan.[7] This commercial polydimethylsiloxane fluid has excellent heat transfer characteristics and is odorless, nontoxic, nonfouling, and noncoking. It has a useful life of at least six months at a temperature of 830°F (443°C) in a stainless steel system as used in this design. The fluid was chosen primarily for its low freezing point and high boiling point (−40 and 390°F (−40 and 199°C) at 1 atm, respectively). Syltherm 800 and two other possible coolants are compared in Table 2.

Many other coolants were considered, including both liquids and gases. First considered was a liquid metal alloy of sodium and potassium (NaK). Although NaK has superior thermal characteristics, it was rejected because of flammability. NaK will burn violently when exposed to water and will also ignite when exposed to air at temperatures above 239°F (115°C).[8] A leak of this coolant in flight could be catastrophic. Water was considered because it is an excellent coolant, but to avoid boiling liquid water would require pressurization to 3000 PSIA at 695°F (20.7 MPa at 368°C). A decision not to allow boiling in the cooling panels was made early in the design process. Gaseous water (i.e., steam) was considered but would require a separate boiler in the system and special start-up procedures. Gaseous hydrogen has excellent heat transfer properties but was rejected because of low density and extreme flammability. Gaseous helium has good heat transfer properties yet is inert. A complete system using helium was designed with a maximum temperature of 851°F (455°C) and a maximum pressure of 1050 PSIA (7.3 MPa) at 4 lbm/s (1.8 kg/s) flow. This system required a 628 HP, 800 CFM (inlet) compressor for each engine. The weight and size of these compressors and the associated APU was judged prohibitive. Characteristics of cooling systems for this aircraft using different coolants are shown in Table 3.

Molten salts were considered because they are excellent heat transfer liquids and are among the safest high temperature fluids available.[9] A eutectic mixture of potassium nitrate, sodium nitrate, and sodium nitrite (known as "heat transfer salt" or HTS) is usable up to 840°F (450°C). Molten salts were not chosen because they have the disadvantage of a high melting point; HTS melts at 288°F (142°C). Dilution with water by 15% lowers the melting point to 176°F (80°C), but a separate melting pot and start-up procedure still would be required. The start-up procedure would take about 2 h starting with the airplane at room temperature. The airplane could be kept ready by circulating and heating the salt, but such a system would mean additional weight, volume, and complexity for the aircraft.

Cooling System Design

The cooling system design was driven by Mach 5 cruise conditions, where the heating loads are greatest. Heat loads will be substantially reduced during descent because of the slower airspeed. The fuel schedule for the airplane must ensure that sufficient fuel is in the insulated tank to prevent tank overheating during descent. Transient effects will add to the descent heat load especially if the descent is rapid. A transient analysis of the fuel system would be necessary to quantify this effect; such an analysis is beyond the scope of this paper but will be considered in future work. Engine fuel flow and ramp configuration can be adjusted to avoid engine overheating during descent.

Operating conditions for Mach 5 flight are shown in Fig. 6. The maximum pressure is 424 PSIA (2.93 MPa) at the inlet to the cooling panels, and the maximum temperature is 838°F

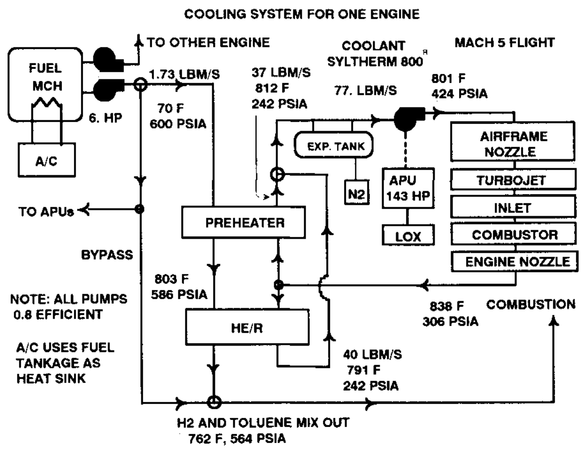

Fig. 6 Cooling system schematic for one engine.

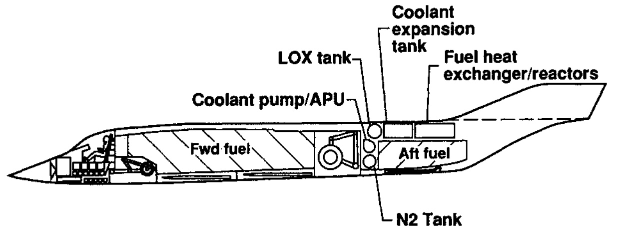

Fig. 7 Carrier-based hypersonic aircraft, inboard profile (to scale).

(448°C) at the outlet of the cooling panels. The cooling panels are arranged in series with the coolant flowing forward through the airframe nozzle, then downward to cool the turbofan inlet close-off door seals and back through the ramjet. The 77 lbm/s (35 kg/s) flow splits after the last cooling panel outlet with 40 lbm/s (18 kg/s) flowing through the heat exchanger/reactor and the rest flowing to the preheater. The flow is remixed at the inlet to the expansion tank. The pump requires 143 HP at 242 PSIA (1.67 MPa) and 80% efficiency. A separate power unit using LOX and MCH drives these pumps. The use of cryogenic liquids in large quantities is not allowed on aircraft carriers; however, small amounts of LOX are used for pilot breathing at high altitudes. It is estimated that the quantity of LOX required for the APU would be comparable to that required for the pilot.

Syltherm 800 has a high coefficient of thermal expansion, making an expansion tank necessary. Between room temperature and 838°F, Syltherm 800 expands 91% in volume. According to the manufacturer, this tank should be designed to be $\frac{1}{4}$ full of coolant when cold and $\frac{3}{4}$ full when hot.[10] The remaining volume is filled with a nitrogen gas blanket. The tank is located at the lowest pressure point in the system, and the constant flow of coolant liquid through the tank ensures that vapors will be isolated in the tank. The physical arrangement of the components is shown in the scale inboard profile of Fig. 7.

A unique feature of this cooling system is that it is self-compensating, i.e., an increase in heat load will lead to an increase in the temperature of the coolant entering the heat rejection devices. The increase in temperature will lead to an increase in the efficiency of the heat exchanger/reactor, increasing the temperature drop in the coolant. Similarly the preheater will also transfer more heat to the fuel. The effect of the increase in heat load on the temperature of the coolant will have been diminished and a new equilibrium will be established. In this off-design case, the coolant and catalyst may be degraded, but the system will operate until the fuel temperature reaches 925°F (496°C); this is the limiting condition for both the catalyst and coolant. Syltherm can operate for five days at a temperature of 932°F (500°C), but the pressure would have to be increased to avoid boiling. Note that the heat capacity of the coolant is so great that a 15% increase in heat load will only cause an increase of 6°F (3.3°C) in the coolant.

Heat Exchanger/Reactor

The heat exchanger/reactor (HE/R) (Fig. 8) is designed to remove heat from the coolant by catalytic endothermic reaction of the fuel. It consists of a five pass shell-and-tube heat exchanger and is made of 316 stainless steel. It is designed to operate at a coolant pressure of 350 PSIA at 1000°F (2.4 MPa, 537°C) and a fuel pressure of 600 PSIA at 850°F (4.15 MPa, 454°C). The platinum-on-alumina catalyst is packed to 70 mesh size in 3709 $\frac{1}{4}$ in. ID tubes. The fuel, in the form of a supercritical fluid, flows through the tubes at a design liquid hourly space velocity (LHSV) of 100. The LHSV is the volumetric fuel flow rate per hour divided by the volume of the catalyst in the reactor. The shell pressure vessel is designed to a factor of safety of 3 on yield strength; the vessel walls are $\frac{1}{16}$ in. thick. The coolant inlet pipe has an ID of 2 in. for an inlet velocity

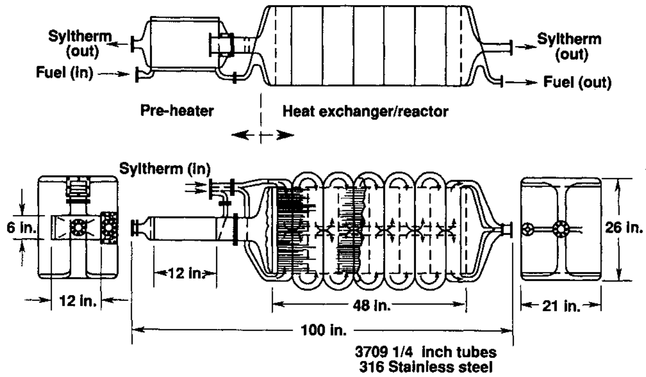

Fig. 8 Preheater and heat exchanger/reactor (to scale).

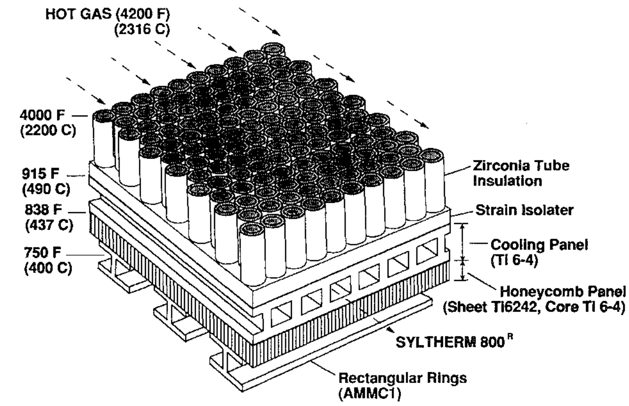

Fig. 9 Ramjet combustor and engine nozzle wall construction (not to scale).

of 60 ft/s. The HE/R is rectangular in cross section and approximately 100 in. long. It is designed to handle a maximum fuel flow of 1.8 lbm/s (0.82 kg/s). The device weighs approximately 964 lb, empty of coolant and fuel.

Insulation

Properties of insulation considered for use in this aircraft are shown in Table 4. The insulation with the highest temperature capability consists of $\frac{1}{8}$ in. O.D. zirconium oxide tubes filled with a foam of the same material and arranged vertically in bundles. The zirconium oxide has 8% yttria stabilizer added to it, increasing the amount of cubic crystalline structure for increased strength. No facesheet is used over the tube bundles. A solid slab of zirconia, although it could be fabricated, would be susceptible to thermal stress cracking. The tube arrangement allows for control of the insulation thickness and accommodates thermal expansion. The top surface of the tube bundles will not be as smooth as a polished sheet of metal but it is expected to be smooth enough that the friction drag and heat transfer rates will not be greatly increased. There is no known substitute for this insulation that is designed to withstand exposure to gases at temperatures above 4000°F (2200°C).[11] Except for its temperature capability, the zirconia is not very good insulation. A lower temperature system, such as carbon-carbon sheets backed by saffil-alumina fiber, might be used if the cooling capacity of the fuel were greater.

Each of the two metallic TPS systems, multiwall and honeycomb, is stiff enough to support its own weight. The multiwall is made of cobalt L605 superalloy. Honeycomb is usually used as load bearing structure and is so used in this design. In addition to its structural usefulness, this honeycomb has good resistance to thermal shock. This honeycomb consists of Ti 6242 facesheets and a Ti 6-4 core, with a core solidity of 1.65% and a core height of 1 in. Titanium alloys and cobalt alloys are particularly good choices for use in a carrier-based

Table 4 Insulation for hypersonic aircraft

	Zirc. oxide tubes	Cobalt L605 multiwall	Titanium honeycomb
Self-supporting	no	yes	yes[a]
↑ Max. use temp., °F	4000	1900	1000
↓ Thermal cond. (BTU/h ft °F)	0.71	0.126	0.706
↓ Density (lb/ft^3)	185	14.9	4.84
↓ Density × cond.	131	1.87	3.42
↑ Therm. diffusivity, ft^2/h	0.032	0.094	0.811

[a]Used as load bearing structure.

Table 5 Cooling System weights and volumes for one engine

Component	Weight, lb	Volume, ft^3
Preheater	33	0.75
HE/R	964	25
Coolant	1415	24.5
Cooling Panels	639	—
Expansion Tank	160	30.4
Insulation	**Weight, lb**	**Thickness, in.**
Inlet[a]	18	0.68
Ramjet Combustor[b]	456	0.67
Ramjet Nozzle[b]	670	0.69
Airframe Nozzle[b]	36	0.68
Total for insulation	1180	

[a]Multiwall.
[b]Zirconium oxide tubes.

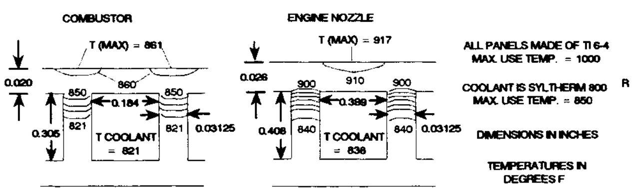

Fig. 10 Cooling panel dimensions and temperatures at Mach 5 cruise conditions (not to scale).

aircraft because of their superior resistance to saltwater corrosion. Saffil-alumina fibrous insulation is a good candidate as fuel tank insulation. It is not used in the actively cooled areas of this design mainly because it is not self-supporting.

Wall Construction

The wall construction for the ramjet inlet and airframe nozzle consists of multiwall Thermal Protection System (TPS), the cooling panels, and a structure of honeycomb supported by rectangular rings. The multiwall is designed to withstand exposure to the inlet air or expanded exhaust gas while the aircraft is cruising at Mach 5. The TPS is 0.68 in. thick in both places and weighs 36 lb on the nozzle; it weighs 18 lb on the inlet. The backside of the TPS will be maintained at 818°F (437°C) by the cooling panel, made of Titanium 6-4. The land thickness is 0.03125 in. whereas all other thicknesses are 0.020 in. The inlet and nozzle panel weigh 148 lb each. The honeycomb is supported by rectangular rings made of the primary structural material and designed to operate at 750°F (400°C). The primary structural material is assumed to be a NASP-derived advanced metal matrix composite (AMMC 1).

The wall construction for the ramjet combustor and engine nozzle consists of zirconia insulation, with the backside temperature maintained by the cooling panel. Behind the cooling panel is a structure of honeycomb supported by rectangular rings. A section of this construction is shown in Fig. 9.

The insulation in the combustor weighs 416 lb and is 0.61 in. thick. The insulation in the nozzle weighs 611 lb and is 0.63 in. thick. The tubes are mounted on a plate that is attached to a strain isolation pad. The cooling panel is made of Ti 6-4. The land thickness is 0.03125 in. in both panels. The combustor wall is 0.020 in. thick, while the nozzle wall is 0.026 in. thick. The combustor cooling panel and the nozzle panel weigh 161 lb each. The honeycomb acts as insulation with a 88°F (49°C) temperature drop through the thickness. The honeycomb is supported by rectangular rings made of the primary structural material and designed to operate at 750°F (400°C).

The cooling system was designed using a thermal/fluid analysis computer program called NASP/SINDA. It was originally developed for the National Aero-Space Plane (NASP) and used hydrogen as the coolant. Special versions of the program were written, first using helium and then using Syltherm 800. Structurally the panels were designed to have a minimum factor of safety of 1.5 on yield against internal pressure. The relatively thick lands are designed to resist buckling by through-the-thickness compressive loads. The results of the thermal analysis of the combustor and engine nozzle panels are shown in Fig. 10. A thermal stress analysis of the panels was not done, but the temperature gradients do not appear to pose a problem. The airframe nozzle and inlet panels were almost isothermal through the thickness of 803 and 810°F (428 and 432°C), respectively.

Conclusions

An operational military Mach 5 aircraft is possible without the use of cryogenic fuel. This aircraft can have a range in excess of 1200 n.mi. (1900 km) and cruise above 90,000 ft (27,000 m). This can be achieved by taking advantage of the chemical heat sink of the endothermic fuel MCH. The commercial heat transfer fluid Syltherm 800 is used as the coolant in a secondary loop. The weights and volumes of various major cooling system components for one engine are presented in Table 5. The heaviest single part of the system is the required 183 gal (693 l) of coolant. All of the primary insulation combined weighs 1180 lb. Because they are made of titanium, both the cooling panels and expansion tank are relatively lightweight. The HE/R, made of stainless steel, is heavier. All combined, the cooling system and engine insulation amounts to about 13% of the aircraft 80,000 lb takeoff gross weight.

The weight of those components is very sensitive to the heating load that is a very strong function of top speed. The zirconia insulation is the only part of the design that is beyond the current state of the art. Research has resulted in the fabrication of several specimens, but more development work is necessary. Catalytic reaction of MCH has been accomplished in the laboratory, but a flight weight system has yet to be built.

Acknowledgments

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