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Aerospace Plane Cooling with H_2, CH_4, He, Ne, N_2, and Ar

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This article presents a comparative study for cooling aerospace planes, using liquid H_2, CH_4, He, Ne, N_2, and Ar. The ascending optimized trajectory to minimize the heat load in the hypersonic part is used to perform the study. The study includes the cooling for the stagnation point, the leading edges for wings and engine, and other parts of the aerospace plane that are close to the leading edges. The laminar case for the stagnation point and both laminar and turbulent cases for the leading-edge heating have been considered. The amount of heat rate (total, radiative, and convective) and the mass of liquid coolant needed for cooling are calculated. A design of minimum inlet–outlet areas for the amount of liquid needed for cooling is made with the consideration of the coolant's physical constraints in liquids and gaseous states. The comparison shows that the hydrogen is a clear winner as a candidate for coolant and it saves mass as compared to all other five coolants. The study shows that there are no fundamental barriers for the cooling system of the vehicle in terms of its coolant mass and area size for coolant passage, especially if H_2 is used.

Nomenclature

A = area, m^2
ARI = coolant inlet area ratio
ARO = coolant outlet area ratio
a = speed of sound, m/s
C_D = drag coefficient
C_L = lift coefficient
C_p = specific heat at constant pressure at about 288.15 K and 1 atm, J/kg K
C_Q = heat constant, its units depend on the value of N' and M' in the heat rate equation $\dot{Q} = C_Q \rho^{N'} V^{M'}$
C_v = specific heat at constant volume at about 288.15 K and 1 atm, J/kg K
D = drag, N
D_i = pipe diameter, m
d = width, m
g = acceleration due to gravity, m/s^2
h = geometric altitude, $r - r_0$, km
h_{ev} = heat of vaporization at boiling point, J/kg
h_t = total enthalpy, J/kg
h_w = enthalpy at wall, J/kg
I_{sp} = specific impulse, s
L = lift, N
l = length, m
M = Mach number
M' = number that represents the exponent of the velocity in the heat rate equation
MR = coolant mass rate ratio or mass ratio
m = total mass of the vehicle including fuel and coolant masses at any time, kg
N' = number that represents the exponent of the density in the heat rate equation
$\dot{Q}$ = heat rate per unit area, W/cm^2
R = radius, m
R' = gas constant at about 288.15 K and 1 atm, J/kg K
Re = Reynolds number
r = vehicle position vector from Earth center, m
T = temperature, K
T_G = coolant gas temperature at outlet of coolant path, a temperature higher than the boiling points temperature at 1 atm, K
T_L = coolant liquid temperature at inlet of coolant path at about melting point and 1 atm, K
T_w = wall temperature of the heating area without active cooling, K
T_{wr} = wall temperature of about maximum radiation = 1500 K, K
t = time, s
V = velocity of the vehicle, m/s
V_{in} = velocity of the coolant at inlet area, m/s
V_{ol} = velocity of the coolant at outlet area, m/s
X = local distance measured along the body surface from the body centerline (for extreme case) = 4, m
X_T = X for turbulent boundary layer (for extreme case) = 4, m
β = reciprocal of the scale height, taken to be constant, $\approx$0.000147, 1/m
γ = flight-path angle of the vehicle, rad
γ' = ratio of specific heat at constant pressure-to-specific heat at constant volume, at about 288–293 K and 1 atm, C_p/C_v
ε = surface emissivity for the material = 0.8
θ = longitudinal position of the vehicle, rad
ν = kinematic viscosity at about boiling point, m^2/s
ρ = atmospheric air density as function of altitude, kg/m^3
ρ_G = coolant gas density, a density higher than the boiling point at 1 atm, kg/m^3
ρ_L = coolant liquid density at about boiling point and 1 atm, kg/m^3
σ = Stefan–Boltzmann constant = 5.67×10^{-8}, W/m^2 K^4
ϕ = local body angle with respect to freestream = 45, deg

Subscripts

c = convected
cl = coolant
E = engine
Ele = engine leading edge
fl = fuel
G = gas at outlet
h = heat

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i = prefix for three cases: stagnation point, leading edges, and other parts
L = liquid at inlet
le = leading edge
max, rad = maximum radiation
min, il = minimum inlet
min, ol = minimum outlet
ot = other parts
radn = radiation
s = standard sea level
sp = stagnation point
T = turbulent
t = total (radiative plus convected)
tl = total (stagnation point, leading edges, and other parts)
W = wing
w = wall
wr = maximum wall radiation
0 = initial at Earth surface
1 = base coolant or base temperature difference
2 = coolant or temperature difference other than base coolant or base temperature difference
∞ = freestream

Introduction

TODAY, attention is being focused towards hypersonic passenger aerospace planes in order to travel in space. However, such a vehicle during its flight through the atmosphere will encounter high aerodynamic heating rates. Therefore, it will require some type of cooling for the stagnation point, leading edges, and some other parts of the wings–engine.

Tauber and Menees[1] have shown that the heat load of the stagnation point for the aerospace plane during ascent is about 10 times greater than the heating load of the Space Shuttle re-entry. The corresponding temperatures for such a high heating load vary from 3200 to 3800 K, which are far beyond the radiative cooling capabilities of existing, nonablating, heat-shield materials. They have also indicated that the leading-edge point experienced high heating load with wall temperatures of 2250–2700 K. This shows that both the stagnation point and the wing leading edge may require some cooling. However, the areas of the vehicle that require active cooling may be relatively small and the radiative cooling should be effective over large areas of the aerospace plane.

Since severe heating occurs during the ascent in the hypersonic flight, it makes designing a cooling system a major challenge. Krause et al.[2,3] studied the active thermal control for hypersonic vehicles. They presented a design and the evaluation of a thermal control system that provided cooling by controlling the fuel flow through a network of heat exchangers. The study shows that efficient use of fuel cooling throughout the vehicles operating envelope requires modulating the flow splits within the network as the flight conditions change. Reich et al.[4] studied a thermal protection system (TPS) for hypersonic vehicles, which was required to limit the heat transfer into the central fuselage of liquid fuel. The study demonstrates that microporous and multiwall insulations are efficient, therefore, lightweight and reliable TPSs for future hypersonic vehicles can be designed. Rainey and Veziroglu[5] studied and compared the two cryogenic fuel candidates for hypersonic vehicle: 1) liquid hydrogen and 2) liquid methane. The results show that the hydrogen-fueled hypersonic vehicle is superior to the methane-fueled one for a long range of flight, also, the study shows that hydrogen is a safer fuel than methane. Al-Garni et al.[6] performed a computation that compares the amount of coolant mass needed with respect to the total mass and the fuel mass through the envelope of the ascending trajectory in the hypersonic region. The study considered the amount of coolant using liquid hydrogen and liquid methane, which is required for cooling the stagnation point region, leading edges of the wing–engine, and other parts of a larger area around the leading edges for an extreme case study.

This study is an extension to the study by Al-Garni et al.,[6] where analytical equations to compare different coolants at different temperatures have been deduced and the helium, neon, nitrogen, and argon as additional coolants have been included. This study considered an extreme case, which done by the addition of the extra parts beside the leading edges of area, which to be cooled, the use of coolant fuel mass in addition to fuel mass, the use of extreme values for some of the variables [e.g., X is taken to be small (4 m) and ϕ is large (45 deg), see Fig. 1], which will increase both the heat and the coolant required. The study includes laminar flow over the stagnation point and laminar and turbulent flow over the leading edges and other parts, respectively. The study also gives a comparative study of six candidates for cooling, namely, liquid hydrogen, methane, helium, neon, nitrogen, and argon. The hydrogen results of Al-Garni et al.[6] will be used as the basis for this comparative study. A model that consists of ascending trajectory equations of motion is used in this study, which is similar to the one used by Vinh.[7] The trajectory used for this study is the same as the one presented by Al-Garni[8] and Barlow and Al-Garni,[9] which minimizes the heat load on the aerospace plane. The mass of a coolant will be added to the fuel mass, assuming the coolant mass will not be used as fuel in order to consider the worse case or any inert liquid coolant. However, if the coolant is used as fuel, then the total mass rate equation will be modified and the total mass rate will be reduced, which can be deduced from the worst case. Apparently, no unclassified studies have been performed in this specific area with the previously stated conditions of flight using these coolants. Therefore, contributions in this area are very much needed.

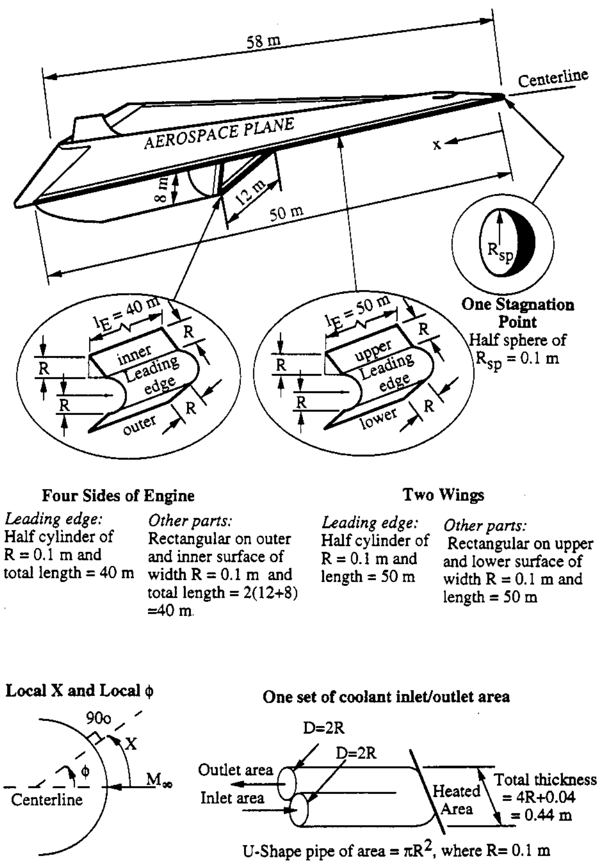

Fig. 1 Aerospace plane and its cooling system.

Model Equations

The vehicle is modeled as a point variable mass with parabolic drag polar and variable lift and thrust as described in the previous study.[8] The Earth is assumed spherical and nonrotating with an exponential atmosphere. The trajectory is taken in the equatorial plane. With these assumptions the equations that describe the ascending hypersonic trajectory have the following nonlinear equations of motion:

$$\frac{\mathrm{d}r}{\mathrm{d}t} = V \sin \gamma \tag{1}$$

$$\frac{\mathrm{d}\rho}{\mathrm{d}t} = -\rho\beta V \sin \gamma \tag{2}$$

$$\frac{\mathrm{d}V}{\mathrm{d}t} = \frac{(T_t - D_t)}{m} - g \sin \gamma \tag{3}$$

$$V\frac{\mathrm{d}\gamma}{\mathrm{d}t} = \frac{L_t}{m} - g \cos \gamma + \left(\frac{V^2}{r}\right) \cos \gamma \tag{4}$$

$$\frac{\mathrm{d}m}{\mathrm{d}t} = -\frac{T_t}{(I_{\mathrm{sp}}g_0)} + (\dot{m}_{\mathrm{cl}})_i \tag{5}$$

Equation (5) considers a worst case where the coolant is not fuel or may be an inert liquid/gas as helium:

$$\frac{\mathrm{d}Q_t}{\mathrm{d}t} = C_Q\rho^{N'}V^{M'} \tag{6}$$

where C_Q can be expressed in terms of R_{sp}, h_w, h_0, ϕ, X, and T_{wr}; and found with N' and M' in Refs. 1, 6, and 10.

Taking $\phi = 45$ deg, $X = X_T = 4$ m from the centerline $(h_w/h_0) = 0.1$, and $T_{\mathrm{wr}} = 1500$ K, where the values of ϕ and X give high values of heat generation in order to consider a worse case. Note that we add Eqs. (1–6) to the mass rate of coolant [Eq. (12)] in order to complete the system to be integrated, also note that Eq. (6) are empirical.

The wall temperature

$$T_w = (\dot{Q}_t/\varepsilon\sigma)^{0.25} \tag{7}$$

where $\varepsilon = 0.8$ and $\sigma = 5.67 \times 10^{-8}$ W/m^2 K^4. Considering that the wall temperature is limited to a maximum of 1500 K by active cooling using H_2, CH_4, He, Ne, N_2, or Ar; and Eq. (7) is used only when $T_w < 1500$ K, then if $T_w \leq 1500$ K, all of the heat will be radiated, i.e., $\dot{Q}_{\mathrm{rad}} = \dot{Q}_t$; if $T_w > 1500$ K, then

$$(\dot{Q}_c)_i = (\dot{Q}_t)_i - \dot{Q}_{\mathrm{max,rad}} \tag{8}$$

where $\dot{Q}_{\mathrm{max,rad}} = 23$ W/cm^2 is the maximum radiation at $T_w = 1500$ K and i represents three cases: 1) the stagnation point, 2) the leading edges, and 3) other parts. Note that $\dot{Q}_c$ is the same for leading edges and other parts.

In the heated area $(A_h)_i$, where i represents the three cases as follows:

1) For the stagnation point, the heated area is taken as half the area of the sphere with radius R_{sp}. This gives

$$(A_h)_{\mathrm{sp}} = 2\pi R_{\mathrm{sp}}^2 \tag{9}$$

where R_{sp} (=0.1 m) is the radius of the sphere.

2) For the leading edge, the heated area is taken as the surface area of a half-cylinder with radius R_{sp} along the leading edge of the two wings $2l_{\mathrm{Wle}}$ and four engine sides l_{Ele}. This gives

$$(A_h)_{\mathrm{le}} = \pi R_{\mathrm{sp}}(2l_{\mathrm{Wle}} + l_{\mathrm{Ele}}) \tag{10}$$

where $l_{\mathrm{Wle}} = 50$ m for each wing and $l_{\mathrm{Ele}} = 40$ m for four sides of the edges of the engine.

3) The heated area for other parts is considered to consist of rectangular areas that are next to the leading edges of the two wings and four sides of the engines. The two sides (i.e., the top and bottom for wings, and the outer and inner for engine) of the rectangular area have widths $d = R_{\mathrm{sp}} = 0.1$ m and the total length of leading edges is $(2l_{\mathrm{Wle}} + l_{\mathrm{Ele}})$. This gives

$$(A_h)_{\mathrm{ot}} = 2d(2l_{\mathrm{Wle}} + l_{\mathrm{Ele}}) \tag{11}$$

The coolant mass rate to cool the heated area is given by

$$(\dot{m}_{\mathrm{cl}})_i = \frac{(\dot{Q}_c)_i(A_h)_i}{h_{ev} + C_p(\Delta T)_i} \tag{12}$$

where $(\Delta T) = (T_G) - (T_L)$ will be chosen; noting that the units of $\dot{Q}_c$ by A_h should be in watts in Eq. (12), and i represents three cases: 1) stagnation point, 2) leading edges, and 3) other parts. Equation (12) will be substituted into Eq. (5) in the first six equations of the system, which will be integrated numerically.

To design the inlet/outlet areas for the coolant one should, at least, get the minimum area for both by the following:

$$(A_{\mathrm{min,il}})_i = \pi(R_{\mathrm{min}})_i^2 = \pi\left(\frac{D_{\mathrm{min}}}{2}\right)_i^2 = \frac{1}{\pi}\left[\frac{2(\dot{m}_{\mathrm{cl}})_i}{\rho_L\nu Re_{\mathrm{max}}}\right]^2 \tag{13}$$

where, for the design of the inlet area, we have used the maximum permissible Reynolds number in a pipe [$Re_{\mathrm{max}} = 3.2 \times 10^6$ (see Ref. 11)] in a liquid state and noting that $Re_{\mathrm{max}} = V_{\mathrm{in}}(D_i)/\nu$ and $V_{\mathrm{in}} = (\dot{m}_{\mathrm{co}})_i/\rho_L\pi r_i^2$. For designing the outlet area, the flow Mach number is taken as 60% of the speed of sound for the coolant in the gaseous state that passes through any shape of the area. This provides a flow Mach number less than the critical Mach number and avoids the shock occurring in the flow. Thus,

$$(A_{\mathrm{min,ol}})_i = \frac{(\dot{m}_{\mathrm{cl}})_i}{\rho_G V_{\mathrm{ol}}} = \frac{(\dot{m}_{\mathrm{cl}})_i}{\rho_G 0.6(a_G)} \tag{14}$$

where $a_G = \sqrt{\gamma' R' T_G}$. It is considered that the minimum area for inlet/outlet can be divided into small sets of areas (Fig. 1) with each set of these consisting of a U-shaped pipe with cross-sectional area of about $\pi(D/2)^2$ (0.0314 m^2, where $D_i = 2R = 0.2$ m). Furthermore, these sets of areas along the wing and the engine sides having a total length of 140 m may be divided by $2D_i + 0.4$ (pipe total thickness), which in turn gives a total of about 318 sets of area for the leading edge and the same number of sets of area for the other parts. Consequently, a maximum allowable coolant area of 9.985 m^2 (0.0314 m^2 $\times$ 318 = 9.985 m^2) is the result for each inlet and outlet area of leading edges, and similarly for the other parts, where this maximum allowable coolant area should not be exceeded. It should also be noted that the mass of coolant is not considered as fuel in the present study in order to accommodate a worst case.

Using Eqs. (12–14) for the ratio of any two coolant mass rates, minimum inlet area, and minimum outlet area, respectively, yield

$$\mathrm{MR} = \frac{[(\dot{m}_{\mathrm{cl}})_i]_2}{[(\dot{m}_{\mathrm{cl}})_i]_1} = \frac{[(m_{\mathrm{cl}})_i]_2}{[(m_{\mathrm{cl}})_i]_1} = \frac{[h_{ev} + C_p(\Delta T)]_1}{[h_{ev} + C_p(\Delta T)]_2} \tag{15}$$

$$\mathrm{ARI} = \frac{[(A_{\mathrm{min,il}})_i]_2}{[(A_{\mathrm{min,il}})_i]_1} = (\mathrm{MR})^2\left[\frac{(\rho_L\nu)_1}{(\rho_L\nu)_2}\right]^2 \tag{16}$$

$$\mathrm{ARO} = \frac{[(A_{\mathrm{min,ol}})_i]_2}{[(A_{\mathrm{min,ol}})_i]_1} = (\mathrm{MR})\left[\frac{(\rho_G\sqrt{\gamma' R' T_G})_1}{(\rho_G\sqrt{\gamma' R' T_G})_2}\right] \tag{17}$$

Table 1 Input data[11–14]

Coolant	h_{ev}	C_p	T_L	ν	ρ_L	ρ_G	γ'	R'
H_2	451,900	14,200	14.01	1.5714×10^{-7}	70	1.329	1.41	4,127
CH_4	577,400	2,253.7	90.67	3.686×10^{-7}	424	1.8004	1.31	518.35
He	23,932	5,230	0.95	1.6875×10^{-7}	125	16.002	1.66	2,079.4
Ne	87,027	1,050	24.48	2.51×10^{-7}	1,200	9.499	1.64	409.8
N_2	199,200	1,038	63.29	3.07×10^{-7}	804	4.613	1.4	296.6
Ar	1,620.76	523	83.95	2.333×10^{-7}	1,390	5.895	1.67	209.83

Equations (15–17) can be used to generate the result of any coolant (with subscript 2) with respect to any base coolant (with subscript 1). In this study liquid hydrogen is considered to be the base coolant, which we compare to the other coolants. Therefore, having the solution of the problem for the hydrogen, we can deduce the other coolants by Eqs. (15–17) for given ΔT. Also, Eqs. (15–17) can be used to generate the result of any coolant at any ΔT (with subscript 2) with respect to the same coolant at the base temperature difference, ΔT (with subscript 1). In this study $\Delta T = 150$ K is considered to be the base temperature difference that we compare the other temperatures to. Therefore, having the solution of the problem for any coolant at any base temperature difference (e.g., $\Delta T = 150$ K), we can deduce the solution at other temperature difference (e.g., $\Delta T = 100$ K) for the same coolant. This leads to the fact that Eqs. (15–17) can be used to generate mass, inlet area, and outlet area for any coolant at any temperature difference, provided that the result of a base coolant at base temperature difference is available.

The aerospace plane data are taken from the early studies in Refs. 8 and 9 such that the initial total mass $m_0 = 454{,}000$ kg, the total length of the vehicle = 58 m, the aerodynamic reference area of the vehicle = 860 m^2, the length for each wings' leading edge (considering worst case) l_{Wle} (=50 m), and the total length for engines on four sides of leading edges l_{Ele} (=40 m) (considering worst case). The physical properties of the coolant are taken from Refs. 12–16 and are shown in Table 1.

Results and Discussions

Results obtained from the study are for the six liquid coolants, H_2, CH_4, He, Ne, N_2, and Ar, for laminar and turbulent cases and for the temperature difference of ΔT. It should be noted that due to high Reynolds number the case appears to be turbulent in nature, for leading edges and for other parts. However, using a coolant may keep the flow close to laminar and will reduce the aerodynamic drag. Liquid H_2 is used as base coolant at base $\Delta T = 150$ K. The results for H_2 are shown in Figs. 2–10 for the mass ratio of coolant and fuel plus coolant with respect to initial total mass m_{cl}/m_0 and $m_{\mathrm{fl+cl}}/m_0$, and minimum areas required for inlet A_{il} and outlet A_{ol} of coolant mass flow for different flow conditions and temperature difference of $\Delta T = 150$ K. This is carried out for the stagnation point, leading edges, and other parts as found in Ref. 6.

The general results show that turbulent flow generates more heat rate and load than laminar flow, hence, it requires more coolant mass and a larger area of inlet/outlet for coolant passage than that corresponding to laminar flow. The increase in the parameter ΔT reduces the mass of the coolant and the area results in the reasonable range. The comparison of H_2, CH_4, He, Ne, N_2, and Ar shows that hydrogen gives better results as a coolant than all of the others.

The change in temperature ΔT can be taken to be a parameter, which can be chosen to start from $\Delta T = 0$ to 400 K. In this study the case of $\Delta T = 150$ K is considered as the base ΔT. The situation with $\Delta T = 150$ K is closer to reality than a lower ΔT. The results of any temperature changes (e.g., $\Delta T = 1$, 200, 300, and 400 K) can be deduced from Eqs. (15–17), which are shown in Table 2 and Figs. 11–16.

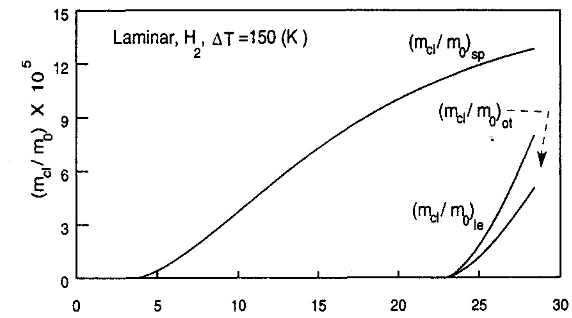

Fig. 2 (m_{cl}/m_0) for H_2, $\Delta T = 150$ K vs M_∞.

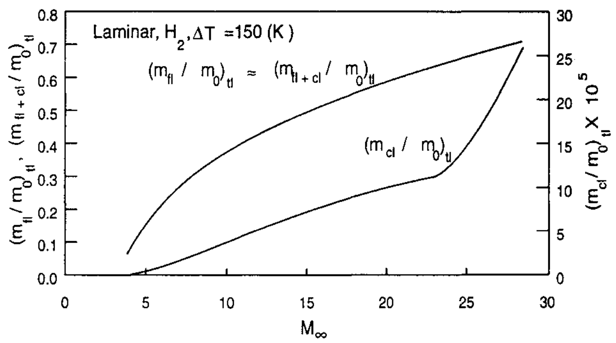

Fig. 3 $(m_{\mathrm{fl}}/m_0)_{\mathrm{tl}}$, $(m_{\mathrm{fl+cl}}/m_0)_{\mathrm{tl}}$ for H_2, $\Delta T = 150$ K vs M_∞.

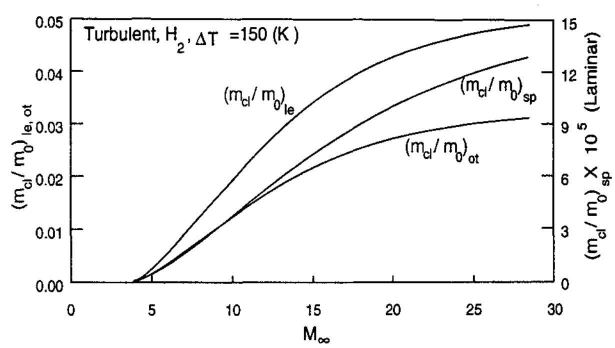

Fig. 4 $(m_{\mathrm{cl}}/m_0)_{\mathrm{le,ot}}$ for H_2, $\Delta T = 150$ K vs M_∞.

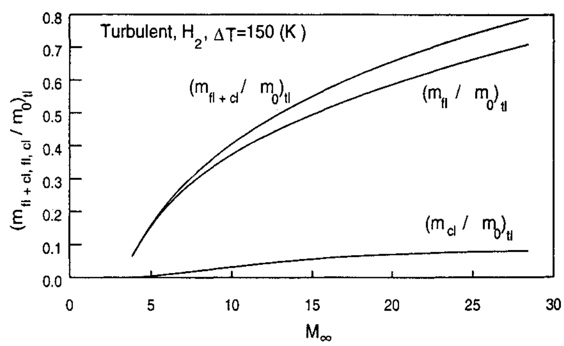

Fig. 5 $(m_{\mathrm{fl+cl,fl,cl}}/m_0)_{\mathrm{tl}}$ for H_2, $\Delta T = 150$ K vs M_∞.

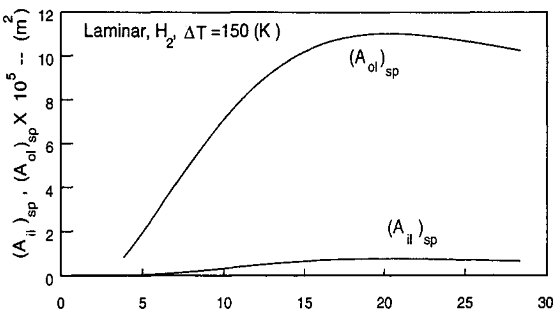

Fig. 6 $(A_{il})_{sp}$, $(A_{ol})_{sp}$ m^2 for H_2, ΔT = 150 K vs M_∞.

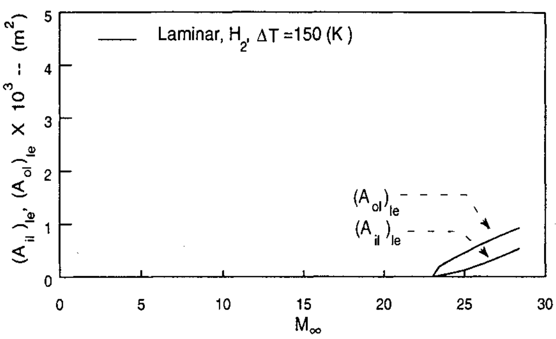

Fig. 7 $(A_{il})_{le}$, $(A_{ol})_{le}$ m^2 for H_2, ΔT = 150 K, laminar case vs M_∞.

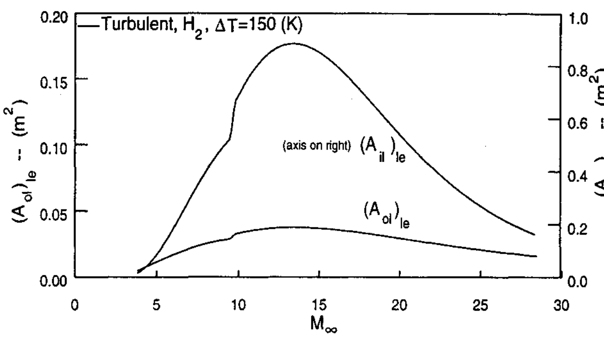

Fig. 8 $(A_{il})_{le}$, $(A_{ol})_{le}$ m^2 for H_2, ΔT = 150 K, turbulent case vs M_∞.

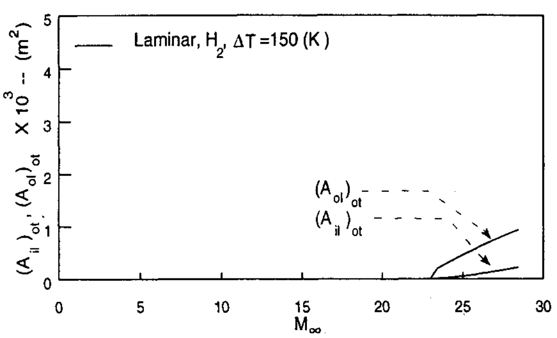

Fig. 9 $(A_{il})_{ot}$, $(A_{ol})_{ot}$ m^2 for H_2, ΔT = 150 K, laminar case vs M_∞.

To demonstrate the use of H_2 results (Table 2) and Figs. 11–16, an example to find the CH_4 coolant mass, its area inlet, and its area outlet for the leading edge in a turbulent case at M_∞ = 15 and ΔT = 1 K is performed. The steps are as follows:

1) The first step is to utilize the H_2 coolant (the base coolant) results in Fig. 8, which gives $(m_{cl}/m_0)_{le}$ = 0.034, $(A_{il})_{le}$ = 0.8 m^2, and $(A_{ol})_{le}$ = 0.035 m^2, at M_∞ = 15 and ΔT = 150 K.

2) The second step is to utilize Table 2, which gives MR = 2.8203, ARI = 0.03934, and ARO = 5.0176 for CH_4 with respect to H_2 (the base coolant), for any M_∞ and ΔT = 150 K.

3) The third step is to utilize Fig. 12, which gives MR = 1.58, ARI = 2.49, and ARO = 2.56 for CH_4 at any M_∞ and ΔT = 1 K with respect to CH_4 at any M_∞ and ΔT = 150 K.

4) The final step is the CH_4 result for turbulence, at M_∞ = 15 and ΔT = 1 K, which is obtained using steps 1–3 as follows:

$$(m_{cl}/m_0)_{le} = 0.034 \times 2.8203 \times 1.58 = 0.15$$

$$(A_{il})_{le} = 0.81 \times 0.03934 \times 2.49 = 0.79$$

$$(A_{ol})_{le} = 0.035 \times 5.0176 \times 2.56 = 0.45$$

These results agree with the results in Ref. 6 (see Figs. 11 and 16 in Ref. 6). However, here there is no need to integrate the equation numerically in order to get the results for each coolant at specific ΔT. The only thing needed is to use the results of the base coolant at base ΔT (i.e., H_2 at ΔT = 150 K) with the utilization of Table 2 and Figs. 11–16, and the results will be obtained for the mass of coolant with its inlet/outlet areas.

The important results are obtained, as in Ref. 6, for mass ratios (both coolant and fuel) with respect to an initial total mass for the stagnation point, leading edges, and other parts as shown in Figs. 2–5 for H_2. This is where the worst case of turbulent flow using ΔT = 150 K needs a total coolant mass for liquid hydrogen of more than 10% of the initial total mass m_0. If we add to this the fuel mass (which has been calculated in Refs. 8 and 9), we then need less than 80% of the initial total mass, this will leave us with more than 20% of the initial total mass as payload and the vehicle mass without fuel and coolant (see Fig. 5). Whereas, in a similar situation, if we use any of the other five liquid coolants, we would need more of the initial total mass as the coolant, which would consequently leave us with less mass of the aerospace plane and its payload

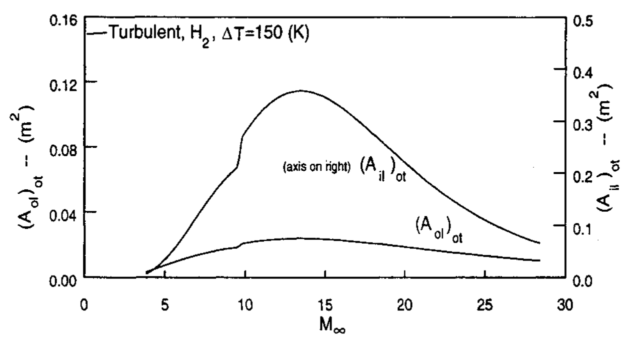

Fig. 10 $(A_{il})_{ot}$, $(A_{ol})_{ot}$ m^2 for H_2, ΔT = 150 K, turbulent case vs M_∞.

Table 2 Comparison between H_2[a] and the other five coolants[b] in terms of mass ratio, area inlet ratio, and area outlet ratio, for any M_∞ and at base ΔT = 150 K

Coolant	H_2	CH_4	He	Ne	N_2	Ar
MR	1	2.8203	3.1937	10.559	7.275	32.84
ARI	1	0.03934	2.769	0.15	0.106	1.241
ARO	1	5.0176	0.35769	4.2146	6.8802	25.26

[a]Base coolant. [b]CH_4, He, Ne, N_2, and Ar.

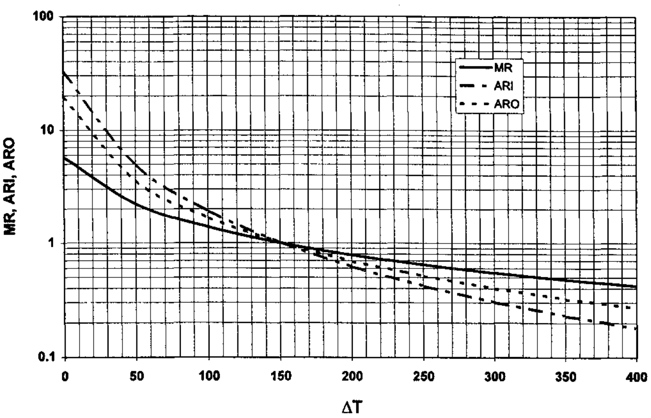

Fig. 11 MR, ARI, and ARO vs ΔT, for H_2 at any ΔT with respect to H_2 at ΔT = 150 K.

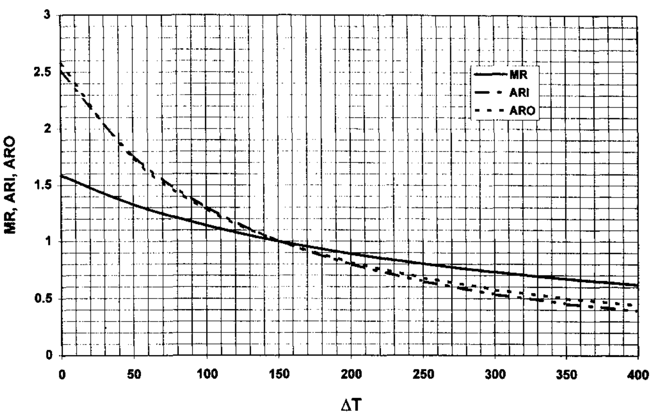

Fig. 12 MR, ARI, and ARO vs ΔT, for CH_4 at any ΔT with respect to CH_4 at ΔT = 150 K.

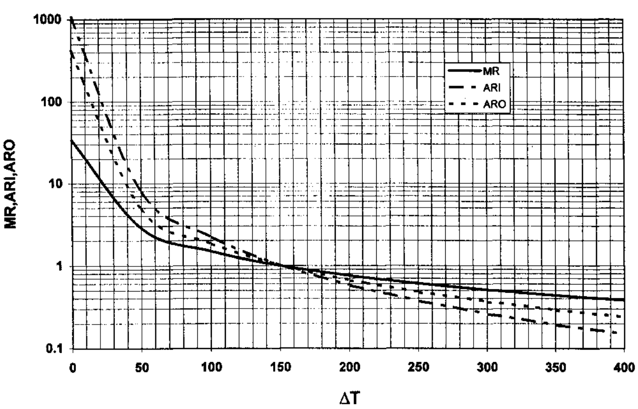

Fig. 13 MR, ARI, and ARO vs ΔT, for He at any ΔT with respect to He at ΔT = 150 K.

without fuel and coolant (see Table 2 and Figs. 11–16). This shows that hydrogen is clearly a better coolant. The areas required for H_2 coolant are shown in Figs. 6–10. The comparison between hydrogen and the other five coolants is done through Eqs. (15–17) and the results in Figs. 11–16. In general, the following trends are observed:

1) Hydrogen is clearly a better coolant compared to the other five coolants as shown in Table 2 and Figs. 11–16, where for a 1-unit mass of coolant of H_2 there are about 2.8203, 3.1937, 3.1937, 10.559, 7.275, and 32.84 unit mass coolants of CH_4, He, Ne, N_2, and Ar, respectively, at ΔT = 150 K.

2) The inlet area is smaller than the outlet area. For the same mass rate of the coolant at the inlet/outlet, the areas will be inversely proportional to density, noting that the density of liquid at the inlet is larger than the density of gas at the outlet. This observation is true for all of the cases except for the hydrogen, turbulent case, because in the later case $(\dot{m}_{cl}/\rho_L V_{in}) > (\dot{m}_{cl}/\rho_G V_{ol})$, as was observed also in Ref. 6.

3) The areas at inlet for H_2 are larger than those corresponding to CH_4, Ne, and N_2, this is due to the fact that liquid hydrogen density is smaller than these three coolant densities, in spite of the fact that mass flow of hydrogen is less than these three coolants, because the difference in mass flow is less than the difference in liquid density between these three coolants and H_2. The advantage of these three coolants over hydrogen becomes less important since for the turbulent case (the worst case), we have the leading-edges inlet area of 0.9 m^2 for liquid hydrogen, which is small enough and will not cause any serious problem because it is less than the maximum

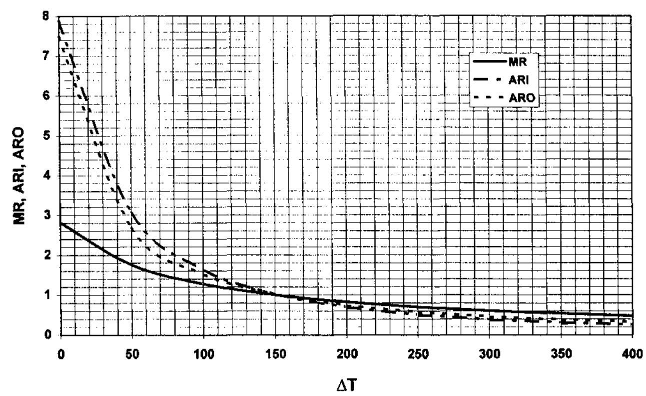

Fig. 14 MR, ARI, and ARO vs ΔT, for Ne at any ΔT with respect to Ne at ΔT = 150 K.

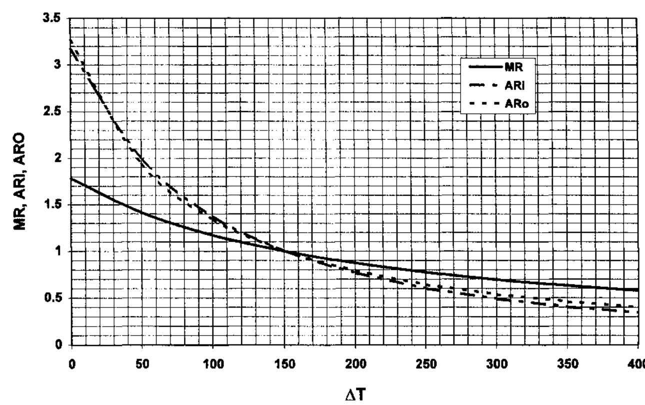

Fig. 15 MR, ARI, and ARO vs ΔT, for N_2 at any ΔT with respect to N_2 at ΔT = 150 K.

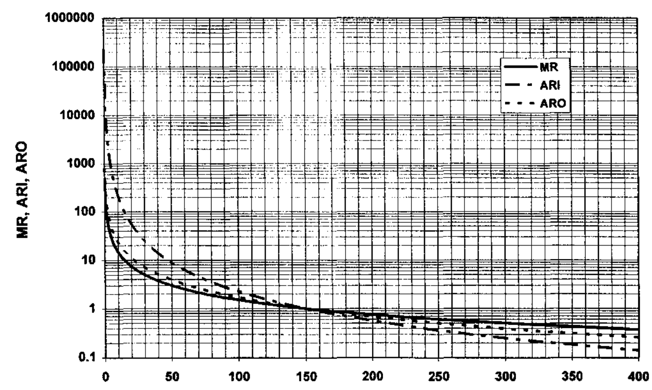

Fig. 16 MR, ARI, and ARO vs ΔT, for Ar at any ΔT with respect to Ar at ΔT = 150 K.

allowable coolant area (i.e., 9.985 m^2). At the outlet we have a smaller area for H_2 than CH_4, Ne, N_2, and Ar, this is due to the fact that the mass flow rate required for these four coolants is larger than that required for hydrogen and also because the difference in mass flow is more than the difference in densities at the gaseous state of the coolants, this shows an advantage of hydrogen over these four coolants. Table 2 shows the comparison between a 1-unit inlet area for hydrogen, with the rest of the coolants at $\Delta T = 150$ K.

4) For the outlet area, Table 2 shows the comparison between a 1-unit outlet area for hydrogen with the rest of the coolants at $\Delta T = 150$ K.

5) The area related to coolant inlet/outlet is directly proportional to the mass rate of coolant as shown in the H_2 results, which in turn is inversely proportional to the change in temperature ΔT. This is evident from Eqs. (12–17) and Figs. 11–16.

Conclusions

The study compares, for the worst case, the six coolants for an aerospace plane, liquid H_2, CH_4, He, Ne, N_2, and Ar. It is found from this study that there is no fundamental problem occurring in cooling the aerospace plane in terms of the coolant mass required and the areas needed for coolant mass flow passage.

The liquid hydrogen proves to be a better coolant than liquid CH_4, He, Ne, N_2, and Ar. The amount of coolant using liquid hydrogen is less by a substantial amount (for worst case) than that using liquid CH_4, He, Ne, N_2, and Ar. Moreover, liquid hydrogen or liquid methane can be used as a fuel, which gives them a great advantage over all other four coolants, which are inert, and this advantage will increase the mass of the payload.

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