UNCLASSIFIED

ADVANCED PROJECT NOTE #3

THE POSSIBILITIES OF HYDROGEN AS AN AIRCRAFT FUEL

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SUMMARY

The improvement in performance of high speed, high altitude aircraft following laminarisation of the boundary layer is discussed in this note. Owing to the very large reductions in skin drag coefficient (from 50% to 90% reduction possible) the possible increase in range is large.

The main factors affecting the Reynolds' Number for transition are discussed and in particular an analysis has been made of the requirements to hold surface temperature below the limit for an entirely laminar boundary layer over a body. This analysis led to the conclusion that fuels of much larger heat capacity than Kerosene must be used.

A P NOTE #3

THE POSSIBILITIES OF HYDROGEN AS AN AIRCRAFT FUEL

1. INTRODUCTION

Large reductions in drag and consequent increases in range are possible for finely shaped aircraft at all speeds if completely laminar boundary layers can be maintained over their surfaces.

However, the attainment of a laminar boundary layer at the high Reynold's numbers that are encountered with modern aircraft is not easy and the state of the boundary layer is determined by many factors. The leading parameters affecting the transition Reynolds' Number i.e. the Reynold's Number at which a change from completely laminar to fully turbulent boundary layer commences are as follows:-

(a) Flight Mach Number;

(b) Stream turbulence;

(c) Leading edge on nose geometry;

(d) Leading edge sweep on wings;

(e) Surface temperature;

(f) Surface finish (roughness);

(g) Pressure gradient;

(h) Angle of attack;

The effects of the above parameters are discussed by Low in reference #1 in the light of theory, wind tunnel test and free flight test on large models.

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(a) Flight Mach Number has a deletri[illegible] effect on the Reynolds' Number for transition (i.e. transition [illegible] forward and the domain of laminar boundary layer is decreased) as Mach Number is increased up to about 4. Above this value Mach. Number has a favourable effect and moves the transition region rearwards again. [illegible] type of body has some effect on the value of M for minimum transition Reynolds' Number. For insulated cones the minimum occurs at a lower Mach number than for insulated flat plates and hollow cylinders.

(b) Stream turbulence has a large effect on transition and this may be one of the factors affecting the difference between results of wind tunnel and free flight test inasmuch as the atmosphere has very low turbulence.

(c) Leading edge geometry also has a large effect on transition and the general conclusion is that a rounded nose or leading edge will cause higher values of transition Reynolds' number to be obtained.

(d) Wing sweep has a marked effect in destabilising an otherwise laminar boundary layer and should be avoided if possible.

(e) Surface temperature is probably the dominant parameter affecting transition and on this point theory and test gives widely different answers. It appears that the ratio of wall temperature to the static temperature at the edge of the boundary layer indicates where transition will occur. Test results from differing sources (see bibliography collected in Reference #1) agree that if the surface temperature Tw can be reduced such that $\frac{Tw}{T} \doteq 1.2$ where T is temp. at [illegible] boundary layer, then a boundary layer initially laminar will remain in [illegible] state. Theory predicts considerably higher values of $\frac{Tw}{T}$ to be permissible i.e. underestimating the cooling required.

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(f) Surface finish is not expected to present grave difficulties unless the protruberances penetrate through the laminar sub-layer of a turbulent boundary layer in which case surface cooling will possibly be unable to delay transition. Flight at high altitudes is beneficial in countering surface roughness owing to the thickening of the boundary layer. The thickness of a laminar or turbulent boundary layer increases as altitude increases, since the thickness $\delta \propto \frac{1}{\sqrt{R_E}}$ and the Reynolds' Number R_{ε} decreases with altitude increasing.

(g) Pressure gradients have a favourable effect on stabilizing a boundary layer if they are negative i.e. $\frac{dp}{dx} < 0$ where p is the local pressure and x the distance from nose or leading edge. Generally, supersonic flight is more amenable to producing favourable gradients than is subsonic flight. A thin bi-convex wing at zero angle of attack has a favourable gradient over the entire surface whereas in subsonic flows a favourable gradient only exists over the forward portion up to the point of maximum thickness.

(h) Angle of attack affects transition by decreasing R_{ε_T} on the upper surface and increasing it on the lower surface, so that for a wing with mixed flow the net change in friction drag would be small.

In the case of flight at high supersonic speeds angles of attack and changes of angle of attack to execute a manoeuvre would be small so that little change in friction drag on this account is expected.

In this note particular emphasis is placed on the achievement of laminar boundary layers due to surface cooling. As will be shown this leads to the [illegible] of [illegible] other than the [illegible].

[illegible]

2. ANALYSIS

An aircraft cruising at a Mach Number of 4 (which according to Ref #1 is the speed for lowest ~~REP~~ R_{E_T}) at an altitude of 100,000 ft. is considered.

Ambient temperature = 216.5°K

If Tr is the stagnation recovery temperature

$$Tr = 216.5\ (1 + .2\ M^2)$$

R being the stagnation temperature recovery factor and M the flight Mach Number.

For laminar boundary layers $R = P_r^{\frac{1}{2}}$

and for turbulent boundary layers $R \doteqdot P_r^{1/3}$

where P_r the Prandtl number is .72 for air

Then $R_{Laminar} = .848$

$R_{Turbulent} = .862$

Owing to the closeness of these values no appreciable difference in Tr will result if either is used.

Taking R = .85

$$Tr = 216.5 \times 3.72 = 804°K$$

In order that an initially laminar boundary layer will stay in that condition free flight tests have shown that Tw must be reduced such that $\frac{Tw}{T} \leq 1.2$

i.e. $Tw \leq 260°K$ (based on the assumption that T is the free stream value).

Further it may be expected (as the result of free flight test) that in spite of high wall temperature near the ...

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of [illegible] there will be a certain length [illegible] the nose or leading edge over which laminar boundary layer flow naturally exists and thus length is defined by the Reynolds' number for the start of transition of 3.0×10^6 for M = 4.0 (see Low's report [illegible]

Then if l is the length in question

$$3.0 \times 10^6 = \frac{4.0 \times 968}{\nu}$$

and ν is dependent of altitude

<u>Altitude = 50,000 ft.</u> $\nu = 8.175 \times 10^{-4}$ ft /sec

then $\ell = \dfrac{3.0 \times 8.175 \times 10^2}{4 \times 968} = .633$ ft. Ta = 216.5°K

<u>Altitude = 70,000 ft.</u> $\nu = 21.317 \times 10^{-4}$ ft /sec.

$\ell = \dfrac{3.0 \times 21.317 \times 10^2}{4 \times 968} = 1.65$ ft. Ta = 216.5°K

<u>Altitude = 90,000 ft.</u> $\nu = 55.346 \times 10^{-4}$

ℓ = 4.29 ft. Ta ₌ 216.5°K

<u>Altitude = 100,000 ft.</u> $\nu = 89.456 \times 10^{-4}$

ℓ = 6.93 ft. Ta = 216.5°K

<u>Altitude = 110,000 ft.</u> $\nu = 157.2 \times 10^{-4}$

ℓ = 12.18 ft. Ta = 228.9°K

<u>Altitude = 120,000 ft.</u> ν = 287.6

ℓ = 22.3 ft. Ta = 251°K

(Above values for ν and Ta are from N.A.C.A. standard atmosphere)

The above values of laminar boundary layer length demonstrate the advantage in flying at high altitude, due to the rapid increase in the kinematic viscosity with altitude. Another advantage of high altitude flying is the greatly reduced heat input from boundary layer to the skin owing to the much reduced air density at high altitude.

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Now the surface temperature of a body at high speed falls from high values at the nose to lower values further back.

In the case of M=4 at 100,000 ft. the first 6 ft. or so will be laminar. The next x ft. will be at a higher [illegible] than 1.2 T and the remaining $l-x-6$ feet may be at a lower surface temperature than 1.2T and so require no cooling (l is the total length of the body).

2.1. <u>[illegible] OF LENGTH x TO BE COOLED BY MECHANICAL METHODS FOR M = 4 [illegible]</u>

The length x is cooled by mechanical means and its value will roughly determine how much of a body has to be cooled and how much will remain cool enough by radiation.

Heat input into body from boundary layer = $h_\omega (A_\omega)_x (T_R - T_\omega)$

where $(Aw)_x$ is the surface area of body up to the length x

Heat radiated from wall = $q_r = \epsilon T_\omega^4 \times 2.78 \times 10^{-12} \times (A_\omega)_x$ C[illegible]

where ϵ is the emissivity constant which is 1.0 for a black body.

In order that the lowest possible estimate of the length to be cooled shall be made ϵ will be taken as 1.0 and no solar heating will be considered.

Then $= (Aw)_x \times 1.0 \times (260)^4 \times 2.78 \times 10^{-12} = .0127\ (Aw)_x$

For equilibrium $.0127\ (Aw)_x = h_\omega (Aw)_x (T_R - 260)$

and the recovery temperature $T_R = 804^\circ K$

Then $h_\omega = \frac{.0127}{544}$

For $h_\omega = S_t \rho u C_p$ where S_t is the Stanton number, ρ the free stream density, u the free stream velocity and C_p the specific heat of air at constant pressure.

At 100,000 ft. $\rho = 1.066 \times 10^{-3}$ lb/ft^3

$u = 3[illegible]2$ ft/sec

$C_p = .24$

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Further the Stanton number is related to the skin friction coefficient by C_f by,

$$S_t = \frac{C_f}{2S}$$

where S is the Reynolds' anology factor and has a value of 0.835 (Ref 2)

Also for a laminar bo undary layer $C_f = \frac{1.209}{\sqrt{R_{Ex}}}$

Where RE_x is the Reynolds number referred to the length x

$$\text{Hence} \quad \frac{1.209}{\sqrt{R_{Ex}}} = \frac{1.65 \times .0127 \times 10^3}{544 \times 1.066 \times 3872 \times .24}$$

$$\text{i.e.} \quad \sqrt{R_{Ex}} = \frac{1.209 \times 544 \times 1.066 \times 3872 \times .24}{1.65 \times .0127 \times 10^3}$$

$$= 3.11 \times 10^4$$

$$\text{i.e.} \quad R_{Ex} = \frac{3872 \times \quad}{89.456} \times 10^4 = 9.69 \times 10^8$$

$$\text{So that} \quad x = \frac{9.69 \times 89.456}{3872} \times 10^4 = 2240 \text{ ft.}$$

This value is much in excess of any practical body length and shows that the entire surface of the body has to be cooled by mechanical methods.

The length required to be cooled is strongly dependent on the temperature ratio $\frac{Tw}{T}$ A calculation for $\frac{Tw}{T}$ of 2.0 for the same Mach Number and altitude has shown that only 3.9 feet need be cooled the remainder behind the portion having a ratio of 2.0 or less maintained by radiative cooling. This calculation also showed that very much greater lengths have to be cooled at lower altitudes viz. 26 feet at 80,000 ft. and 174 feet at 60,000 ft. (Fig. 1)

2.2. Feasibility of skin Cooling Using JP-4

A hypothetical aircraft 50 feet in length has been considered cruising at M = 4.0 and 100,000 ft.

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[illegible]

= .04 [illegible]

Ambient density ρ, at 100,000 ft. = 1.066×10^{-3} lb/ft

Flight velocity = 3072 feet/sec.

Emissivity = .05 (polished metal)

Stefan - Boltzmann constant B = 2.73×10^{-12} CHU/ft^2 sec. (°K)^4

Heat input $q_w = 1.212 \times 10^{-4} \times .04 \times 1.066 \times 10^{-3} \times 3072$ [illegible]

$.05 \times 2.73 \times 10^{-12} \times$ [illegible]^4

q_w = [illegible] CHU / ft². sec.

[illegible]

[illegible] (Hrs.)	WEIGHT OF FUEL (lbs)	VOLUME (Ft^3)	[illegible]
0 - .2	19,500	406	[illegible]
.2 - .4	15,000	312	[illegible]
.4 - .6	11,100	231	[illegible]
.6 - .8	7,900	165	[illegible]
.8 - 1.0	5,500	115	[illegible]
1.0 - 1.165	3,400	71	[illegible]

Heat transferred to fuel = $C \rho_f \tau_f \frac{d\bar{T}}{dt}$

Where C = specific heat of fuel = .5, $\bar{T}$ is mean [illegible] fuel [illegible]

ρ_f = fuel density = 48 lb/ft^3

τ_f = mean thickness of fuel layer.

Then $\frac{d\bar{T}}{dt} = \frac{1}{\tau_f} \times \frac{.0647}{48 \times .5} = \frac{.02627}{\tau_f}$

Using this equation and the above table the approximate rise in temperature of the fuel during cruise is found to be 260°C. so that even if at the commencement of cruise the fuel is at its freezing point a rise of 260°C. will bring the temperature up to approximately 170°C. which is considerably above the practical upper limit.

2.3. FEASIBILITY OF SKIN COOLING USING LIQUID HYDROGEN

It is assumed that an insulated tank of liquid hydrogen can [illegible] installed in an aircraft fuselage and that suitable pipi[illegible] whereby the liquid hydrogen is [illegible] from the tank [illegible] outer skin of the [illegible] [illegible]

If w lb/sec is the weight flow of hydrogen being evaporated, T_v the vaporising temperature and T_c the temperature entering the combustion chambers then

$$.0647 = \frac{w}{S}\left[h_v + C_{P_H}(T_c - T_v)\right]$$

where h_v is the latent heat of evaporation and C_{P_H} is the mean specific heat of gaseous hydrogen over the temperature range T_v to T_c . S is the cooled surface area.

As an approximation it will be taken that the skin temperature T_w is equal to T_c

Then $.0647 = \frac{w}{S}\left[h_v + C_{P_H}(T_w - T_v)\right]$

Using a cooled surface area of 750 sq. ft., a latent heat of 108 CHU/lb, a mean specific heat of gaseous hydrogen of 3 CHU/lb.°C , a vaporising temperature of 20°K and a desired surface temperature of 260°K then

$w = .0647 \times 750/(108 + 720) = .0586$ lb/sec.

3. RAMJET PERFORMANCE

An estimate of the specific net thrust and specific fuel consumption has been made for the following conditions:-

(a) M = 4.0

(b) Altitude = 100,000 ft. (ambient temp. = 216.5°K)

(c) Max burning temperature = 2,000°K.

(d) A.I.A. supersonic diffusion efficiency i.e. 48.7% at M = 4

(e) Subsonic diffuser loss = 5% of total load after normal shock.

(f) Flameholder loss = 2x dynamic lead at entrance to combustion chamber.

(g) Mach Number of 0.3 at entrance to combustion chamber.

(h) Standard losses due to heat addition in a constant area duct.

(i) Zero final nozzle losses and full expansion to ambient static pressure of 22.4 lb/sq.ft.

With these assumptions the specific net thrust is 56.6 lb.sec/lb.

The heating value of hydrogen is 28,640 CHU/lb. and assuming a combustion efficiency of .95 (due to the high flame velocities with burning hydrogen) the fuel/air ratio required is .0116

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[illegible]

... heating to 2[illegible]0°K the SFC [illegible] ... of .85 and an ideal fuel/air ratio of .0307 is found to be 2.3 lb./[illegible]

4. [illegible] DRAG

An estimate of the drag of the aircraft with both uncooled and [illegible] ... has been made for the following assumptions.

4.1. Uncooled Surfaces (JP-4 fuelled)

(a) Take off weight = 50,000 lb.

(b) Empty weight = 25,000 lb.

(c) Landing speed = 150 m.p.h. = 220 ft/sec.

(d) Lift coefficient for landing = 1.0

Then the gross wing area of the aircraft is given by

$$S_W = \frac{25,000}{1/2 \times .00238 \times 220^2 \times 1.0}, \quad S_w = 434.5 \text{ ft.}^2$$

Mean cruising weight = 37,500 lbs.

Mean cruising wing loading = 86.3 lb/ft.^{2}

Further assumptions are:-

(e) Wing aspect ratio = 5

(f) Root chord = 2 x tip chord ($\text{ie } C_R = 2 \times C_T$)

If Wing span is 2b then $5 = \frac{4b^2}{434.5}$ i.e. b = 23.3 ft.

and if the body diameter is 5 ft.

Then $434.5 = 5\,C_R + 1/2\,(C_R + C_T)(b - 2.5)$
$434.5 = 10\,C_T + 3\,C_T \times 20.8$
i.e. $C_T = 6$ ft. and $C_R = 12$ ft.

Wing mean chord = 9 ft.

Body Skin Drag - Turbulent boundary layer.

Body length = 50 ft.

Hence, Reynolds' Number, [illegible] = [illegible]

[illegible] friction [illegible] C_F = [illegible]

Body Drag = 750 x .0[illegible] x 25[illegible] = 1[illegible] lb.

where the dynamic pressure of the free stream [illegible] Mach Number of [illegible]

at altitude 100,000 ft. is 251 lb./ft.^{2}.

Body Wave Drag

Taking a nose semi angle of 15° the wave drag coefficient referred to the body frontal area is 0.2.

Then body wave drag = $0.2 \times 251 \times \frac{\pi \times 5^2}{4}$ = 980 lb.

Wing Skin Friction $R_E = \frac{9}{50} \times 2.16 \times 10^7 = 3.89 \times 10^6$

C_F = .0014, Wing skin drag = .0014 x 251 x 2 x 374.5 = 263 lb.

Wing Wave Drag

The thickness chord ratio for the wings is set at 0.03.

Wing wave drag coefficient, $C_{D_{WW}} = \frac{4}{\sqrt{M^2-1}}\left[\alpha^2 + \left(\frac{t}{c}\right)^2\right]\left[1 - \frac{1}{2R\sqrt{M^2-1}}\left(1 - \frac{C_2}{C_1} \ldots\right.\right.$

where α is the angle of attack in radians

$\frac{t}{c}$ is the thickness chord ratio

R is the aspect ratio

C_1 and C_2 are the Busemann coefficients.

For M = 4, C_1 = .5164 and C_2 = 1.232

Then $C_{D_{WW}} = 1.025\left[\alpha^2 + .009\right]$

The lift coefficient $C_L = \frac{4\alpha}{\sqrt{M^2-1}}\left[1 - \frac{1}{2R\sqrt{M^2-1}}\left(1 - \frac{C_2}{C_1} \ldots\right.\right.$

i.e. $C_L = 1.025\alpha$

Cruise lift coefficient = $\frac{86.3}{\text{Dynamic pressure}} = \frac{86.3}{251}$ = .34

Then α = .3316 radians

$C_{D_{WW}}$ = 1.025 ($.3316^2$ – .009) = .1135

This coefficient is based on the exposed plan area of the wings.

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[illegible] area = 434.5 - 5 x 12 = 374.5 ft^2

[illegible] drag = .1135 x 251 x 374.5 = 10,680 lbs.

Induced Drag

Induced drag coefficient $C_{Di} = \dfrac{C_L^2}{\pi R e}$

where e is the span efficiency and will be taken as .5

Then $C_{Di} = \dfrac{.34^2}{\pi \times 5 \times .5} = .0147$

Induced drag = .0147 x 434.5 x 251 = 1,600 lbs.

Total drag = 190 + 980 + 263 + 10680 + 1600 = 13713 lb.

Mean lift ÷ drag ratio = $\dfrac{37500}{13713}$ = 2.73

4.2. Cooled Surface (Hydrogen fuelled)

Assumptions

(a) It is assumed that the cooled surface is sufficient to stabilise the wholly laminar boundary layer over the body.

(b) Cooling permits use of aluminum structure with attendant airframe weight benefits over the JP-4 aircraft which will necessarily have a steel skin.

(c) Fuel weight fixed at 2290 lb. This is derived from the same volume of hydrogen as JP-4 and a density of 4.4 lb/ft^3 compared with 48 lb/ft^3 for JP-4.

(d) Landing speed = 150 m.p.h.

(e) Lift coefficient for landing = 1.0

Similar calculations to those outlined in section 4.1 have been made for a series of airframe weights and the values of Gross weight ÷ Empty Weight with the corresponding values of Life ÷ Drag ratio are tabulated below. Body dimensions are the same as those for the JP-4 aircraft:

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GROSS WEIGHT / EMPTY WEIGHT, $\frac{W_g}{W_e}$	$\frac{L}{D}$	DRAG (lb)
2.0	1.85	1857
1.75	2.032	2067
1.5	2.38	2404
1.248	2.958	3503

Using the Breguet range equation in the form

$$X = \frac{3600}{SFC} \times \frac{L}{D} \times \log_e \frac{W_g}{W_e}$$

Where X is the range

$$\frac{X_{H_2}}{X_{JP4}} = \frac{(SFC)_{JP4}}{(SFC)_{H_2}} \times \frac{(L/D)_{H_2}}{(L/D)_{JP4}} \times \frac{\log_e \left(\frac{W_g}{W_e}\right)_{H_2}}{\log_e \left(\frac{W_g}{W_e}\right)_{JP-4}}$$

where $(SFC)_{JP4} = 2.3$

$(SFC)_{H_2} = 0.738$

$(L/D)_{JP4} = 2.73$

$(W_g/W_e)_{JP4} = 2.0$

Values of $\frac{X_{H_2}}{X_{JP-4}}$ are tabulated below:-

$\left(\frac{W_g}{W_e}\right)_{H2}$	$\frac{X_{H_2}}{X_{JP4}}$
2.0	2.12
1.75	1.875
1.5	1.59
1.248	1.08

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4.[illegible] [illegible] FLOW REQUI[illegible]

In Section 2.3 it was shown that the approximate w[illegible] of hydrogen required for skin cooling was [illegible] .0584 lb/sec.

From the table given in section 4.2 the lowest aircraft dr[illegible] is [illegible]

Then for an SFC of .738 lb/lb.hr

$$\text{Hydrogen weight flow} = \frac{1857 \times .738}{3600} = .381 \text{ lb/sec}$$

This figure is much larger than that estimated for cooling alone so that some insurance is provided, to offset the deletrious effects of skin roughness on laminar boundary layer stability, through the greater skin [illegible]

Flow Passage Areas

With the most pessimistic figure for weight ratio viz. 1.248 the hydrog[illegible] flow required is .718 lb/sec.

If the flow area is evaluated at the entrance to the combustion cha[illegible]

$$.718 = \rho A v$$

where ρ is the density at temperature T_c (lb/ft^2)

A is the passage area in sq. ft.

v is the flow velocity in ft/sec.

$$\text{Now } \rho = \frac{p}{R T_c}$$

and for hydrogen the gas constant $R = \frac{2780}{2} = 1390$

$$\text{Then } A = \frac{.718 \times T \times 1390}{p \times a_h \times M_f}$$

where a_h is the speed of sound in hydrogen at temperature T_c and M_f is the flow Mach Number.

The temperature $T_c = T_W = 260°K$ [illegible]

Since the ratio of specific heats for hydrogen is 1.3

Then $a_h = \sqrt{32.2 \times 1.3 \times 1390 \times 260} = 3890$ ft/sec.

$$\text{Then } A = \frac{.718 \times 260 \times 1390}{p \times 3890 \times M} = \frac{66.7}{p M_f} \text{ sq. ft.}$$

A [illegible] value for M_f might be 0.2 so that A = $\frac{333.5}{p}$ sq. ft.

The [illegible] pressure at the [illegible] section of the [illegible]

[illegible] .[illegible] x .[illegible] x 22.4 $(1 + \frac{1.3}{5})^{3.5}$

$= 1555$ lb/sq. ft.

Static pressure at this section $= \frac{1555}{1.0644} = 1462$ lb/sq.ft.

Since the fuel pressure at the nozzles need not be greater than [illegible] then the largest flow area required for the hydrogen is

$$A = \frac{333.5}{1462} = .228 \text{ sq. ft.}$$

The width d of an annular type nozzle of 4 feet diameter would be

$$d \simeq \frac{.228 \times 12}{\times 4} \simeq .218 \text{ inches}$$

With a final discharge coefficient of .6

$d \simeq .363$ inches.

CONCLUSIONS

It is apparent that large decreases in drag and consequent increases in range are possible if laminar boundary layers can be maintained at high flight speeds. The greatest advantages of course accrue with bodies of very large fineness ratio, the surface area of these bodies being large compared with frontal area.

Among the many factors which determine the state of a boundary layer in compressible flow, surface temperature has a strong effect and apart from its importance to maintain structural integrity at high speeds, skin cooling can pay large dividends in range if it is adequate to laminarise an otherwise turbulent boundary layer.

There are two basic methods of skin cooling:

(a) Film or 'sweat' cooling;

(b) Cooling by internal absorbtion of heat generated in the boundary layer.

Film cooling has not been discussed in this note but it is pointed out that a disposable liquid must be carried by the aircraft and while it may adequately cool the surface it is difficult to visualise its giving a large contribution to propulsive thrust.

Cooling by internal absorbtion usually [illegible]

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[illegible] the bulk of the heat absorbed [illegible] into the [illegible] [illegible] possibly giving an increase in overall efficiency [illegible]

As pointed out in Ref. 4 [illegible] an ordinary kerosene fuel is inadequate to deal with the high [illegible] transfer given by flight at M = 4.0 at 100,000 ft. and fuels of [illegible] heat capacity must be considered.

In spite of its bulk liquid hydrogen [illegible] promising owing to its large heat capacity, (determined by its low boiling point and its large specific heat) and by its high calorific value of 28000 CHU/lb.

NOTE: If hydrogen could be stored as a liquid in its atomic form (nacent hydrogen) the heat of re-association could bring the theoretical heating value of hydrogen up to 78000 CHU/lb.

1. Boundary - layer transition at supersonic speeds:

G. M. Low NACA RM E56 E10

2. The Problem of Aerodynamic Heating. Van Driest Paper presented

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3. The Determination of Skin Temperature Attained in High Speed Fli

Davies and Monaghan RAE Report Aero 2454

4. Surface protection and Cooling System for High Speed Flight

Masson and Gazley.

Aero Engineering Review, November 1956.