Table 2 Comparison of (Y/Y_i) based on integral approach of Baronti and Libby with those based on present analysis

				Estimated values of (Y/Y_i) based on		
Author(s)	M_∞	Re_∞	T_w/T_∞	Integral approach of Baronti and Libby	Present method using local properties	Present method using wall properties
Coles	1.982	6.18×10^6	Adiabatic wall	2.23	2.12	2.40
	2.578	8.32×10^6	"	3.25	3.07	3.65
	3.697	7.25×10^6	"	6.66	5.93	7.82
	4.554	6.83×10^6	"	11.09	9.33	13.22
Matting et al.	2.95	9.0×10^6	"	3.97	3.87	4.74
	4.20	6.20×10^6	"	8.33	7.72	10.70
	6.70[a]	7.20×10^6	"	16.54	16.29	23.53
Moore and Harkness	2.669	1.41×10^9	"	3.54	3.68	3.99
Hill	9.07	3.70×10^6	8.30	26.95	28.84	35.79
Winkler and Cha	5.24	5.11×10^6	4.97	10.43	10.78	14.60

[a] Helium data.

It is of interest to compare our estimates with those based on the integral approach of Baronti and Libby[8] for correlating velocity profiles in high-speed flows. It may be remarked here that their approach makes use of properties at the outer edge of sublayer to form the so-called sublayer Reynolds number that is assumed to be invariant. A comparison for a few selected values of experimental results (for which necessary data has been tabulated in Ref. 8) is given in Table 2. It shows that for adiabatic flows in the low supersonic range the difference between three estimates is not of any significance. However, it turns out that especially for nonadiabatic flows in the high Mach number range, the estimates based on the integral approach of Baronti and Libby agree remarkably well with those based on the present analysis using local fluid properties. This may be, perhaps, a sheer coincidence, in view of the fact that Eq. (3) is an experimental criterion based on wall properties.

Concluding Remarks

Based on experimental criterion for laminar-sublayer thickness in compressible turbulent boundary layers, a general but simple and explicit expression has been derived for predicting laminar-sublayer thickness in compressible turbulent boundary layers. The expression, which is in the form of a ratio with respect to the corresponding value in incompressible case, has been presented as an explicit function of Mach number, Reynolds number, Prandtl number, and wall-to-freestream temperature ratio. It is found that this ratio is a weak function of Reynolds number. However, it increases rapidly with Mach number but decreases with cooling (Fig. 1). The difference between the estimates based on the use of wall properties and those based on local fluid properties becomes appreciable in the high Mach number ranges and also for nonadiabatic walls.

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Active Cooling of a Hydrogen-Fueled Scramjet Engine

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Nomenclature

BL = boundary layer
I_{sp} = specific impulse, sec
T_{c_1} = hydrogen coolant inlet temperature, °F
T_{c_2} = hydrogen coolant exit temperature, °F
T_w = wall temperature, °F
T_{w_2} = wall exit temperature, °F
α = angle of attack, deg
η = heat exchanger efficiency as defined by Eq. (1)
ϕ_T = combustion equivalence ratio, defined as the fuel-to-air ratio required for propulsion divided by the stoichiometric fuel-to-air ratio

Introduction

ANALYTICAL studies were performed to determine engine cooling requirements for a hydrogen-fueled, scramjet powered, high-altitude Mach 12 cruise aircraft. The thermal environment requires heat protection for the internal engine walls that would otherwise reach temperatures in the order of 7500°F. Consideration of various cooling methods led to the conclusion that regenerative cooling using the hydrogen fuel as a coolant is the only feasible approach. This Note considers some of the thermal requirements and performance tradeoffs associated with the regeneratively cooled engine.

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Integrated Heat Exchanger Design

Rectangular, half-round, and round tube heat exchanger core geometries were studied to determine the effect on scramjet weight and aircraft performance. Mach 12 cruise and a range of 8000 naut miles were used as the bases for comparison. The cross-sectional area of coolant passages was the same for all three geometries, and heat exchanger weights were based on a coolant pressure of 700 psia and a wall temperature of 2200°F. Designs were based on T.D. nickel-chrome construction. The primary structure of the engine was assumed independent of heat exchanger core configuration.

The rectangular core was selected as the most desirable shape even though it results in the highest unit engine weight. The height of flow passages can be tailored to accommodate varying heating rates and pressure drop requirements, and the width of flow passages can be adjusted in accordance with the width of the heat exchanger panel. The half-round shape offers a slight improvement in aircraft range (0.9%) over the rectangular shape but was rejected because variations in cross-sectional area would have been more difficult and expensive to achieve. Round tubes result in the lowest unit engine weight because of their inherent advantage as a pressure vessel. However, the wetted surface area is theoretically 57% greater, causing this configuration to operate at an equivalence ratio greater than one. The saving in scramjet engine weight, 36.5%, is more than offset by the increased coolant required. The net result is a 12% range penalty compared to the rectangular shape.

Hydrogen coolant flow requirements were determined with the aid of a heat exchanger efficiency defined as follows:

$$\eta = (T_{c_2} - T_{c_1})/(T_{w_2} - T_{c_1}) \qquad (1)$$

When defined in this manner, the efficiency is equal to the ratio of ideal to actual coolant flow rate. The ideal (η = 100%) is the theoretical minimum flow rate that would be required if the hydrogen coolant could be heated to the exit wall temperature of the heat exchanger. Parametric studies identified small flow passage dimensions as the key variable in achieving high heat exchanger efficiency. Flow passages in the order of 0.05 to 0.15 in. are considered practical limits because of pressure drop and fabrication considerations, and coolant side heat-transfer characteristics, respectively. For the purpose of this study, a nominal core dimension of 0.10 in. was assumed, which results in a heat exchanger efficiency of nearly 80% at a typical Mach 12 scramjet heating condition.

Gasdynamic Heating

A flat plate turbulent heating expression, with the fluid properties evaluated at a reference temperature to account for compressibility effects, was utilized to predict engine heating rates as presented in Fig. 1.

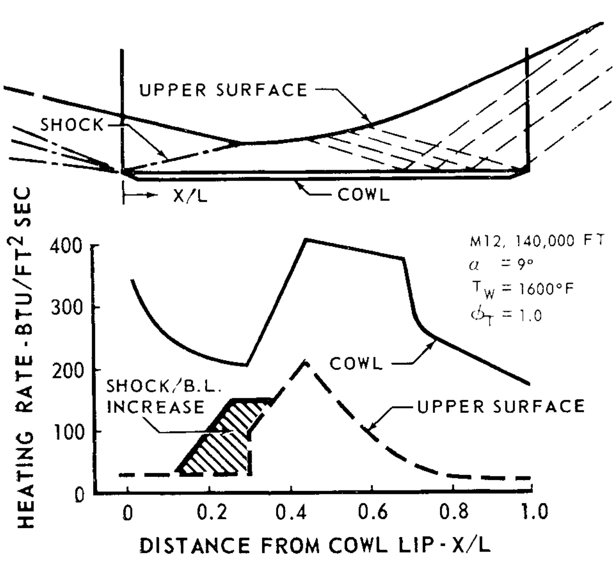

Fig. 1 Engine heating rates.

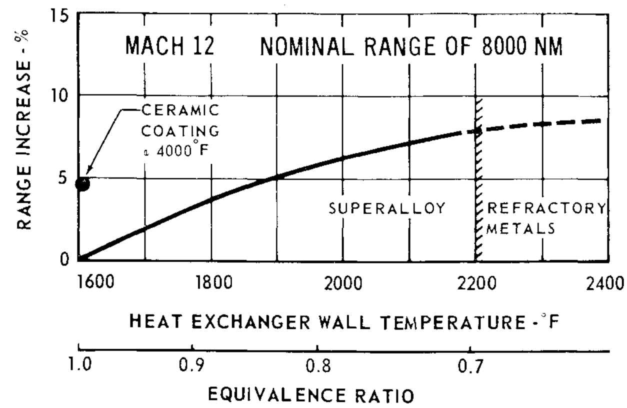

Fig. 2 Range increase vs wall temperature.

In the inlet, higher heating rates occur on the cowl surface than on the upper engine wall because of a shorter characteristic distance and higher pressures and temperatures behind the internal shock wave at the cowl lip. The increase in local heating rates due to this strong shock wave interacting with the turbulent boundary layer on the upper engine wall is noted in the figure. Although peak heating rates are increased by a factor of 5 relative to undisturbed values, the over-all increase in engine heat load is a modest 4%.

Strut heating rates, which vary between cowl and upper wall heating rate values, have been omitted from Fig. 1 for clarity. Cooling of the struts accounts for about 25% of the total absorbed heat load in the engine. The cowl and upper surface account for about 60 and 15%, respectively.

Engine Cooling

Since specific impulse (I_{sp}) and aircraft range increase as combustion equivalence ratio decreases, it is desirable to operate at as low an equivalence ratio as possible. At Mach 12, the principle constraint upon operation at reduced equivalence ratios is engine cooling. That is, as equivalence ratio decreases, the physical size of the engine (more capture area required to maintain thrust) and the total engine heat load increase. At the same time, the hydrogen flow available for cooling (fuel flow required by engine) decreases. To absorb the increased heat load, the available heat sink capacity of the hydrogen coolant must increase; this is accomplished at the expense of higher engine wall temperatures. Hence, there is a tradeoff to be performed. Although reducing the equivalence ratio increases range due to an increase in I_{sp}, it also results in increased engine size and higher wall temperatures. Both of these latter effects increase the weight of the engine, which tends to nullify the range benefit gained through an increase in I_{sp}. A tradeoff in terms of percentage range improvement as a function of wall temperature is presented in Fig. 2; the reference point is ϕ_T = 1.0 operation at a heat exchanger wall temperature of 1600°F. As indicated by the figure, the range improvement due to higher I_{sp} overrides the range penalties associated with increased engine weight. For a fixed range aircraft, operation at a reduced equivalence ratio would reduce cruise fuel costs.

On the basis of performance alone, the designer would select the highest temperature heat exchanger material available and accept the associated increased in engine weight. However, other factors such as development time and cost, fabricability, maintenance, and life influence the final selection for a given aircraft application.

At a given heat exchanger wall temperature, coolant requirements can also be reduced by adding a suitable coating (e.g., ceramic) to the hot gas side of the heat exchanger, thereby increasing the effective wall temperature and decreasing the hot gas convective heat input. The resultant increase in range (4.9%) associated with the addition of a ceramic coating (T_w = 4000°F, ϕ_T = 0.7) to a heat exchanger operating at 1600°F is noted in Fig. 2. Even though the addition of a ceramic coating permits operation at the minimum equivalence ratio considered in this study, ϕ_T = 0.7, the range increase (4.9%) is less than for the uncoated heat exchanger (7.7% at T_w = 2200°F, ϕ_T = 0.7) because of the large increase in engine weight associated with the addition of a ceramic coating. Furthermore, since the coating must be sized to accommodate the increased heating rates that would be experienced during a turning maneuver or off-design conditions, it will operate less effectively than previously indicated because of its lower effective wall temperature at cruise conditions. The designer must therefore examine the use of thermal coatings very carefully to insure that the gain in performance justifies added development and maintenance costs.

Conclusions

The results of this study indicate that, with a regenerative system using superalloy heat exchangers and the hydrogen fuel as coolant, Mach 12 cruise at equivalence ratios less than one can be achieved. The increase in range and/or decrease in operating costs possible with T. D. nickel-chrome heat exchangers suggest their development. The development of refractory metal heat exchangers or ceramic coatings appears less urgent. Present and planned ground test facilities are considered adequate for initial determination of heat-transfer characteristics and heat exchanger development. However, to insure successful operation of the complete engine cooling system, flight test of a representative engine module is required.

Airfoil Pressure Distributions in Low-Speed Stall

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GOOD accuracy in the calculation of pressure distributions upon arbitrary, two-dimensional, unstalled airfoils satisfying a Kutta condition has been achieved.[1] Much less attention has been given to the estimation of pressures upon stalled airfoils, i.e., wing sections from which the flow has separated completely to form a broad wake that dominates the flowfield. In order to analyze the dynamics of stall recovery, it is necessary to have such pressure distributions in hand.

It has been observed that the near wakes of stalled airfoils tend to support only mild pressure gradients, in both the streamwise and lateral directions.[2] Large pressure gradients cannot exist unless there are substantial velocity gradients, and the separated region is characterized by slow and even recirculating velocities. These ideas are implicit in the many hodograph treatments (such as Ref. 3) that concern the inverse problem, i.e., finding the streamline pattern for a given pressure level.

The present Note describes a computerized method for estimating the pressure distribution on a stalled airfoil with an arbitrary known or assumed wake geometry. The method employs the approximation that pressures are transmitted undiminished from the outer boundary of the wake in the direction normal to the airfoil. This work complements the treatment of Ref. 1 for airfoils without separation.

To obtain the perfect fluid (potential) flow solution about the sheath defined by the airfoil and its wake, a Schwarz-Christoffel transformation of plane stagnation flow is used. The transformation is attractive in its generality, for it relates a straight line to any shape that can be approximated by a closed or open polygon.[4] Plane stagnation flow is appropriate because it features a streamline that splits at a stagnation point and travels to infinity along two separate paths, as does

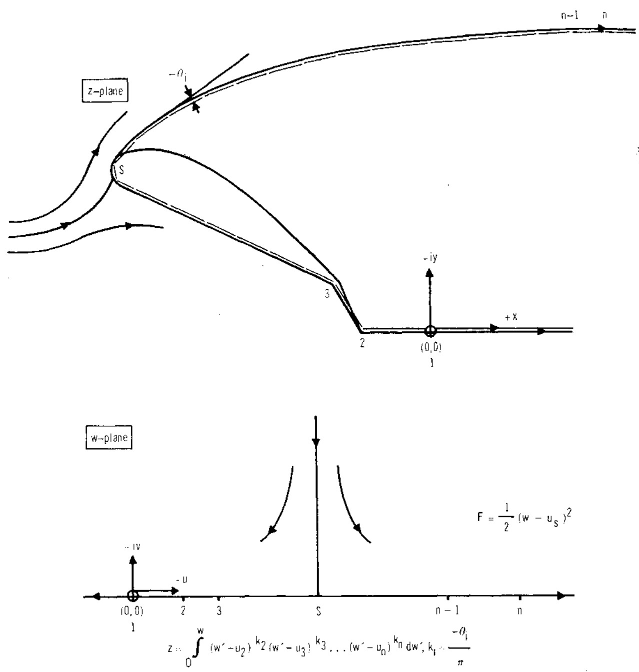

Fig. 1 The Schwarz-Christoffel transformation.

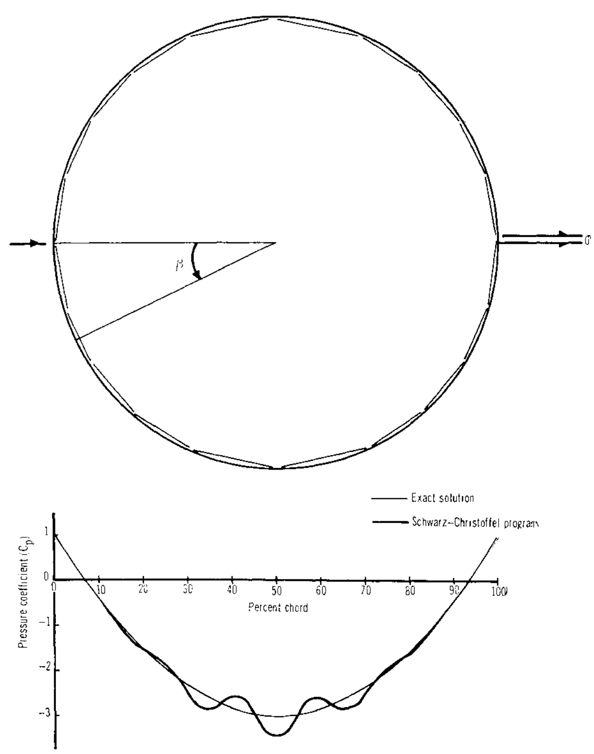

Fig. 2 Pressure distribution on a round cylinder with zero circulation.

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