DRACO Flowpath Performance & Environments

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BACKGROUND

The Advanced Space Transportation (AST) project office has challenged NASA to design, manufacture, ground-test and flight-test an axisymmetric, hydrocarbon-fueled, flight-weight, ejector-ramjet engine system testbed no later than 2005. To accomplish this, a multi-center NASA team has been assembled. The goal of this team, led by NASA-Marshall Space Flight Center (MSFC), is to develop propulsion technologies that demonstrate rocket and airbreathing combined-cycle operation (DRACO) [1]. Current technical activities include flowpath conceptual design, engine systems conceptual design, and feasibility studies investigating the integration and operation of the DRACO engine with a Lockheed D-21B drone [2]. This paper focuses on the activities of the Flowpath Systems Product Development Team (PDT), led by NASA-Glenn Research Center (GRC) and supported by NASA-MSFC and TechLand Research, Inc.

The objective of the Flowpath PDT at the start of the DRACO program was to establish a conceptual design of the flowpath aerodynamic lines, determine the preliminary performance, define the internal environments, and support the DRACO testbed concept feasibility studies. To accomplish these tasks, the PDT convened to establish a baseline flowpath concept. With the conceptual lines defined, cycle analysis tasks were planned and the flowpath performance and internal environments were defined. Additionally, sensitivity studies investigating the effects of inlet reference area, combustion performance, and combustor/nozzle materials selection were performed to support the Flowpath PDT design process. Results of these tasks are the emphasis of this paper and are intended to verify the feasibility of the DRACO flowpath and engine system as well as identify the primary technical challenges inherent in the flight-weight design of an advanced propulsion technology demonstration engine. Preliminary cycle performance decks were developed to support the testbed concept feasibility studies but are not discussed further in this paper. However, see reference [2] for an overview this activity.

DESIGN & ANALYSIS TOOLS

Performance analysis and internal environment definition were accomplished using several different modeling tools. This includes the MSFC ejector-ramjet performance analysis code (MERPAC), the NASA-Langley Research Center (LaRC) hypersonic ramjet performance code SRGULL, and the ramjet performance analysis (RJPA) code developed at the Johns Hopkins University's Applied Physics Laboratory (APL).

Ejector performance modeling was completed using the MERPAC. MERPAC is a greatly extended version of the enhanced Penn State University rocket-based combined-cycle (EPSURBCC) code. EPSURBCC was developed to calculate the ideal one-dimensional (1D) ejector performance of hydrogen/oxygen rocket-based combined-cycle (RBCC) engine that operate in a classical diffusion and afterburning (DAB) mode. In addition, the original code has the capability to predict aerodynamic choking in the constant area mixer. This phenomenon effectively limits the secondary mass flow and overall ejector performance. Over the last year, significant code modifications have been made to extend the capability beyond that of the original ideal cycle analysis. The primary extensions to EPSURBCC resulting in MERPAC are: 1) use of either the MIL-E-5007D inlet performance model [3] or user specified mass capture and total pressure ratio across the inlet; 2) direct integration of Chemical Equilibrium with Applications (CEA) source code [4] to handle mixing/combustion processes for arbitrary working fluids; 3) ability to model simultaneous mixing and combustion (SMC) cycles in non-constant area ducts with momentum losses;

and 4) algorithm changes enabling solution of thermal throat cases. The integration of CEA decreases code run time and allows the user to specify the fuel fraction burned at each flowpath station.

SRGULL was used for ramjet performance analysis, internal environment and heat load definition. SRGULL is an engineering model for assessing the tip-to-tail performance of hydrogen-fueled, airframe-integrated dual-mode ramjet's [5]. With funding support from NASA-MSFC, SRGULL was recently upgraded to utilize a generalized equilibrium chemistry package that allows a broader range of propellants and species to be considered [6]. See references 5 and 6 for further details on the analysis methods & processes used in SRGULL to determine cycle performance.

RJPA is a widely used 1D-cycle analysis tool for calculating hypersonic engine performance [7]. This code was used to conduct rapid turnaround ramjet cycle analysis and trades when insufficient design detail was available to utilize SRGULL. Modifications to the nozzle routine were required to enable the prediction of convergent-divergent nozzle performance.

FLOWPATH CONFIGURATION

DRACO is a hydrocarbon-fueled, axisymmetric, ejector-ramjet designed for airbreathing operation up to Mach 6 flight conditions. Although DRACO is an engine development program that aims to demonstrate technologies, the fact that the engine will be flight-tested must be considered in the conceptual design. Current plans are to integrate DRACO into the existing engine envelope of the D-21B. Utilizing the existing envelope will minimize the modifications and cost associated with the DRACO/D-21B integration effort. Thus, it should be noted that the DRACO flowpath was sized to fit the existing envelope vacated by the Marquardt RJ-43 engine used to power the D-21B reconnaissance drone.

The flowpath is designed to operate in three distinct modes: ejector to approximately Mach 3, ramjet from Mach 3-6, and rocket above Mach 6. DRACO's design features include a translating spike inlet, an ejector strut assembly with integrated fuel injection and flameholding provisions, a constant area mixer/combustor, and a variable geometry (VG) nozzle with a physical VG throat and a VG exit for altitude compensation.

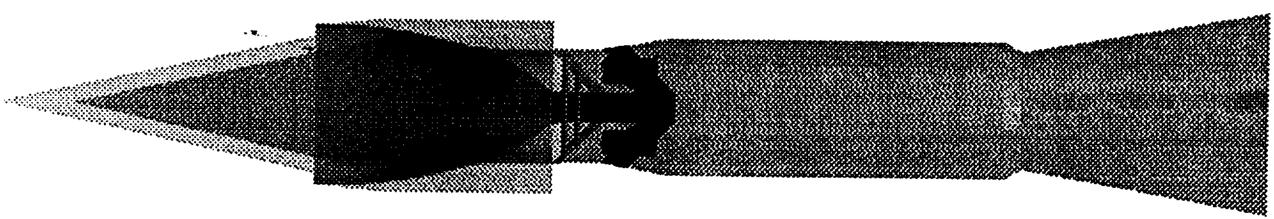

Figure 1. DRACO Flowpath Concept.

The Flowpath PDT baselined a Mach 6 shock-on-lip (SOL) inlet designed and tested in the NASA-GRC Trailblazer program [8]. This inlet configuration, designated 8a, has an extensive CFD performance database that served as the basis for the DRACO inlet operation and performance. This allowed the design team to get started early on the preliminary flowpath performance and environments definition while TechLand Research conducted trades on a new inlet concept with improved mass capture below the design point. TechLand has completed the conceptual design of a Mach 5 SOL inlet (designated TL1) with improved mass capture characteristics (calculated one-dimensionally) as compared to the 8a inlet. Presently, computational fluid dynamics (CFD) has not been completed to determine the pressure recovery, 3D mass capture, drag performance and operability characteristics of the inlet. However, preliminary cycle performance calculations were made for the DRACO engine using the TL1 inlet using the 1D mass capture efficiency and the 8a inlet pressure recovery schedules.

The ejector operates as an SMC cycle from thruster ignition through transition to ramjet. The fuel-rich primary rocket exhaust mixes & burns with the entrained secondary flow in a constant area mixer/combustor. Additional fuel is added to maximize engine thrust. Significant throttling of the primary rocket is expected to facilitate the transition to ramjet mode. In ramjet mode, fuel is injected upstream of the ejector/flameholder assembly and burned in the constant area combustor. The ramjet mode is operated stoichiometrically unless throttling the ramjet fuel supply is necessary to avoid premature thermal choking.

The baseline nozzle concept assumes a VG throat and exit. Feasibility studies are in work to investigate the mechanical and thermal design of the nozzle. As part of the performance analysis tasks, throat and exit area schedules were developed to define the degree of VG required by the nozzle. In addition, the thermal environments were defined to aid the design team in the nozzle design feasibility effort.

Three propellant combinations were originally considered for DRACO: JP-7/oxygen, JP-7/peroxide, and propane/oxygen. Since the performance level of the different hydrocarbon propellant combinations does not vary significantly, the primary design driver from the flowpath design point-of-view is the ability of the selected fuel to regeneratively cool the flowpath up to Mach 6 flight conditions. Thus, calculated flowpath environments and heat loads were used to determine the fuel heat sink requirement over a range of Mach numbers. In addition to flowpath propellant requirements (i.e. – high fuel heat sink capability), vehicle system, engine system, and subsystem/component considerations are being evaluated to form a consensus regarding propellant selection.

FLOWPATH PERFORMANCE

Baseline Ejector Performance

The following assumptions were used in the ejector cycle performance calculations:

- Mach 0-1.8 the inlet is unstarted
    - 8a inlet pressure recovery
    - limit secondary flow Mach number to 0.8
- Mach 1.8-3.5 the inlet is started
    - 8a inlet pressure recovery
    - JTL1 inlet mass capture
- JP-7/peroxide (90% H_2O_2/10% H_2O) primary rocket propellants at a mixture ratio of 7
- SMC operating cycle with makeup fuel added to yield overall stoichiometric engine
- 90% stream thrust (momentum) efficiency in mixer/combustor
- 95% primary fuel reacts in thruster; 100% primary fuel reacts in mixer/combustor
- 90% makeup fuel reacts in mixer/combustor
- Nozzle throat area ratio ($A8/A8_{max}$) optimized for maximum thrust
- Nozzle exit area ratio ($A9/A9_{max}$) optimum or under-expanded
    - maximum exit area limited to existing D-21B envelope

Ejector performance maps were developed for all flight conditions from Mach 0-3.5 and altitudes from 0-100,000 feet. In addition, the matrix of ejector mode flight conditions was completed for four different chamber pressures to quantify the effect of throttle on performance.

Figures 2-3 illustrates the ejector thrust and specific impulse trends for the maximum throttle case only. Lines of constant dynamic pressure are included to illustrate the performance trends following a constant dynamic pressure flight profile. Figures 4-5 show the calculated non-dimensional nozzle throat and exit area variations for the full throttle case. The maximum nozzle throat and exit areas are based on the existing D-21B engine envelope. Figure 5 clearly shows the nozzle exit area becomes under-expanded around Mach 1.5 for flight at around 500 pounds per square foot (psf).

Figures 6-7 show the effect of throttling the primary rocket at each representative flight condition. These conditions were chosen to represent a linear 0-650 psf dynamic pressure increase from Mach 0-2. Above Mach 2 the altitudes were chosen to coincide with a 650 psf constant dynamic pressure flight.

Baseline Ramjet Performance

Ramjet performance was calculated for both the 8a and TL1 inlets. All cases using the 8a inlet were run using SRGULL. For the cases using the TL1 inlet, RJPA was used with the TL1 mass capture efficiency along with the 8a inlet pressure recovery. Figure 8 shows the inlet performance parameters as a function of Mach number for the 8a and TL1 inlets. Reference area (A_{ref}) was based on the existing D-21B inlet envelope for both inlets.

Ramjet performance and environments were calculated at 650 psf dynamic pressure for Mach 2.5, 3, 4, 5 and 6 flight. For each case octane (C_8H_{18}) fuel was injected and the engine was operated stoichiometrically with a 90% combustion heat release. For the SRGULL cases, this heat release was distributed between the fuel injection location and the end of the constant area combustor. In addition, the combustor and nozzle walls were cooled to 1000 degrees Fahrenheit (degF) while the inlet and ejector strut assembly maintained an adiabatic wall temperature (uncooled).

Figure 9 shows clearly the improvement in engine thrust coefficient due to the increased mass capture efficiency of the TL1 inlet. However, the Mach 2.5 case with the TL1 inlet required a reduction in equivalence ratio to avoid premature thermal choking in the combustor. Figure 10 qualitatively shows the specific impulse reduction of the flowpath using the TL1 inlet. This reduction in efficiency is due to the increased mass flow through the engine, which increases the Mach number, thus, the Rayleigh losses, throughout the engine. Figure 11 illustrates the difference in nozzle throat area variation between the two inlet configurations.

SRGULL cases with the 8a inlet were used to determine the preliminary internal environments. Figure 12 and 13 show the static pressure and temperature distributions from the inlet throat to a location downstream of the nozzle throat for the range of Mach number cases run.

Ramjet Performance Sensitivity

Sensitivity of combustor heat release and inlet reference area on flowpath thrust performance was calculated using the 8a inlet model in SRGULL. The baseline case (100% A_{ref}, 90% heat release) is shown in Figure 14 for comparison to cases where the heat release is increased to 99% (case 1), the reference area is increased to 113% (case 2), and both the heat release and reference area are increased (case 3). Up to an 18% increase in thrust performance can be attained with the larger reference area and the increased combustion performance, with heat release accounting for 6% improvement and reference area accounting for 12% improvement. However, at Mach 2.5 the 113% reference area flowpath could not operate stoichiometrically without prematurely choking in the combustor.

The combustor and nozzle design wall temperature was varied to quantify the heat sink required by the fuel to actively cool the combustor and nozzle. The inlet and ejector assembly is assumed uncooled and was assumed to have an adiabatic wall. Combustor and nozzle wall temperatures were varied from 1000-3000 degF to represent the range of potential materials under consideration for the flowpath flight-weight design. SRGULL was run over the range of ramjet Mach numbers for each wall temperature and the combustor and nozzle heat flux was integrated to determine the total heat load. The ratio of the heat load to fuel flow rate represents the total fuel heat sink required to carry away the combustor and nozzle heat. Air-cooling was not considered in this sensitivity study. Results of this sensitivity are shown in figure 15. As shown in the figure, utilization of high temperature materials can significantly reduce the heat sink required from the fuel.

FUTURE WORK

Near-term plans for the DRACO program and Flowpath Systems PDT include the following: 1) baseline an appropriate propellant combination; 2) quantify the performance of the next TechLand inlet design iteration which aims to provide increased mass flow at low speed; 3) establish a feasible nozzle design that can satisfy the mechanical (variable geometry) and thermal (heat load) design requirements; and, 4) perform ejector thrust, ramjet equivalence ratio, and nozzle exit area trades and sensitivities.

SUMMARY

An overview of the performance and internal environments of the baseline DRACO flowpath was presented. Ramjet cycle performance trends were shown illustrating the inlet design improvements to date. Determination of the baseline performance and environments indicates that the DRACO flowpath requires additional definition to determine its aerodynamic and mechanical design feasibility. However, results from the present study will support further aerodynamic, mechanical and thermal design efforts. Additionally, the baseline performance data will be used to support on-going D-21B testbed feasibility efforts that will further define the technology demonstration flight scenarios and testbed design requirements.

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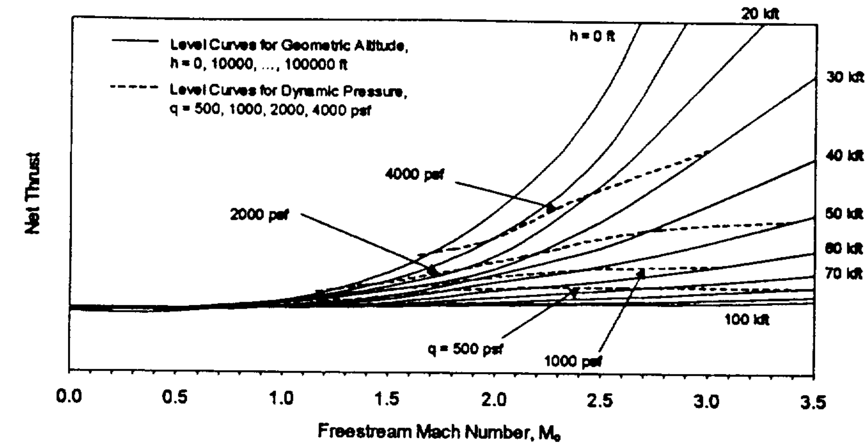

Figure 2. DRACO net engine thrust variation with Mach Number and altitude

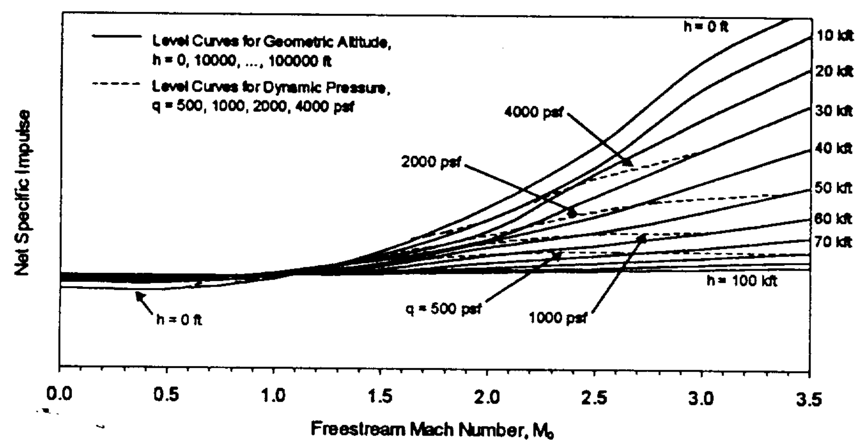

Figure 3. DRACO net specific impulse variation with Mach Number and altitude

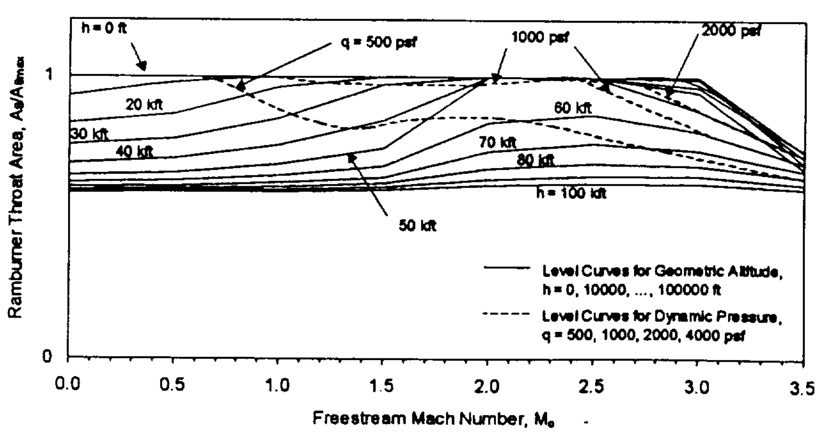

Figure 4. DRACO normalized ramjet throat area variation with Mach Number and altitude

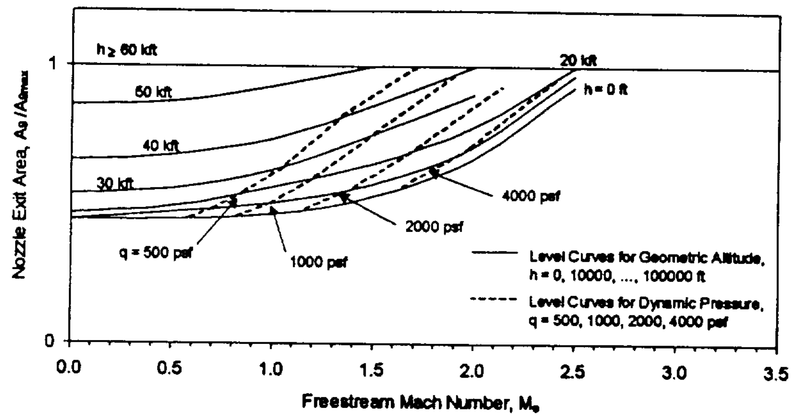

Figure 5. DRACO normalized nozzle exit area variation with Mach Number and altitude

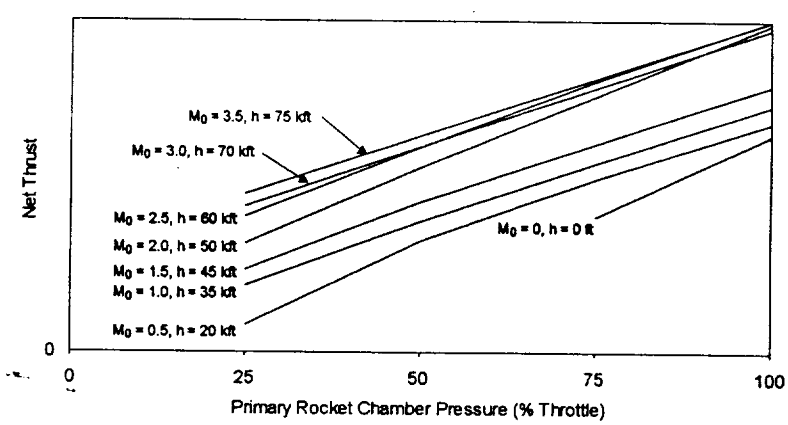

Figure 6. DRACO net thrust variation with primary chamber pressure

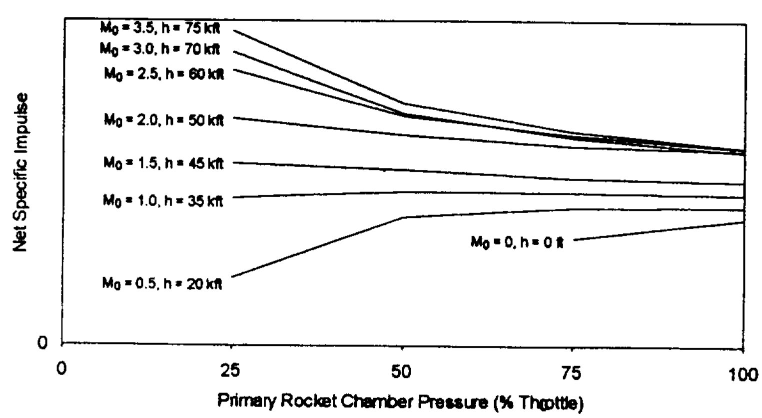

Figure 7. DRACO net specific impulse variation with primary chamber pressure

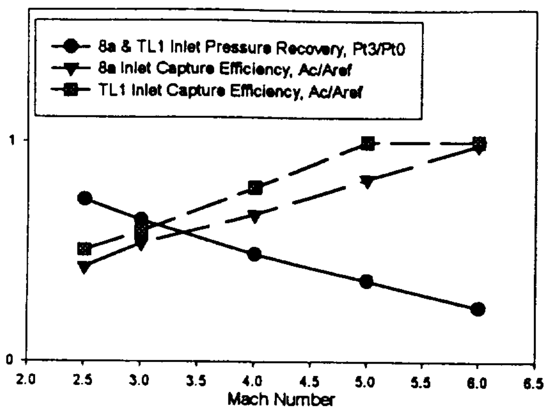

Figure 8. DRACO inlet pressure recovery and mass capture efficiency

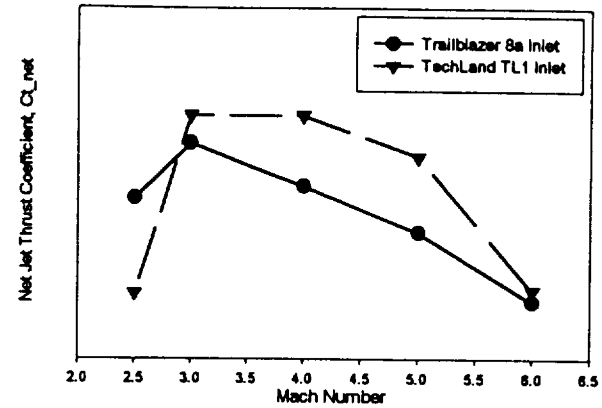

Figure 9. DRACO net jet thrust coefficient

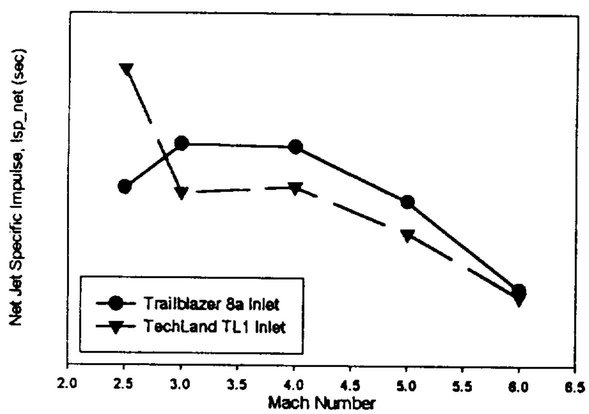

Figure 10. DRACO net jet specific impulse

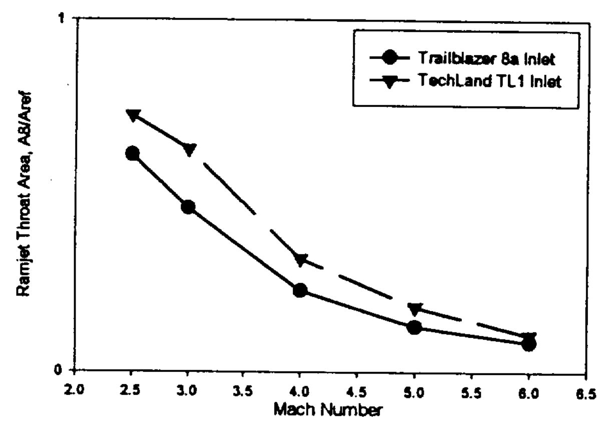

Figure 11. DRACO ramjet throat area ratio

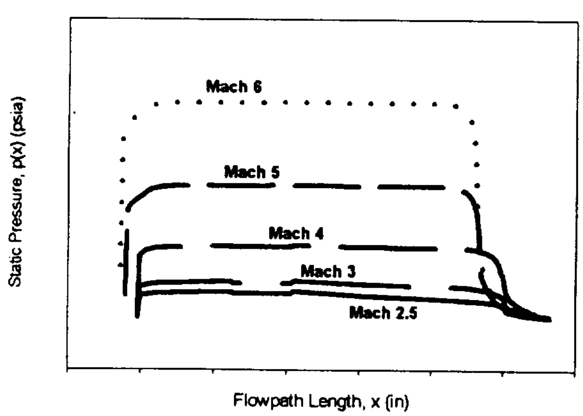

Figure 12. DRACO diffuser/combustor static pressure distribution

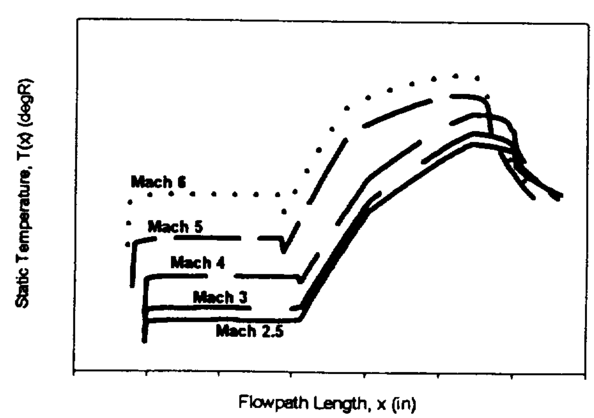

Figure 13. DRACO diffuser/combustor static temperature distribution

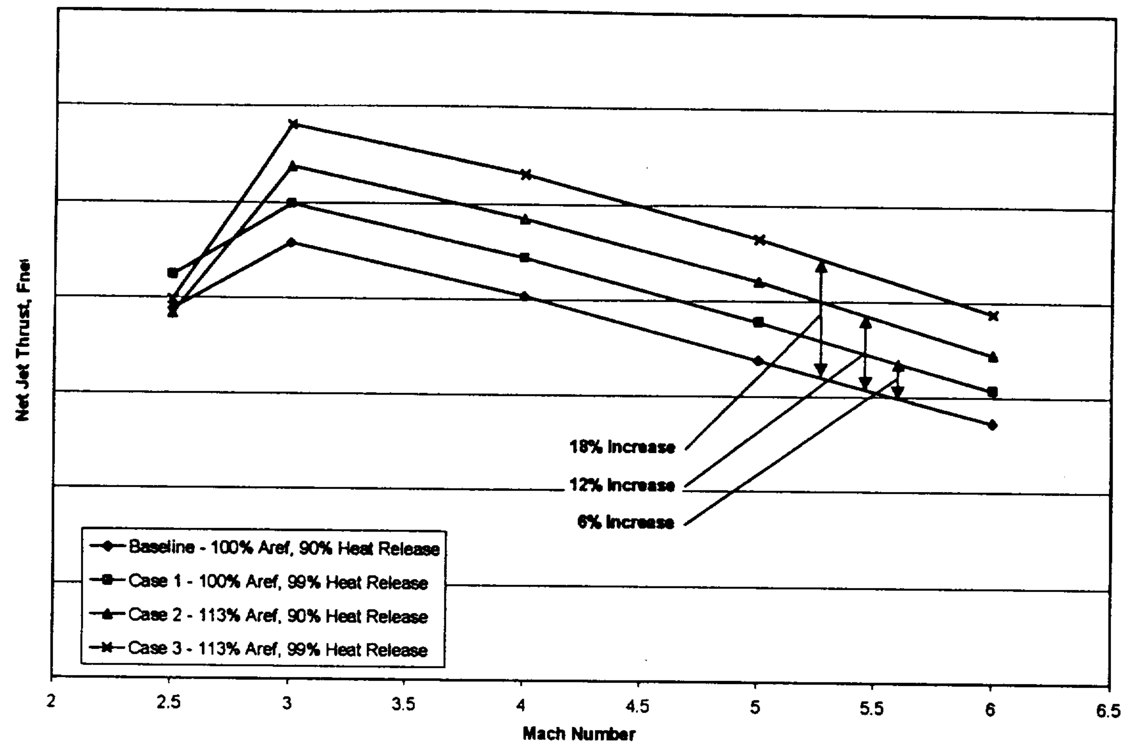

Figure 14. Ramjet thrust sensitivity for variations in inlet reference area and combustor heat release.

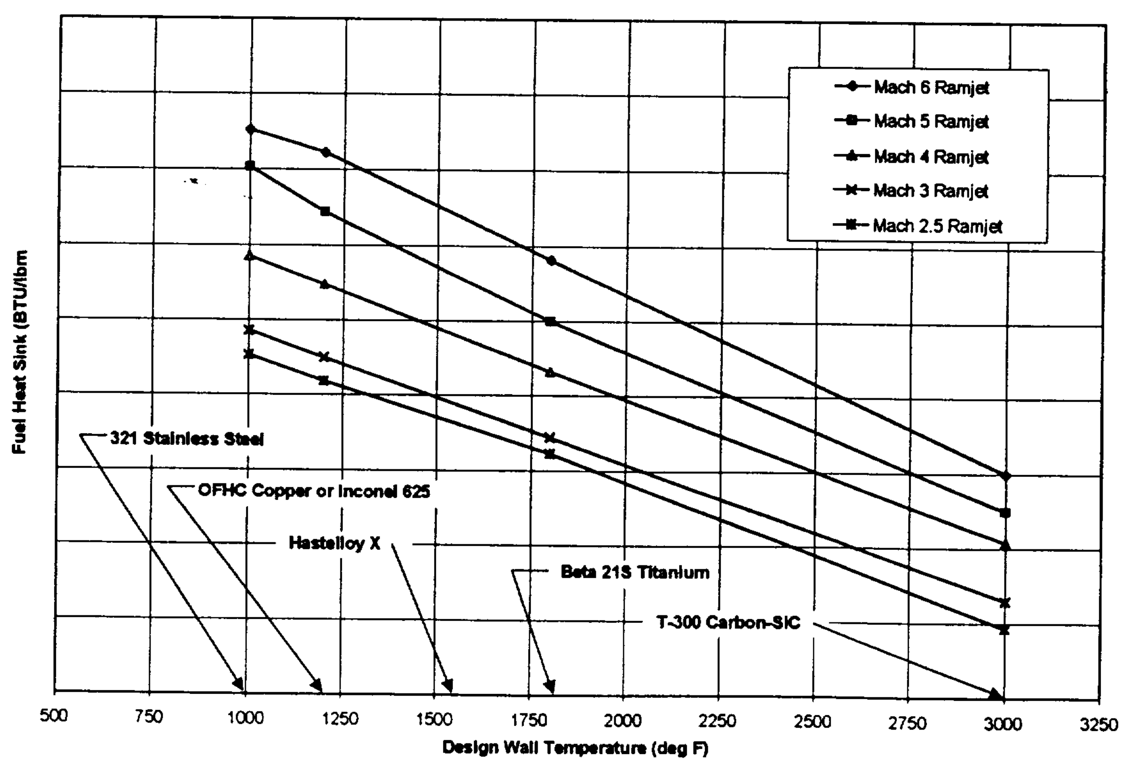

Figure 15. Fuel heat sink requirement for various constant design wall temperatures and Mach numbers.