

The NASA Turbulent Heat Flux Experiments: Summary and Lessons Learned

Nicholas J. Georgiadis,¹ Mark P. Wernet,² and Michael R. Borghi³
NASA Glenn Research Center, Cleveland, OH 44135

Chad M. Winkler⁴
The Boeing Company, St. Louis, MO 63042

Michael A. Emory⁵
Luminary Cloud, San Mateo, CA 94401

Randy J. Locke⁶
HX5 LLC, Cleveland, OH, 44135

The Turbulent Heat Flux (THX) experiments were conducted at NASA Glenn Research Center (GRC) in order to collect measurements of velocities and temperatures for computational fluid dynamics (CFD) validation of heated flows, with a focus on propulsion system components. The experiments spanned 5 phases; four of which were conducted in the GRC AeroAcoustic Propulsion Laboratory (AAPL) using the Small Hot Jet Flow Rig (SHJAR). In addition to making velocity measurements with Particle Image Velocimetry (PIV), the THX experiments introduced a new Raman-scattering based capability to measure temperatures. Computational studies were also conducted for each of the experimental configurations, in order to provide a baseline of expected CFD results and conduct an assessment of the capability of various CFD approaches for calculating flows where the turbulent transport of heat was important. Two of the collected sets of data were used for American Institute of Aeronautics and Astronautics (AIAA) Propulsion Aerodynamic Workshops (PAWs). The data set from the 5th phase, collected for heated supersonic jets, was also used to construct new validation cases for the NASA Turbulence Model Resource (TMR). This paper provides an overview of the experiments and associated computations for each of the 5 test phases. Key experimental findings are presented. Lessons learned are provided concerning the effect of computational modeling choice on accuracy of predicting turbulent flows where thermal transport is important. Emphasis is placed on comparing Reynolds-averaged Navier-Stokes approaches with large-eddy simulation approaches. The benefits of utilizing a conjugate heat transfer method in conjunction with CFD solver for film cooling is demonstrated.

Nomenclature

BR	blowing ratio for film cooling configurations = $(\rho_{inj} U_{inj})/(\rho_j U_j)$
D	injector hole diameter for Phases 1 and 2, nozzle exit diameter for Phase 5.
k	turbulent kinetic energy
M	Mach number
Ma	acoustic Mach number = jet velocity / ambient speed of sound
T	mean static temperature
T', T _{rms}	rms of static temperature

¹ Aerospace Engineer, Inlets and Nozzles Branch, Georgiadis@nasa.gov, AIAA Associate Fellow

² Senior Research Engineer, Optics and Photonics Branch, Mark.P.Wernet@nasa.gov, AIAA Associate Fellow

³ Aerospace Engineer, Turbomachinery and Turboelectric Systems Branch, Michael.R.Borghi@nasa.gov, AIAA Member

⁴ Boeing Technical Fellow, Chad.M.Winkler@boeing.com, AIAA Associate Fellow

⁵ Product Manager, memory@luminarycloud.com, AIAA member.

⁶ Research Engineer, retired, rjlocke66@gmail.com.

T^*	nondimensional static temperature = $(T - T_\infty) / (T_j - T_\infty)$
T_j, T_{jet}	temperature at jet exit for all phases
T_∞, T_{amb}	ambient static temperature
u	mean axial velocity
u', u_{rms}	rms of axial velocity
$u'T'$	turbulent heat flux
$u'v'$	turbulent shear stress
U_j, U_{jet}	axial velocity of jet at nozzle exit
U_{inj}	area-averaged velocity magnitude of coolant stream
x	axial distance from nozzle exit
Y	vertical distance from plate surface
ρ_j	gas density of jet at nozzle jet exit
ρ_{inj}	gas density of the injected coolant stream
ΔT	difference between jet exit temperature and ambient temperature = $(T_j - T_\infty)$

Introduction

Flow turbulence has been recognized by the aerospace research and development (R & D) community as one of the most important factors affecting aerospace system performance and one of the most difficult physical phenomena to understand and predict. Computational fluid dynamics (CFD) technologies have become widely used in the R & D community, and while there have been continual advances in computing capability, the ability to simulate complex aerospace flows is limited by the ability to replicate the effects of turbulence. A primary shortcoming has been the lack of validation data, particularly for complex geometries, with sufficient details needed to understand the effects of turbulence and identify trends that can result in improved computational models for turbulence. These details include measurements of important mean flow quantities and turbulent statistics with sufficient spatial resolution, but also enough geometry and flow condition information (i.e. boundary conditions) to adequately model the flow problems of interest. Slotnick et al.¹ concluded that improving techniques for computing turbulent flows remains of paramount importance, and that there is a need for experimental testing programs that are planned with computational turbulence modeling development and validation activities in mind.

In recent years, research into improving turbulence modeling has been sponsored by NASA's Transformational Tools and Technologies (TTT) project of NASA's Transformative Aeronautics Concepts Program (TACP). This research has consisted not only of advanced computational method development and validation, but also fundamental experiments with detailed measurements needed to make computational modeling assessments. Historically, work into turbulence modeling for problems where turbulent heat flux is important has been rather limited. Turbulent temperature variations and heat flux are prevalent in high Mach number applications, propulsion systems, and systems that require active cooling, just to name a few. There had been a lack of experimental data to improve and validate such models, which is primarily due to the difficulty of obtaining detailed time-resolved measurements of temperature where there is turbulent transport of heat. As discussed in Refs. 2-4, standard Reynolds-averaged Navier-Stokes (RANS) based techniques perform poorly when considering prediction of turbulent heat flux and temperatures in propulsion flows involving turbulent shear layers. Scale-resolving simulation (SRS) methods, utilizing large-eddy simulation (LES) for all or a portion of the flow, have begun to be used for more complex problems, but are still limited by the computational resources required for such calculations. The standard practice for computational analysis of heated propulsion flow components had been to use two-equation RANS models for the in-flow aerodynamics and thermodynamics, including constant turbulent Prandtl number modeling for the turbulent heat flux and then employ relatively simple boundary conditions for wall surface temperatures or heat flux.

To address the shortfall of experimental data available to improve and validate turbulence modeling for flows where turbulent thermal transport is important, the TTT project sponsored a series of test campaigns referred to as the Turbulent Heat Flux (THX) experiments. The THX experiments consisted of five separate phases and focused on measurement flow properties related to the turbulent transport of both momentum and heat. Particle Image Velocimetry (PIV), which has been routinely used for a number of years, was employed to make measurement of mean and rms velocities. Mean and rms temperature measurements were made using a rotationally-resolved Raman spectroscopy technique introduced in the second phase and used through Phase 5.

Computational studies were also performed for each of the experimental configurations, in order to provide a baseline of expected CFD results and conduct an assessment of the capability of various CFD approaches for calculating flows where the turbulent transport of heat was important. In nearly every case, both RANS and LES were examined. Two of the phases provided data that were utilized in AIAA Propulsion Aerodynamic Workshops (PAWs).

The following sections provide the primary experimental findings and comparisons with computations for each phase, lessons learned, and recommended areas of future research towards advancing turbulence modeling for high-energy, thermally-dominated flows.

THX Experimental Phases

1. Low-speed and low-temperature film cooling

The first phase of the THX experiments were low-speed / low-temperature cooling hole investigations in the NASA Glenn Research Center (GRC) Fundamental Film Cooling and Heat Transfer Facility (SW6). Figure 1 shows a schematic of the facility with the primary air inflow and configuration with three injector flows. For testing convenience in these fundamental flow measurements, the film cooling configuration is inverted. The injected flow is heated in the SW6 configuration, and the primary air is at room temperature, which is in contrast to a typical film-cooling arrangements in practice where the injected flow is at lower temperature and designed to provide a layer of air colder than the primary stream to cover and protect the surface, whether it be a turbine blade, engine combustor, or nozzle afterbody. Hot-wire anemometry and PIV were used to make measurements as documented in Refs. 5 and 6. This was the one phase of the THX experiments where the actual heat flux quantity (or $u'T'$) was measured. Compared with the other four phases, the Phase 1 experiments were conducted with much lower velocities and temperatures. This makes this set of experiments probably the easiest to examine with SRS techniques and to attempt to resolve nearly all of the turbulent spectrum.

Borghini et al.^{7,8} performed calculations of the configuration with tunnel flow set to 9.1 m/s, blowing ratio (BR) of 1.2, and temperature difference (ΔT) of 40 K between the tunnel and injector flows. The calculations included RANS, hybrid RANS-LES, and wall-resolved LES (WRLES). An instantaneous snapshot of the flow computed from a WRLES computation is shown in Fig. 2. Only the WRLES computation was able to correctly replicate the fluid physics of the inclined jet-in-crossflow problem. RANS calculations significantly overpredicted the length of the jet penetration and the hybrid RANS-LES computations exhibited a significant delay in the turbulent shear layer developing for the inclined jet-in-crossflow evolving from a fully-modeled state (the RANS limit) to resolved turbulent state (the LES limit). Hybrid RANS-LES has offered improved predictions over RANS for many aerodynamic problems involving large scale separations, and related problems where RANS and LES regions are relatively isolated and where there is not a risk of an important part of the flow field that is neither being largely resolved (i.e. LES region) not fully modeled (i.e. RANS region) but something in between. For problems like this inclined jet-in-cross flow where such ambiguous regions arise, which is sometimes called a “model stress depletion” zone, this interface between RANS and LES regions may contaminate the fully RANS and LES regions away from the interface and adversely affect the quality and accuracy of the simulation. A comparison of experimental to WRLES computed rms of temperature is provided in Fig. 3. In this figure, note that the projection of the centerline of the hole extends from $x/D = 0$ to $x/D = 2$ because the cooling tube, inclined at 30 degrees from the plate surface results in a projection of the hole centerline to a length of 2D. The experimental results showed that periodicity could be assumed with reason and the computations only computed a single cooling hole with periodic boundary conditions in the spanwise direction. The results showed peak mean axial velocities and normalized temperatures to be within 5 percent agreement between the measurements and computations, while the rms of both velocities and temperatures to be within 15 percent agreement.

2. Subsonic convergent nozzles

Phases 2-5 were all conducted in the GRC Aeroacoustic Propulsion Laboratory (AAPL) using the Small Hot Jet Acoustic Rig⁹⁻¹⁰ (SHJAR). A schematic of the SHJAR with a nozzle connected to the heated flow supply line and a large traversing mechanism used to enable non-intrusive flow measurements, referred to as “Big Blue,” is shown in Fig. 4. For subsonic jets exhausting from round convergent nozzles, a broadly used set of PIV data was collected by Bridges and Wernet¹¹⁻¹² several years prior to the THX effort. Jet velocity data were collected for operating points at several Mach number and temperature settings. PIV data for three of these operating points had been used to create subsonic jet validation cases for the NASA Turbulence Modeling Resource (TMR).¹³ The data sets consisting of mean velocities and turbulent statistics were augmented in THX Phase 2 with temperature data taken with a new rotationally resolved Raman spectroscopy technique developed by Locke et al.^{14,15} Rotationally resolved Raman scattering is the only technique that is relatively simple to setup/align and robust against the hostile environments found in facilities like the SHJAR; therefore it was the technique of choice for measuring the gas temperatures and densities in the THX series of experiments. Raman spectroscopy is a point-based technique, requiring 1000 single-shot spectra to be acquired at each measurement location to enable statistical convergence. The same nozzle and many of the same

operating points from Refs. 11-12 were utilized to obtain measurements of mean and rms temperatures in the jet plumes. Multiple set points were investigated for convergent nozzles with jet exit Mach numbers up to nearly sonic and jet exit temperature up to 800 K with maximum ratio of jet exit temperature to ambient temperature, $T_{\text{jet}} / T_{\text{amb}} = 2.7$.

Figure 5 shows centerline velocities and temperatures, including both mean and rms quantities, for two of the set points investigated in phase 2. Both setpoints were for jets having acoustic Mach numbers, defined as the jet speed divided by the ambient speed of sound corresponding to $Ma = 0.9$. Set point 27 had the jet heated such that the ratio of the jet exit temperature to ambient temperature, $T_{\text{jet}} / T_{\text{amb}} = 1.765$, while set point 46 had $T_{\text{jet}} / T_{\text{amb}} = 2.7$. For both of these cases, the decay of mean centerline velocity and temperature difference $(T - T_{\text{amb}}) / (T_{\text{jet}} - T_{\text{amb}})$ were quite similar. The behavior of the rms of these quantities, however, is quite different, with the rms of temperature decaying much more quickly than velocity. DeBonis¹⁶ obtained LES calculations of select subsonic test points from consensus data and compared computed solutions to the previously collected PIV data and the temperature data collected during the second phase of the THX experiments. An older data set obtained by Mielke et al.¹⁷ using the Rayleigh measurement technique to obtain temperatures but also collected in the GRC SHJAR, was also used in DeBonis's comparisons. The LES demonstrated the same trends as the experimental data for the mean and rms values of both velocity and temperature.

3. Single Cooling Hole Experiments

PIV measurements for velocities and Raman-based temperature measurements were made for the fundamental configuration of a square nozzle producing heated subsonic flow over a plate with a single large injector cooling flow in Phase 3 of the THX experiments, as described in Wernet et al.¹⁸. The experimental configuration is shown in Fig. 6. The convergent nozzle, with a square exit, transitioned from the upstream round pipe that joined the SHJAR supply pipe, via a series of super ellipses. The single cooling hole, with a 12.7 mm diameter, was inclined at 30 degrees to the primary stream direction, and had an length to diameter ratio of 40 to enable a fully developed pipe flow to exit at the plate surface and hence facilitate computational boundary condition settings for CFD validation efforts using this data. Standard two-component PIV measurements were taken through the center-plane of the nozzle, also passing through the centerline of the injector nozzle. Three-component stereo PIV (SPIV) planar measurements were obtained at several planes downstream of the nozzle exit. Surface temperatures on the plate were obtained with thermocouples. The test matrix consisted of several nozzle operating points with nozzle exit Mach numbers varying from 0.376 to 0.9 and nozzle exit static temperature ratios of 1.765 and 2.7. While the geometric configuration of the Phase 1 and Phase 3 experiments are similar, the Phase 3 experiments examined higher speeds and temperatures than those of Phase 1. This makes the Phase 3 experimental flow environment more characteristic of actual propulsion systems, but more challenging for computational modeling due to the higher Reynolds numbers and significantly larger computational grids required to resolve most of the turbulent energy spectrum. For the cooling hole, blowing ratios (BR) from 0.5 to 2.0 were examined.

Computational simulations were obtained with both wall-resolved LES (WRLES) and RANS calculations as documented in Refs. 19-22. The two LES approaches used very different algorithms. The FDL3DI solver²³ is based upon a high-order compact scheme designed for structured grids. The Glenn Flux Reconstruction (GFR) code²⁴ also employs a high-order method, but it is the Flux Reconstruction scheme, designed for unstructured methods. For comparison with the two LES results, a RANS result using a Reynolds-stress model in the FUN3D code was chosen from those studied in Ref. 21, as this RANS approach enabled better prediction of three-dimensional turbulent flow structures than linear one- and two-equation turbulence models.

Figure 7 provides instantaneous axial velocity contours from WRLES calculations using the FDL3DI code for cases with the nozzle exit Mach number = 0.376, jet exit static temperature ratio of 1.765, and BRs of 1 and 2. The BR=1 flow enables more of an attached cooling film while the BR=2 flow results in more of a detached inclined jet-in-crossflow situation. A comparison of temperatures from the experiment with the two LES approaches and the Reynolds-stress model RANS approach is provided in Fig. 8. It may be observed that the two LES approaches agree well with each other and better with experiment than the RANS approach. The RANS computed flow results in a cooling jet that does not mix as quickly as indicated by experiment nor the LES approaches. In Borghi et al.²², there are many detailed comparisons that show the superiority of LES over RANS for prediction of this canonical film cooling experiment. Very notably, the prediction of the actual turbulent heat flux quantities ($u'T'$ and $v'T'$) in the energy equation was found to be much more directionally dependent in the RANS computations than that directly calculated with LES. Beyond the shortfall of the RANS in capturing the jet-in-crossflow penetration, this deficiency in accurately calculating the turbulent heat flux vector is likely due to the gradient-diffusion hypothesis used in calculating the heat flux in the RANS approach.

4. Film-Cooling Hole Experiments

The same nozzle from Phase 3 of the THX experiments was used in Phase 4. However, instead of the simple plate with single cooling hole, a dual plenum film cooling system was designed and attached to the nozzle in order to provide an arrangement with heated exhaust over a film-cooled plate, as shown in Fig. 9. The plate was cooled with three sections of film-cooling patches each having 45 holes inclined at 30° to the heated primary stream direction. These holes were 1 mm in diameter, much smaller than the single cooling hole of the Phase 3 experiments, and more realistic in terms of what was required to produce a cooling film over the plate. Experimental techniques similar to those of Phase 3 were employed for the Phase 4 experiments, namely PIV measurements for velocities and Raman-based measurements for in-flow temperatures. A background-oriented Schlieren (BOS) measurement system²⁵ was used during initial tests to identify the thickness of the cooling film over the plate for various flow conditions, and in turn was used to define and optimize the spatial matrix where the individual point Raman temperature measurements should be acquired. Surface infrared (IR) temperature measurements were also acquired during the Phase 4 tests. The experimental measurements are documented in Refs. 26-27. Flow conditions for these experiments had the nozzle jet exit Mach number varying from 0.3 to 0.9, $T_{\text{jet}} / T_{\text{amb}}$ from 1.765 to 2.7, and BR from 0 to 3.

Similar to the other phases, computational efforts accompanied the experimental measurements for the THX – Phase 4 configuration. The American Institute of Aeronautics and Astronautics (AIAA) 5th Propulsion Aerodynamic Workshop (PAW5) used this configuration as the nozzle problem. One of the key areas of investigation was determining whether CFD computations with a conjugate heat transfer (CHT) approach were required to obtain accurate calculations of surface temperatures. Figure 10 shows a typical comparison of a RANS approach applied to the configuration with and without fully three-dimensional CHT and clearly shows that the combined CFD-CHT approach is required to provide accurate calculations of film-cooled surface temperatures. More simple approaches to the conductive heat transfer problem of the solid surface, including one-dimensional conductive heat transfer models, resulted in less accurate surface temperature predictions.

Several individual computational investigations from the AIAA PAW5 are presented in Refs. 28-32 and a summary of all the results is presented in Ref. 33. Participants examined RANS and SRS approaches to the problem with and without CHT. Because of the geometric complexity of the problem with 135 cooling holes, there were no approaches attempted for the workshop that attempted to resolve the near wall details (i.e. WRLES) of the core flow boundary layer interacting with the individual cooling holes. Figure 11 shows instantaneous contours of velocity and temperature over the plate as reported in Ref. 28, where HLRES was employed with CHT. However, WMLES and HRLES were attempted with some notable improvements over RANS approaches. Overall, the WMLES of Emory et al.³⁰ provided the most accurate plate surface temperatures of any presented during the workshop. In Fig. 12, a comparison of measured and computed thermal boundary layers is made at 270 mm from the nozzle exit, which is on the overhang of the plate shown in Fig. 9, downstream of the last patch of cooling holes. The RANS solutions as a whole did not replicate the entirety of the thermal boundary layers nearly as well as the scale-resolving (LES-based) solutions did. In addition to the inferior prediction of the interaction of the cooling flow jets with the heated flow, the Reynolds analogy used in the RANS approach to relate thermal transport to momentum transport, with standard gradient diffusion approximation and a simplistic constant turbulent Prandtl number setting, is not sufficient to yield accurate results.

5. Supersonic Jet Mixing

The same PIV and Raman measurement techniques used in Phases 2-4 were used to obtain detailed aerodynamic and thermodynamic data for turbulent supersonic jets issuing from round convergent-divergent nozzles having divergent section coordinates designed to be isentropic (i.e. shock-free) using the method of characteristics. The temperature measurement diagnostic was extended in this phase to also extract density data from the acquired Raman spectra, as described in more detail by Wernet et al.^{35,36} This report only considers temperature measurements from the Raman spectra.

Operating points from the work of Seiner et al.³⁴ for heated Mach 2 jets were used as a starting point for the matrix of cases examined here using a geometrically similar Mach 2 nozzle. Figure 13 shows one of the supersonic nozzles installed in the GRC SHJAR facility. Two additional nozzles were examined having design Mach numbers of 1.36 and 1.63. For each Mach number, multiple jet temperatures were investigated with greatest difference between the jet exit and ambient static temperatures, $\Delta T = T_{\text{jet}} - T_{\text{amb}} = 333$ K for one of the Mach 1.36 cases. This ΔT was not achievable for the higher Mach number nozzles. Jet measurements were first made with stagnation pressure and temperature probes for comparison to the previous experiments. Very close agreement with the previous data of Seiner et al.³⁴ was established when comparing the older data set's probe measurements, the THX5 probe measurements, and the nonintrusive measurements enabled by PIV and Raman techniques in THX5.

Experimental results, shown in Fig. 14 for both a temperature-matched jet and a jet that exited the nozzle 233K higher than ambient, showed that as jet Mach number increases, the overall jet mixing rate decreases, as indicated by the potential core length, downstream mean velocities, and turbulence levels. Similar behavior was observed in the mean temperatures and rms temperatures, as shown in Fig. 15. The suppression of turbulence as the jet Mach number increases is due to compressibility. It is emphasized that the jets termed “temperature-matched” with $\Delta T = 0$ K, does not imply that the jets were unheated but rather that the SHJAR supply was heated such that when the flow expanded to the supersonic nozzle exit state, the static temperature matched the static temperature of the ambient. For a fixed jet Mach number, increasing the jet temperature resulted in faster mixing by all measures as may be observed for the Mach 1.36 jets in Fig. 16. Peak turbulent stress levels, located along the lip line of the jet, demonstrated trends in individual stress components that differed depending on whether jet Mach number or temperature was varied, but indicated that the turbulence structure parameter ($u'v' / k$) was very similar when varying either jet temperature or Mach number. The experimental results are presented in detail in Refs. 35-38.

The THX5 supersonic jet data set was used as the nozzle case in the 6th AIAA PAW. Similar to the 5th PAW, a variety of RANS and LES approaches were attempted for the supersonic jet cases. Figure 17 shows a typical result obtained with two-equation RANS for subsonic and supersonic jets. The supersonic jet experimental data was collected in THX5, while the subsonic data was collected prior to the THX experiments, and documented in Refs. 11-12. All of the jet data was collected in the same GRC SHJAR facility with nozzles having the same exit diameter, and using the same measurement techniques. Therefore, comparing these subsonic and supersonic jet data sets, albeit collected many years apart, should enable direct comparison concentrating on fluid physics differences. In Fig. 17, examining only the experimental data while first holding the jet exit temperature fixed, suppression of turbulence due to compressibility results in the supersonic jets mixing more slowly than low subsonic jets, as is well understood. When holding jet exit Mach number nearly fixed, it is observed that a heated jet will mix more quickly than an unheated (or more correctly termed temperature matched case for the supersonic jets) jet case. Standard RANS approaches have no capability to capture the effects of compressibility in even simple round jets like these, while at least the trends due to temperature effects are captured to some extent.

In the workshop, some of the participants investigated RANS with compressibility corrections. These corrections have been available for approximately 3 decades, and while in some cases they are able to replicate the trends of reduced mixing rates as jet Mach number increases, they are based on empirical findings and are also used with underlying RANS models that do not replicate key features of an actual jet such as the initial mixing rate and intermittency of turbulence near the end of the potential core. In contrast, highly-resolved LES does appear to capture the actual flow physics more accurately, as indicated by the WMLES predictions in Figs. 18 and 19 which were obtained with the same approach as used for the Phase 4 WMLES using the CharLES code³⁰. For the jet cases operated on-design and shown in Fig. 18, LES is generally able to replicate the trends of both jet compressibility and jet heating, although the mixing rates and accompanying rise in centerline turbulent kinetic energy are somewhat more rapid than indicated in the experimental results, especially as the jet Mach number increases. Figure 19 shows comparisons for one of the off-design jet cases measured during the THX5 experimental campaign. The farfield mixing rate and centerline turbulent kinetic energy is actually predicted more accurately than for the on-design cases. This is possibly due to the effective jet speed by the end of the shock train in the potential core being lower than that for the Mach 1.36 on-design case, and hence the compressibility effects are lower than for the on-design cases. A summary of all the PAW6 results is provided in Ref. 39, with more comprehensive details of several of the investigations provided in Refs. 40-43. Despite the lack of perfect agreement between the LES calculations and experimental data in Figs 18 and 19, these LES results are superior to any of the RANS results that were obtained in PAW6 and documented in Ref. 39.

Three new supersonic jet cases were built for the NASA Turbulence Modeling Resource (TMR) and were taken from the PAW6 nozzle test problem. The three jet cases all used the THX Phase 5 Mach 1.63 axisymmetric nozzle: a heated jet for each at on-design condition, a temperature-matched on-design condition at Mach 1.63, and an off-design heated condition for the Mach 1.63 nozzle. RANS solutions were obtained using three established Reynolds-averaged Navier-Stokes (RANS) codes: Wind-US, FUN3D, and VULCAN-CFD, all using the Menter Shear Stress Transport $k-\omega$ turbulence model with vorticity source term (SST-V). The NASA TMR requires at least two CFD codes yielding essentially identical results to certify that the approaches are verified, meaning that they solve the posed CFD equations as intended. The three codes generally provided very close agreement with each other for jet plume quantities, with the only exception being static temperature. This discrepancy was determined to be the result of whether or not turbulent kinetic energy was considered in the definition of total energy.

The links for all of the cases, with computational grids, experimental data, and sample computational results as documented in a summary report (Ref. 44) of the new TMR cases are as shown below:

AJM163TM: Temperature-Matched Mach 1.63 Axisymmetric Jet:

https://turbmodels.larc.nasa.gov/jet_mach1p63_tmached_val.html

AJM163H: Heated Mach 1.63 Axisymmetric Jet:

https://turbmodels.larc.nasa.gov/jet_mach1p63_hot_val.html

AJM163OD: Off-Design Mach 1.63 Axisymmetric Jet:

https://turbmodels.larc.nasa.gov/jet_mach1p63_offdesign_val.html

Conclusions

Turbulent thermal transport is an important consideration for aerospace flows; however it has not been studied experimentally as extensively as basic aerodynamic flows. For one reason alone, making measurements in heated flows naturally involves more harsh testing environments. The dearth of available experimental data for heated flows has limited the quantity and detail of computational fluid dynamics (CFD) validation and new model development. The NASA Turbulent Heat Flux (THX) series of experiments was an effort consisting of five phases that was designed and executed to collect data for several heated flow problems, with focus on propulsion applications. Phases 1, 3, and 4 examined film cooling configurations. Phase 2 examined heated subsonic jets and phase 5 examined heated supersonic jets. Computational studies were also conducted for each of the experimental configurations, in order to provide a baseline of expected CFD results and conduct an assessment of the capability of various CFD approaches for calculating flows where the turbulent transport of heat was important. These included production-ready Reynolds-averaged Navier-Stokes (RANS) methods as well as scale-resolving (SRS) approaches, utilizing large-eddy simulation (LES) for all or at least the focus region of the flow under consideration.

A new rotationally-resolved Raman scattering temperature measurement capability was advanced in technology via these THX efforts and used in Phases 2 through 5 to make measurements in model scale facility experiments at temperatures more characteristic of actual operating propulsion systems than the lower temperatures of typical fundamental experiments. Velocity measurements were made using particle image velocimetry (PIV) for all of the phases except Phase 2. The Phase 2 subsonic jet experiments took temperature measurements for cases where extensive PIV data had previously been obtained in the same facility, and for the same nozzles and jet operating points as that previously collected. This previously collected PIV data is known as The NASA Subsonic Jet Particle Image Velocimetry (PIV) dataset.

For the film cooling experiments of Phases 1 and 3, detailed measurements were made in the near field of the inclined jet-in-cross flow configurations. Phase 4 consisted of a multi-hole film cooling hole problem that is more representative of modern film cooling arrangements and focused more on the aggregate effects of the interaction of a cooling film with heated primary stream. Surface temperature measurements were also made for this phase in order to characterize the resultant surface temperatures with various blowing ratios.

Considering results of the CFD investigations that complemented the experimental efforts for the film cooling configurations, it was shown that SRS methods (i.e. simulation including LES for the most important part of the flow under consideration) are significantly more accurate than RANS-based techniques for turbulent heat transport. The RANS gradient-diffusion hypothesis used to calculate turbulent thermal transport is likely inadequate for many flows and inferior to near-wall thermal boundary layer treatments using scale-resolving approaches. A conjugate heat transfer (CHT) modeling approach, in conjunction with the CFD solver, was required to obtain accurate surface temperature measurements, regardless if a RANS- or LES-based approach was utilized.

The experimental data for the heated subsonic jets tested in Phase 2 provided temperature data to be utilized in conjunction with velocity data obtained for the same nozzles and operating points, as well as tested in the same facility, but prior to the THX experimental campaign. Phase 5 examined supersonic jets using the same temperature measurement techniques as Phase 2 and PIV for the velocity components. Experimental results for the supersonic jets showed that as jet Mach number increases, the overall jet mixing rate decreases, as indicated by the potential core length, downstream mean velocities, and turbulence levels. Similar behavior was observed in the mean temperatures and rms temperatures. The suppression of turbulence as the jet Mach number increases is due to compressibility. For a fixed jet Mach number, increasing the jet temperature resulted in faster mixing by all measures.

In considering CFD assessments of the round jet data, the RANS approaches were unable to capture the trend of reduced mixing as the jet Mach number increased. Incorporation of compressibility corrections improved the agreement with experimental measurements in terms of trends with varying jet Mach number, but also increased the jet potential core lengths to be in worse agreement with data. RANS methods were able to reasonably capture trends

of jet heating. The LES-based approaches as a whole were able to capture the effects of compressibility more accurately. Effects of jet heating were better characterized by LES.

Hopefully this paper will serve as an archival source of the entire THX campaign results for those wishing to use the experimental data for thermal modeling assessment. More details of the individual phases may found in the references provided in this paper. There are many other flow turbulent problems for which more detailed temperature measurements are needed. These include high Mach number boundary layers and the flows within propulsion system components including turbomachinery stages, combustors, and the internal surfaces of exhaust nozzles.

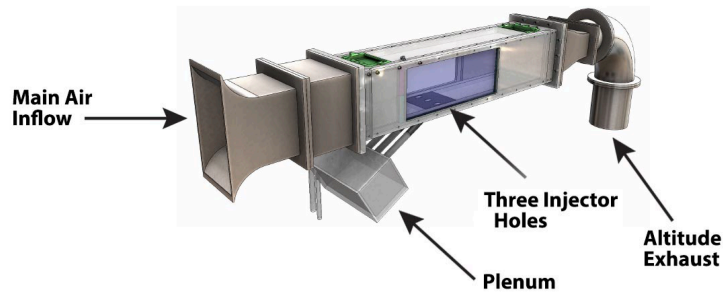
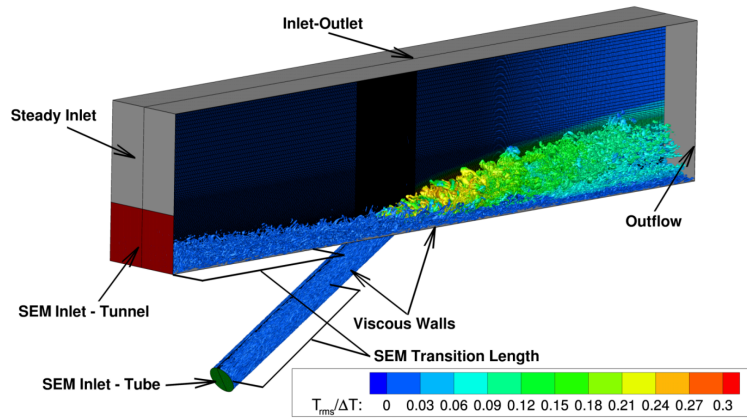
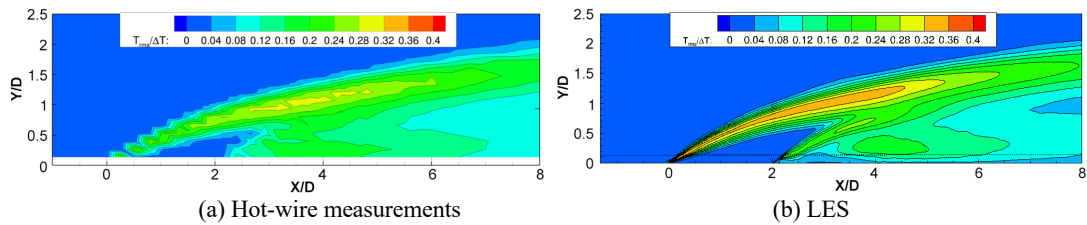
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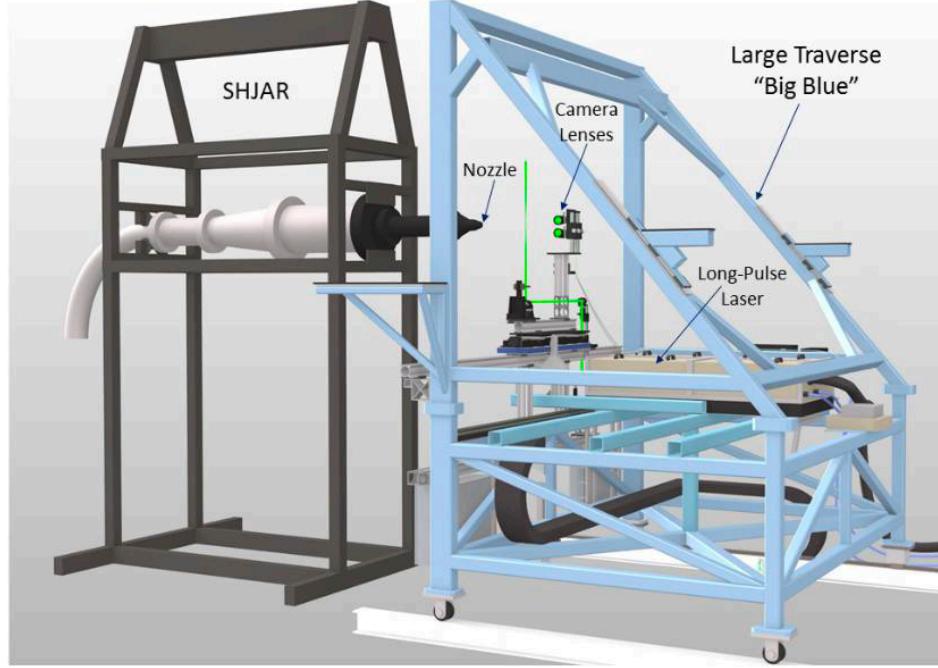
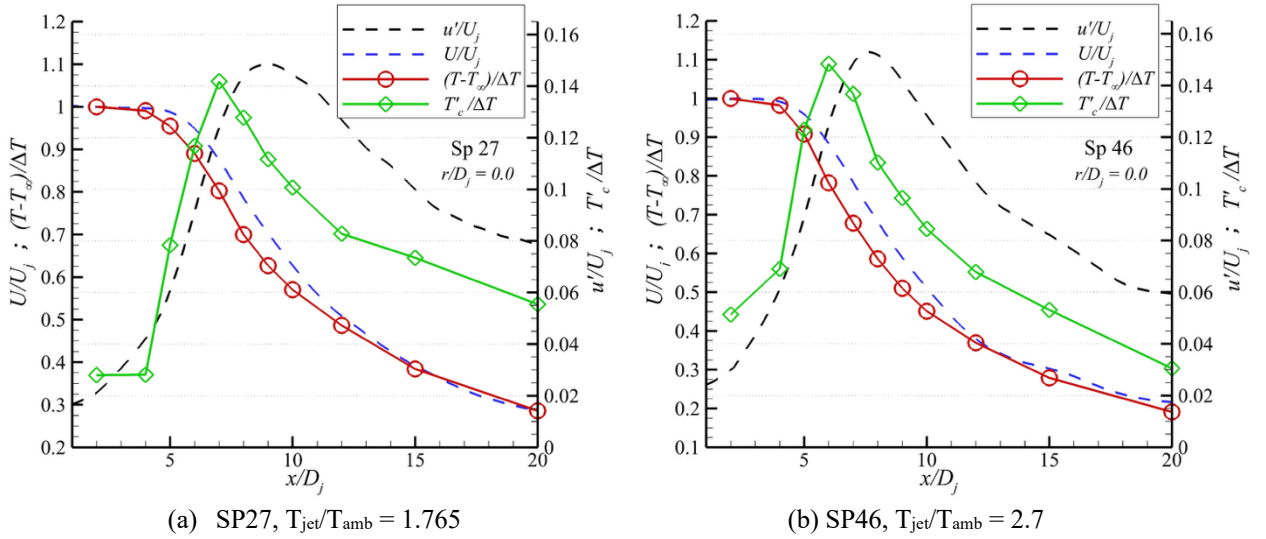


Figure 4.—Test rig (SHJAR) setup with “Big Blue” traversing measurement assembly.



(a) SP27, $T_{jet}/T_{amb} = 1.765$ (b) SP46, $T_{jet}/T_{amb} = 2.7$
 Figure 5.—Comparison of momentum and thermal mixing for two subsonic jet set points, both with $M_a = 0.9$; mean and rms for both centerline velocity and temperature are presented.

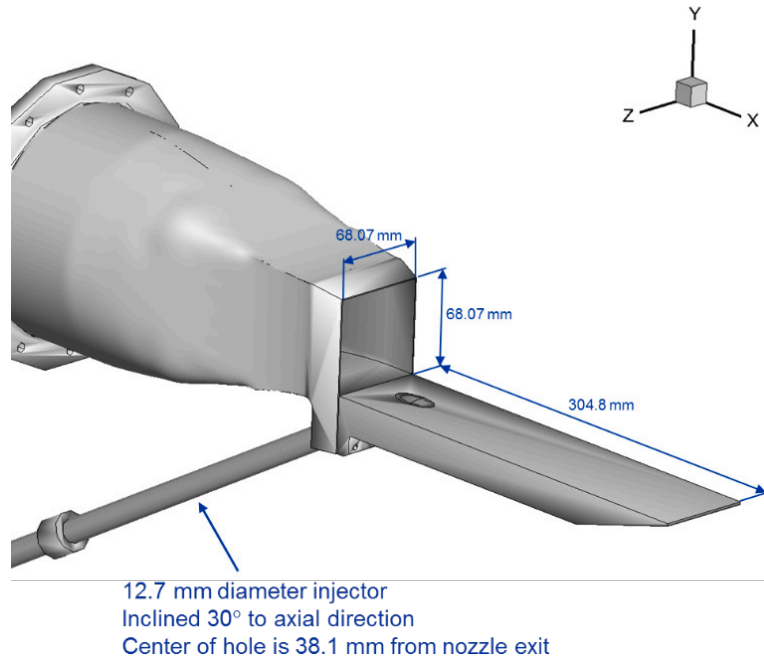
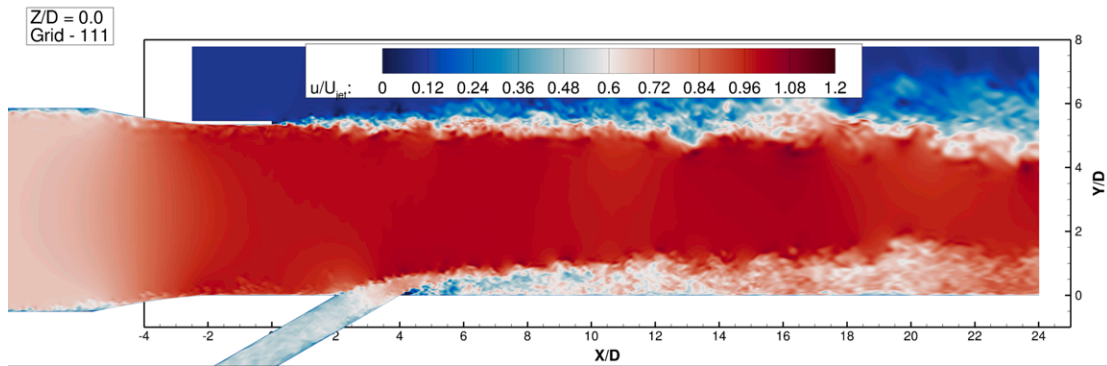
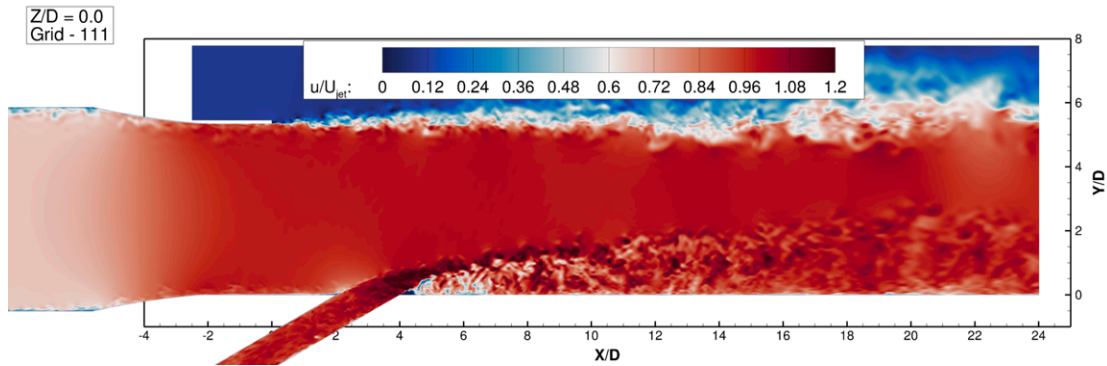


Figure 6.—THX – Phase 3 test configuration.



(a) BR = 1



(b) BR = 2

Figure 7.—Axial velocity contours from WRLES calculations using FDL3DI.

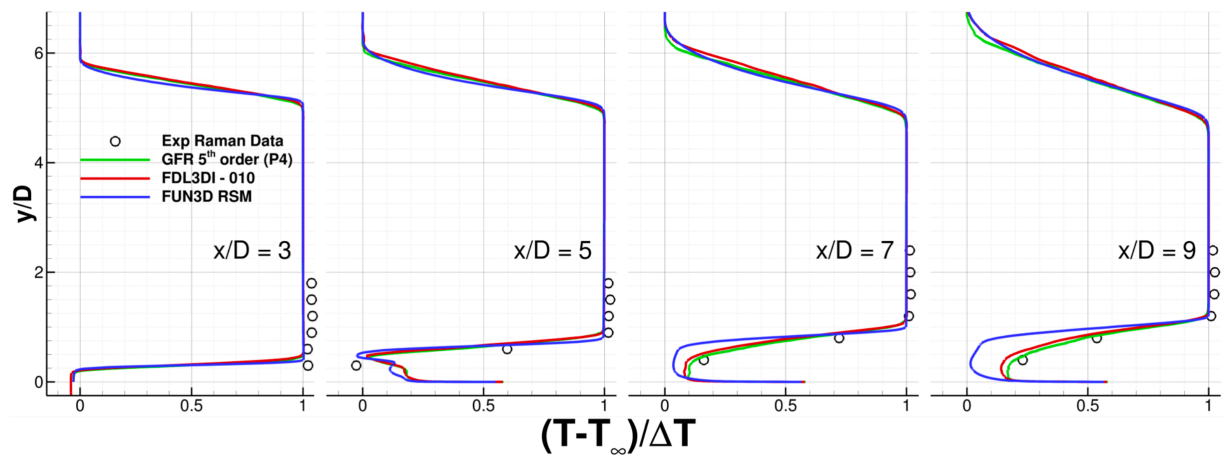


Figure 8.—Centerline mean temperature profiles; comparison of experiment to LES and RANS computations.

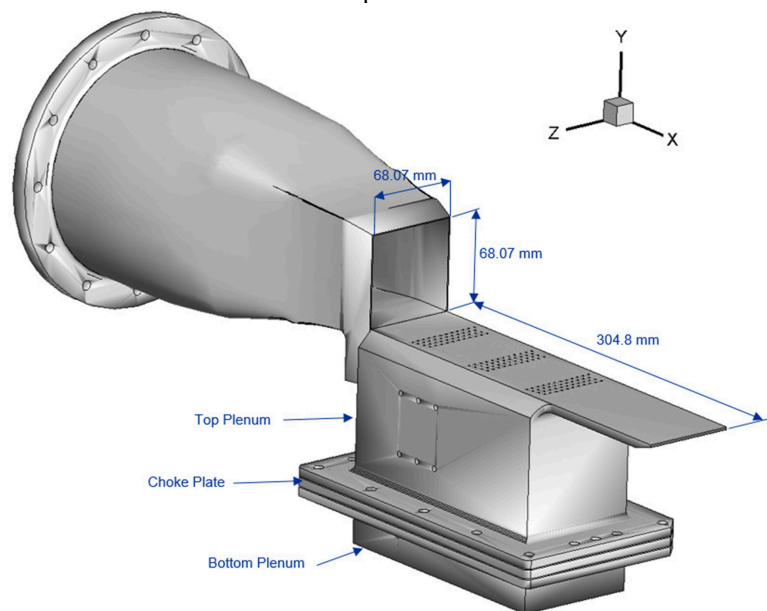


Figure 9.—THX – Phase 4 test configuration.

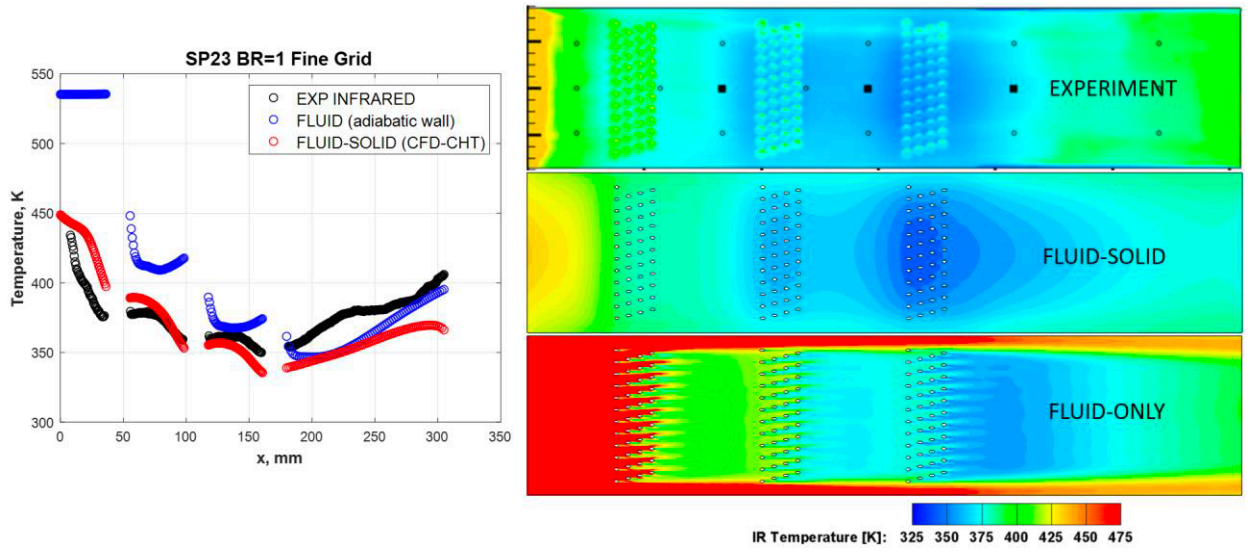


Figure 10.— Centerline temperature (left) and plate surface temperature contours (right) for $T_{jet} / T_{amb} = 1.765$, $BR=1$, calculated using RANS with and without CHT.

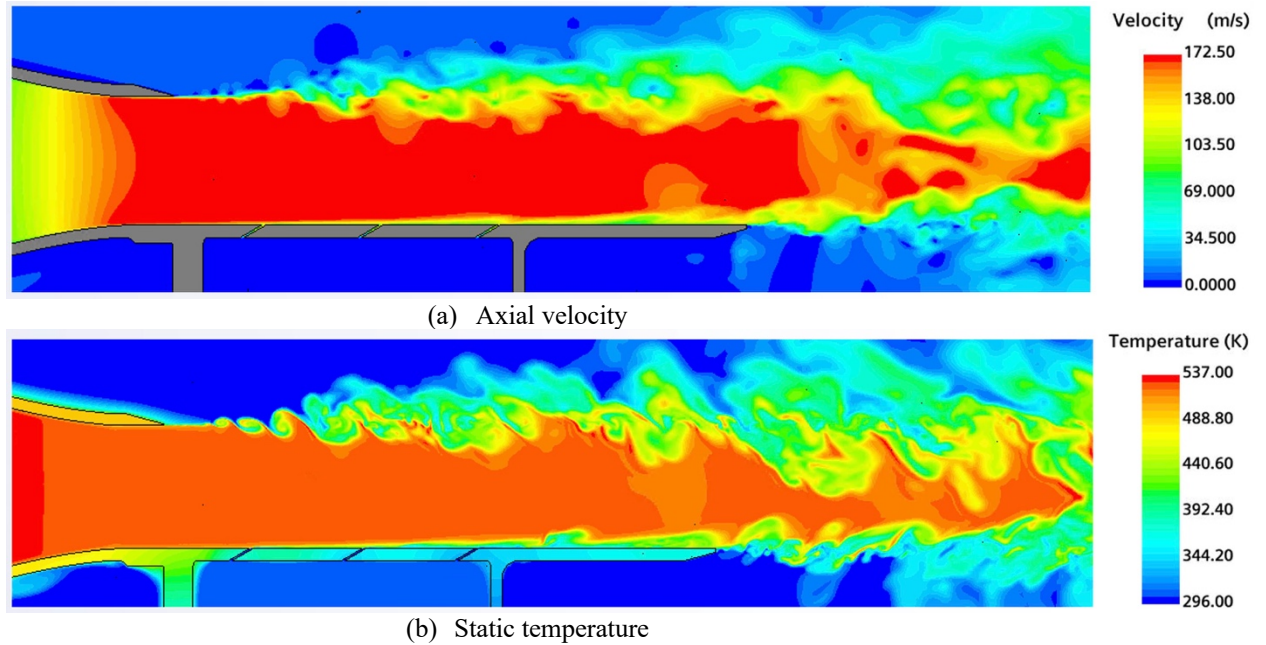


Figure 11.—Instantaneous center-plane contours using HRLES with CHT.

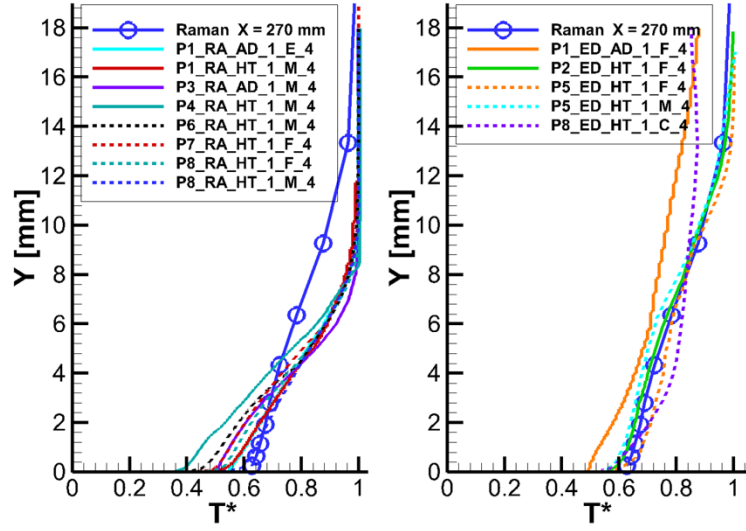


Figure 12.— Temperature profiles comparisons along the plate overhang (left-RANS, right-LES).

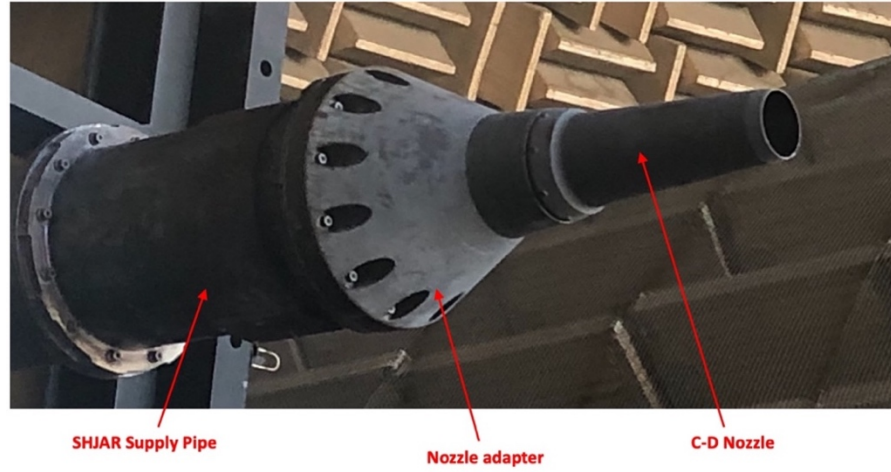


Figure 13.—Test rig (SHJAR) setup with C-D nozzle installed.

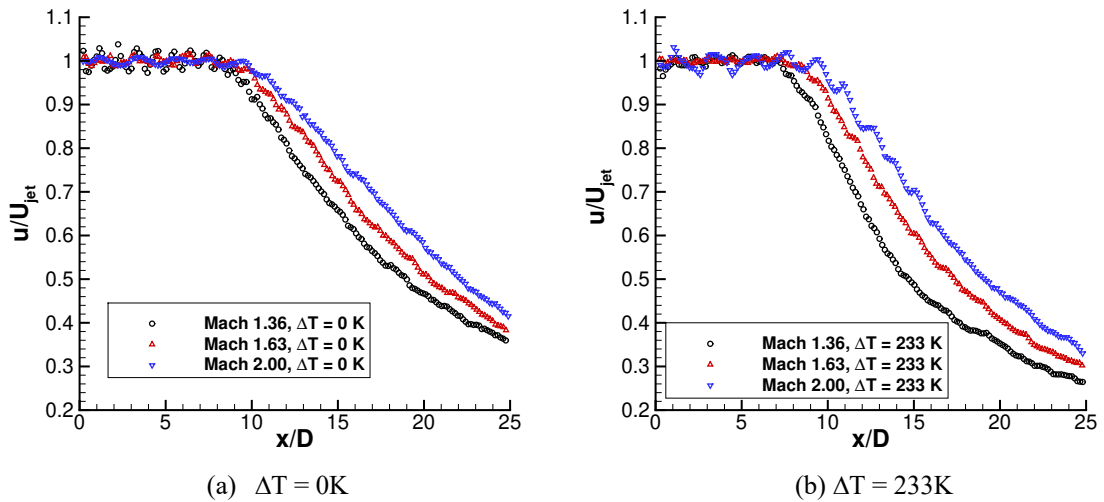
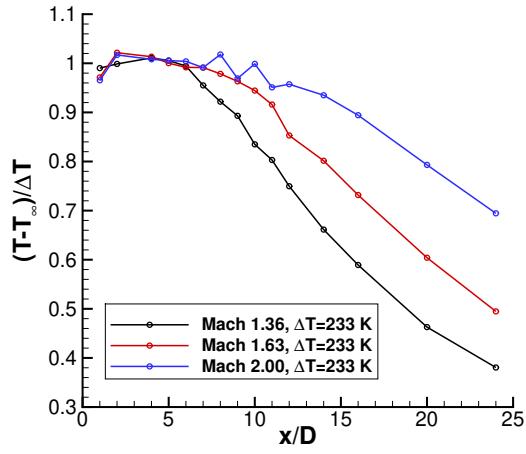
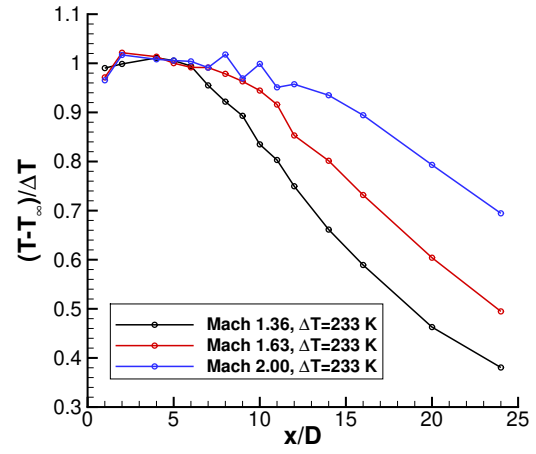


Figure 14.—Centerline jet velocity variations with jet Mach number for $\Delta T = 0K$.

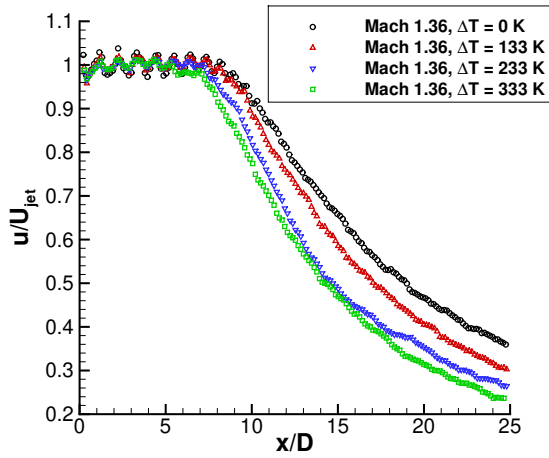


(a) mean temperature

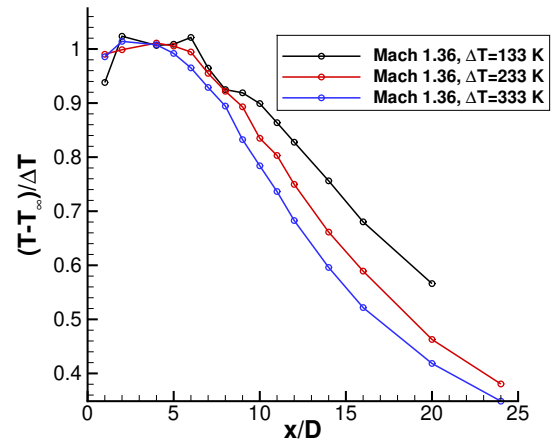


(b) rms of temperature

Figure 15.—Effects of Mach number variation on centerline temperature statistics for $\Delta T = 233$ K.



(a) mean axial velocity



(b) mean temperature

Figure 16.—Effects of jet temperature variation on centerline properties.

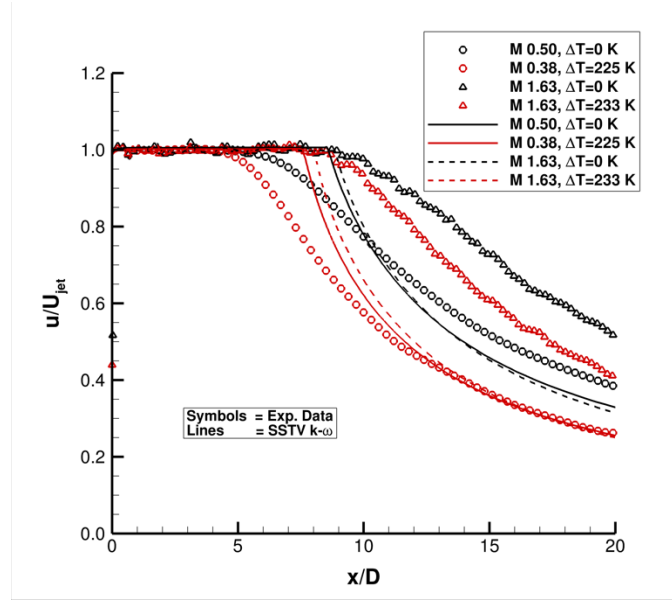
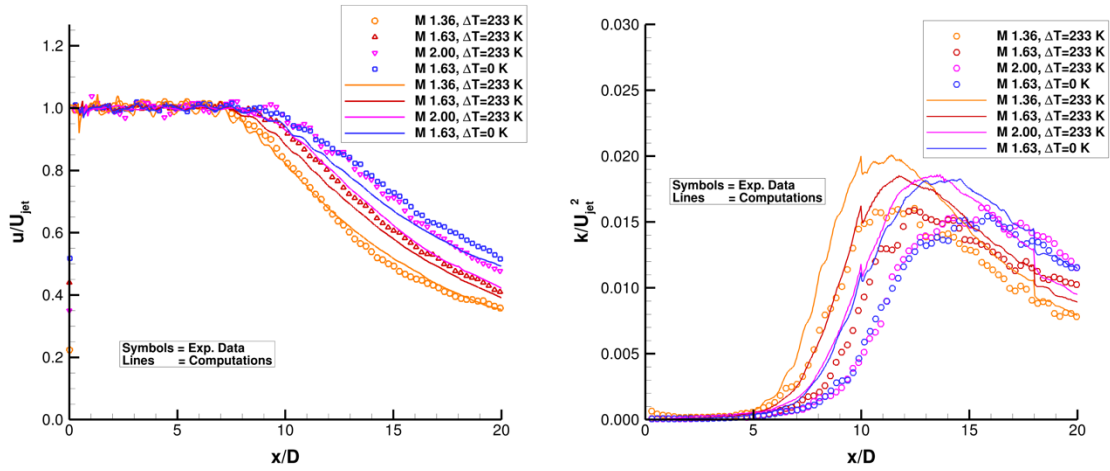


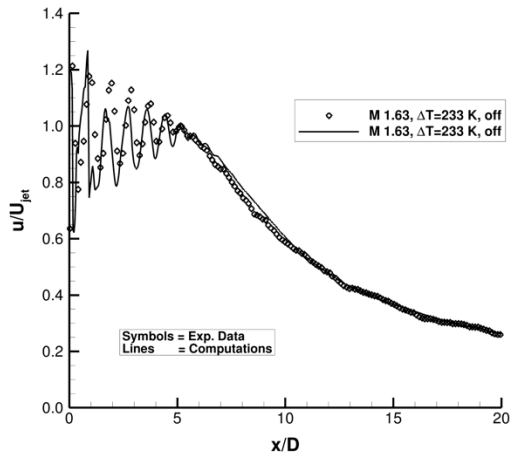
Figure 17.— Comparison of subsonic and supersonic round jets.



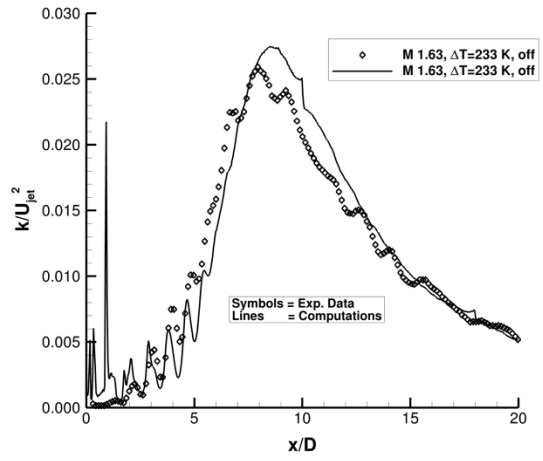
(a) axial velocity

(b) turbulent kinetic energy

Figure 18.—Centerline velocity statistics for WMLES solutions, on-design cases.



(a) axial velocity



(b) turbulent kinetic energy

Figure 19.—Centerline velocity statistics for WMLES solutions, off-design case.