NASA Space Launch System (SLS) Base Heating Test: Preliminary Design/Improvements

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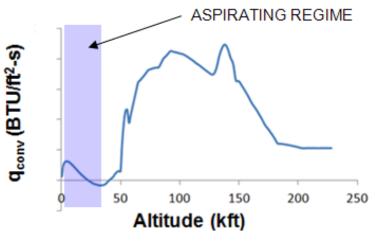
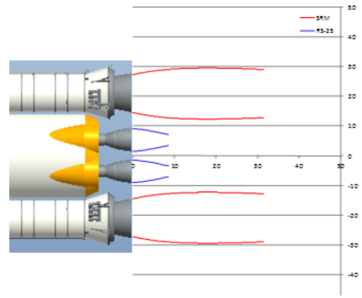
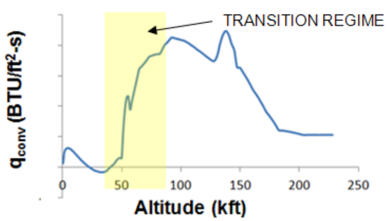
SLS – 10001 Base Heating Physics

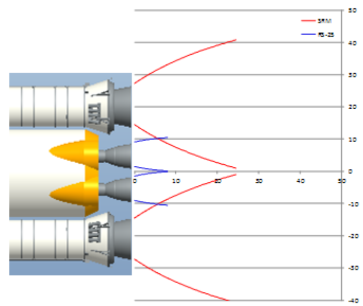
- **Aspirating Flow, 10Kft**



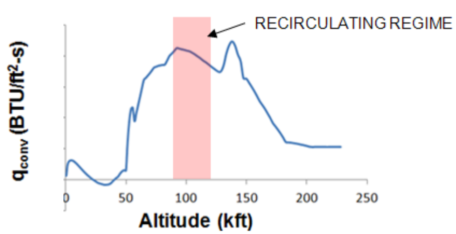


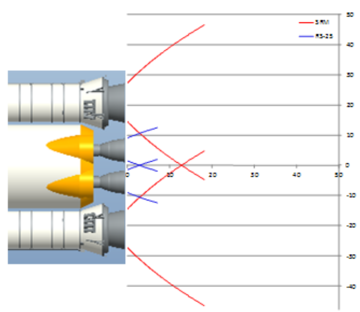
- **Core Transition, 48-50Kft**



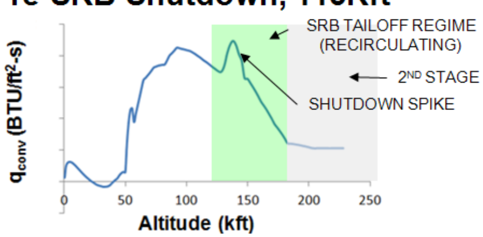


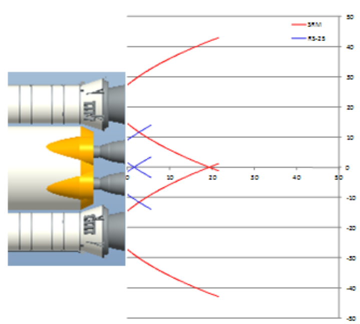
- **Peak Heating, 85Kft**





- **Pre-SRB Shutdown, 115Kft**





Motivation

- CFD and semi-empirical methodologies show poor comparisons
  - CFD solutions underpredict the semi-empirical results by a factor of ~5 for all base heat shield body points

- Base flows demonstrate complex flow physics
  - No pure analytical methods have been developed for base environment prediction

- New base geometry and performance requirements for the SLS vehicle – cannot blindly use heritage data

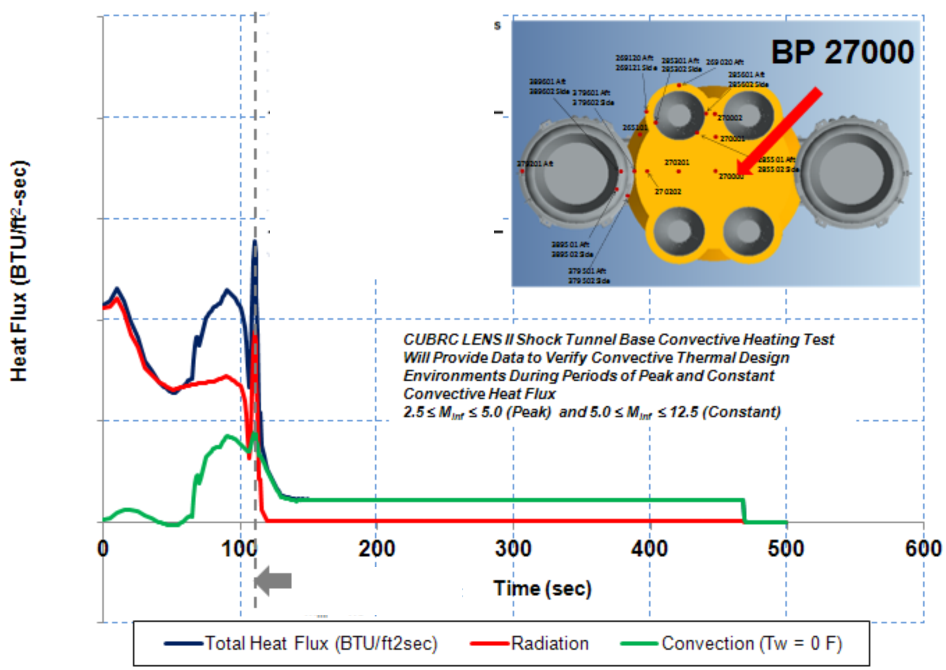
- Base flow environments are needed to efficiently size the TPS

Test Goals and Objectives

Goal 1: Test data to be used to scale to SLS flight environments

Goal 2: Test data to be used for validating the semi-empirical base heating and CFD methodologies

DAC-1 SLS Body Point 270000



Goal 3: Measure convective heat flux, static pressure and gas temperature distributions along the base heat shield and external model RS-25D and RSRM nozzles.

Goal 4: Measure these distributions at various points along the launch vehicle trajectory with an altitude sweep from 0 to 145 kft with the first stage core and boosters and a sweep from 145 kft to 200 kft with the second stage core.

Goal 5: Measure base flow parameter distributions for various mission critical cases such as: (a) RS-25D engine-out; (b) RS-25D/RSRM engine gimbal angle sweep*; (c) angle of attack sweep; (d) SRB thrust mismatch

Test Goals and Objectives

Pathfinder Test Program

- Objective 1: Design the internal propulsion system for the ~2% model core stage RS-25D engines.
  - Core stage is composed of 4 RS-25D GO_2 and GH_2 engines
- Objective 2: Fabricate the internal propulsion system for the model core stage RS-25D engines.
- Objective 3: Test the model core stage internal propulsion system
  - TBD hot-fire tests
  - Provide raw/reduced test data of engine performance
- Objective 4: Design the internal propulsion system for the ~2% model booster RSRM elements
  - Booster element is composed of two 5-segment SRBs
- Objective 5: Fabricate the internal propulsion system for the model booster RSRM elements
- Objective 6: Test the model booster internal propulsion system
  - TBD hot-fire tests
  - Provide raw/reduced data of engine performance

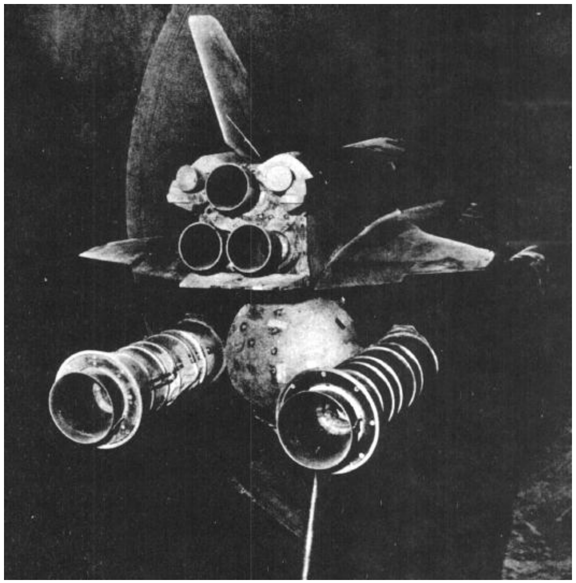
SLS – BHT Program

- Objective 1: Design the model SLS-10001 outer mold line shell (OML).
  - Finalize instrumentation layout and specifications
- Objective 2: Fabricate the model SLS-10001 OML and layout instrumentation
  - Integrate SLS-10001 OML with the internal propulsion system developed within the Pathfinder Test Program
- Objective 3: SLS Base Heating Test
  - 100 test runs
  - Altitude, angle of attack and gimbal angle sweeps
  - SRB thrust mismatch, Reynolds effect, engine-out and repeat run cases
  - GTP cases
  - Provide raw/reduced data of entire test (if possible)

CUBRC LENS II Facility

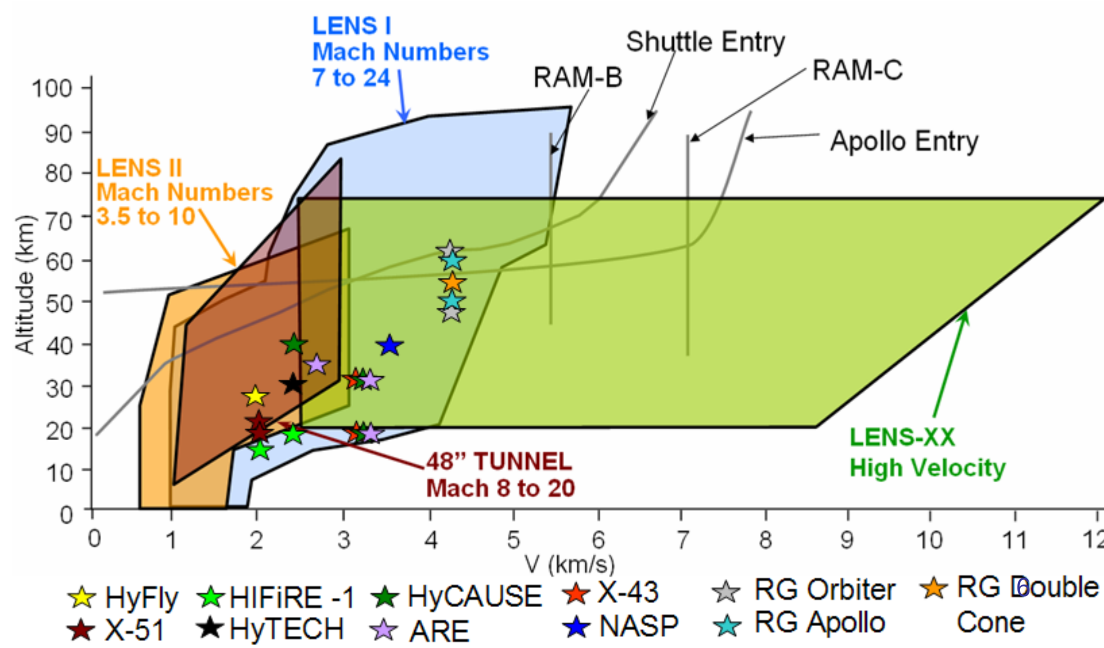
CUBRC = Calspan – University of Buffalo Research Center

2.25% Scale Space Shuttle Model

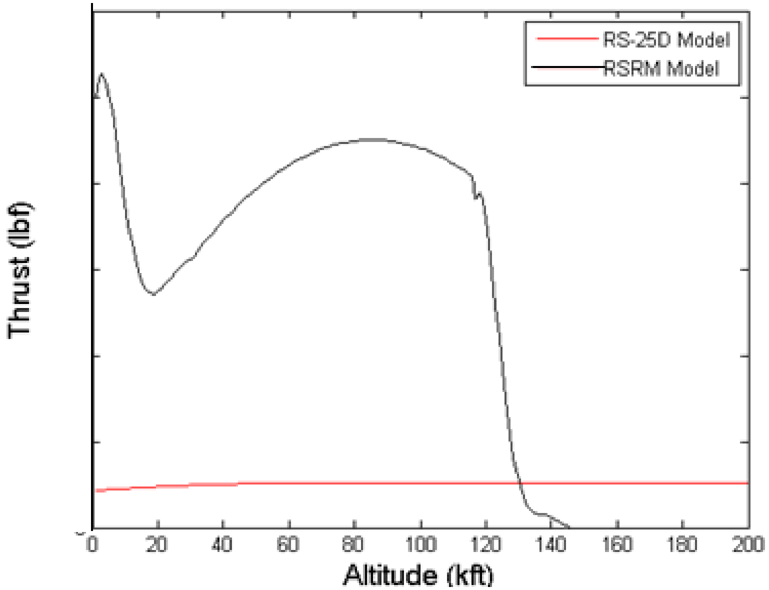


CUBRC 60 ft. Ludwieg Tube

Test Section: 42” diameter, 60” length

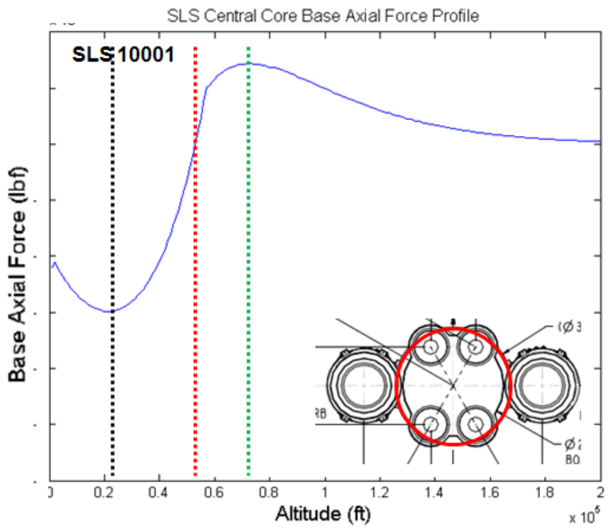


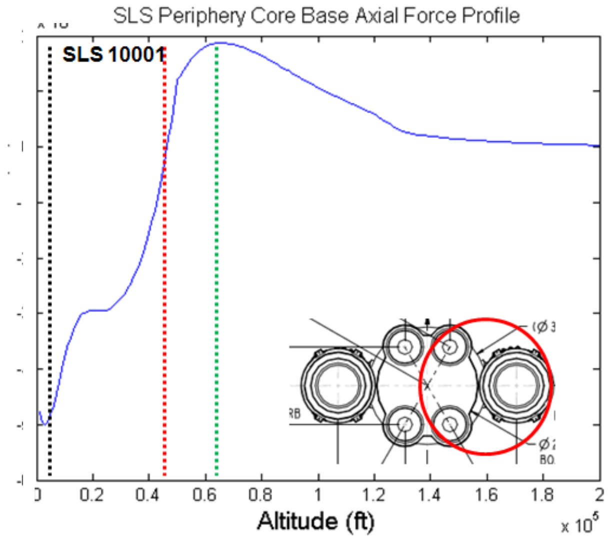
Test Run Condition Selection



The base pressure and base heating characteristics between central core and periphery core base regions are different.

This is due to the difference in plume-plume interactions between the RS-25D engines and the RSRMs (note the large difference in thrust between the two elements).





Test Run Condition Selection

- To accurately characterize base flow during vehicle ascent and areas of high heating rates
  - Avg Max Drag, Avg Transition Point, Avg Max Base Force, Avg Max Convective heating are needed
- To accurately quantify the heat loads
  - Booster – Sep and second stage flight data points are needed.

FLT Mach	WT Mach	FLT Altitude (kft)	WT Equiv Pressure Alt (kft)	WT Ambient Pressure (psia)	FLT Ambient Temperature (°R)	Model SSME Pc (psia)	Model RSRM Pc (psia)	Design Point
0.3	2.5	12	12	9.92	521.0	2956.0	1090.0	**Avg** Max Drag
1.4	2.5	33.5	33.5	3.55	406.0	2956.0	689.5	IR1
2.0	2.5	49.5	49.5	1.71	371.0	2956.0	780.0	**Avg** Transition Point
2.8	2.8	69	69	0.69	375.0	2956.0	852.0	**Avg** Max Base Force
3.4	3.4	83.5	83.5	0.35	405.0	2956.0	888.0	**Avg** Max Convective Heating
4.0	4.0	95	95	0.21	405.0	2956.0	885.0	IR2
4.5	4.5	110	110	0.10	430.0	2956.0	820.0	IR3
4.8	4.8	120	120	0.06	445.0	2956.0	600.0	SRB Tailoff
4.9	4.9	130	130	0.04	460.0	2956.0	108.0	Shutdown Spike
4.9	4.9	145	145	0.02	480.0	2956.0	0.0	Booster Sep - Start
5.1	5.1	175	175	0.005	475.0	2956.0	0.0	Second Stage Flight
5.4	5.4	200	200	0.002	450.0	2956.0	0.0	Second Stage Flight

WT = Wind Tunnel
FLT = Flight

Preliminary SLS-BHT Test Matrix

Table 1 PRELIMINARY TEST RUN MATRIX FOR SLS-BHT

EVENTS	Mach No. (M)	Pressure Altitude (kft)	Angle of Attack (AOA)	Gimbal Angle (GA)	Engine Out (E/O)	SSME Reynolds No.	RSRM Reynolds No.	Gas Temp Probe (GTP)	Repeat Runs (RR)	Full-Stack /Core-Stage	SRB Thrust Mismatch (SRB TM)	Tw/To	To	h	Number of Tests
Avg Aspirating	2.5	12	NOM	NOM	NOM	Pc	Pc	GTP1,NULL		Full-Stack					2
IR1	2.5	33.5	NOM	NOM	NOM	Pc	Pc	GTP1,NULL		Full-Stack					2
Avg Transition	2.5	49.5	AOA1,AOA2, NOM	NOM	E/O1, NOM	Pc, 3/4 Pc, 1/2 Pc	Pc, 3/4 Pc, 1/2 Pc	GTP1,NULL	RR1	Full-Stack					13
Avg Max Base Force	3	69.5	NOM	NOM	NOM	Pc	Pc	GTP1,NULL		Full-Stack					2
Avg Max Q-Rate Heating	3.5	83.5	AOA1,AOA2, NOM	GA1, GA2, NOM	E/O1, NOM	Pc, 3/4 Pc, 1/2 Pc	Pc, 3/4 Pc, 1/2 Pc	GTP1,NULL	RR1	Full-Stack		Tw/To1	To1	h1	24
IR2	4	95	NOM	NOM, GA1, GA2	NOM	Pc	Pc	GTP1,NULL	RR1	Full-Stack					7
IR3	4.5	110	NOM	NOM	NOM	Pc	Pc	GTP1,NULL		Full-Stack					2
SRB Tailfoff	5	120	NOM	GA1, GA2, GA3, GA4, NOM	NOM	Pc	Pc, 3/4 Pc, 1/2 Pc	GTP1,NULL	RR1	Full-Stack	SRB TM1, SRB TM 2				15
Shutdown Spike Comparison	5	130	NOM	NOM	NOM	Pc	Pc	GTP1,NULL		Full-Stack					2
Booster Separation	5	145	AOA1,AOA2, NOM	GA1, GA2,NOM	E/O1, NOM	Pc, 3/4 Pc, 1/2 Pc	0	GTP1,NULL	RR1	Core-Stage					18
Second-Stage Flight	5	175	NOM	NOM	NOM	Pc	0	GTP1,NULL	RR1	Core-Stage					2
Second-Stage Flight	5.5	200	NOM	NOM	E/O1, NOM	Pc, 3/4 Pc, 1/2 Pc	0	GTP1,NULL	RR1	Core-Stage					10
														Total	99

Note: NOM = Nominal Flight Trajectory; NULL = without; AOA1 = +TBD; AOA2 = -TBD, GA1,GA2 = +/- TBD; GA3,GA4 = TBD, correlates to SRB mismatch;
Tw/To = Model to freestream temperature ratio; To = freestream temperature; h = freestream total enthalpy; GTP1 = with Gas Temperature Probe Measurement

Dynamic Similarity Analysis

Nondimensional flow parameters derived from the compressible Navier-Stokes equations.

Reynolds No. / Mach No. / Prandtl No. / Specific heat ratio

$$\mathrm{Re} = \frac{\rho_e U_e D}{\mu_e}, \quad \cancel{\mathrm{Fr} = \frac{U_e}{\sqrt{gD}}}, \quad \mathrm{Ma} = \frac{U_e}{a_e} = \frac{U_e}{\sqrt{\gamma R T_e}}, \quad \mathrm{Pr} = \frac{\mu_e c_{pe}}{\lambda_e}, \quad \gamma = \frac{c_{pe}}{c_{ve}}$$

Thrust Coefficient

$$\frac{T}{P_\infty A_b} = \boxed{\frac{\dot{m} u_e}{P_\infty A_b}} + \boxed{\frac{\Delta P}{P_\infty}\frac{A_e}{A_b}} = C_T$$

Momentum Flux Contribution

Plume Expansion Contribution

Nondimensional flow parameters derived from the free surface and solid surface boundary conditions.

$$NPR = \left(\frac{P_0}{P_\infty}\right)_{steady}$$ Nozzle pressure ratio

$$e = \frac{P_e}{P_\infty}$$ Plume expansion ratio

Nusselt No.

$$\mathrm{Nu} = \frac{q_s D}{\lambda_e (T_s - T_e)} = \frac{h_e D}{\lambda_e} = 0.026 (\mathrm{Re}_D)^{0.8} \mathrm{Pr}^{0.4}$$

$$v'_{f\!o} = \frac{U_\infty}{U_e} = \frac{M_\infty}{M_e}\sqrt{\frac{\gamma_\infty R_\infty T_\infty}{\gamma_e R_e T_e}}$$

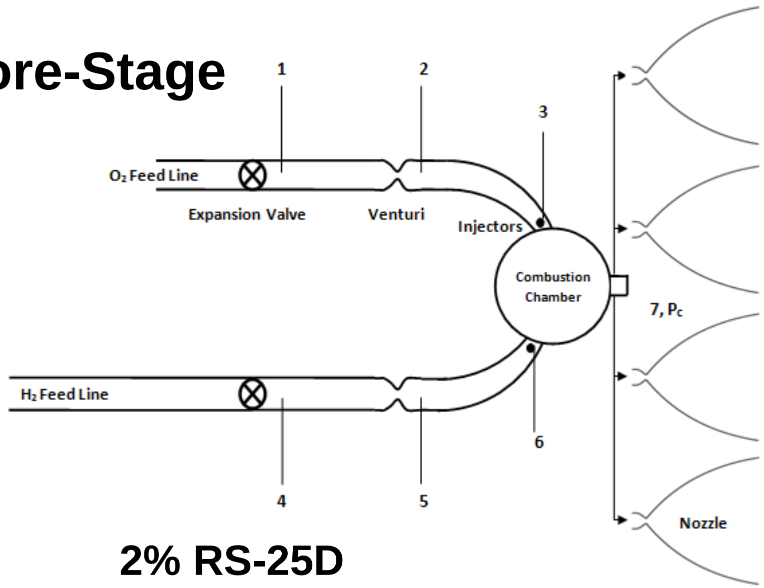
$$T'_{f\!o} = \frac{T_e - T_\infty}{T_e} = 1 - \frac{T_\infty}{T_e}$$

Dynamic Similarity Analysis

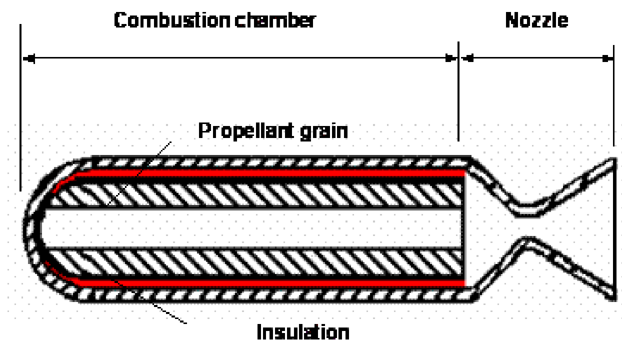
- Three groups of parameters need to match with flight to ensure dynamic similarity:
  - Thrust coefficient
    - Derived from the conservation of momentum
    - Ensures that the base pressure/force will be adequately modeled
  - Nozzle exit boundary layer specific enthalpy
    - Derived from the conservation of energy
    - Ensures that the specific energy that initially convects into the base is adequately modeled
  - Nondimensional flow parameters
    - Mach number and plume expansion ratio
      - Ensures that the boundary layer flow direction within the base region is adequately modeled
      - Ensures the compressibility effects and shock structure of the plume are adequately modeled
    - Prandtl number and specific heat ratio
      - Determines plume properties and state parameters
    - Oxidizer/fuel ratio
      - Critical in accurately modeling the chemical species and temperature distributions and plume properties within the nozzle
    - Reynolds number
      - Important in accurately determining the boundary layer properties/thickness and turbulence
      - Not able to accurately simulate for test
    - Nusselt number
      - Important in determining the heat transfer to the nozzle wall
      - Not able to accurately simulate for test

Preliminary Internal Propulsion Design

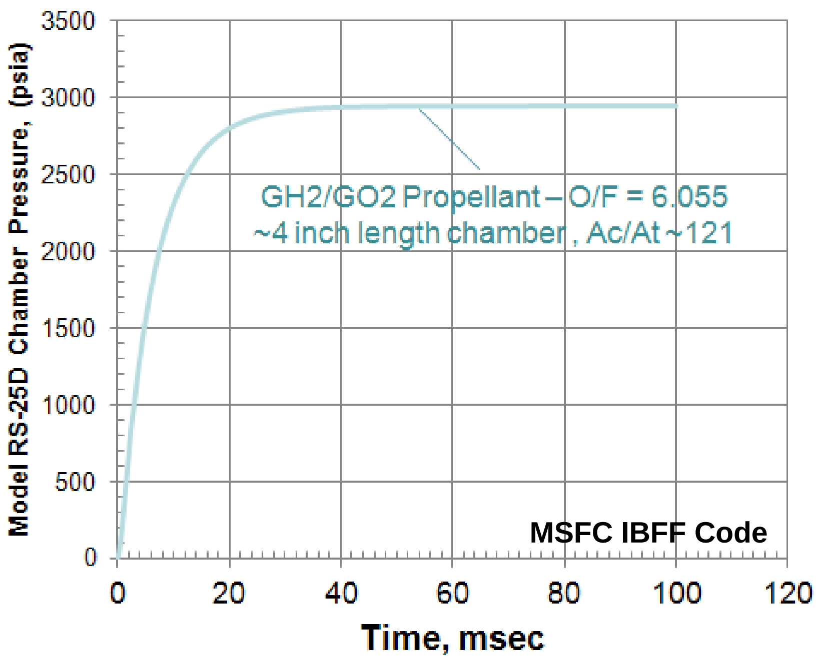
Core-Stage



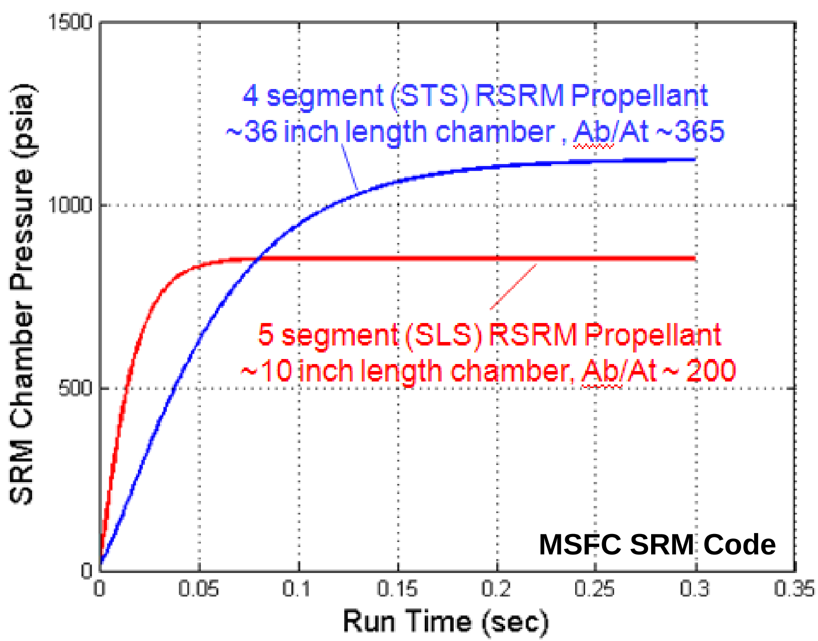
Booster



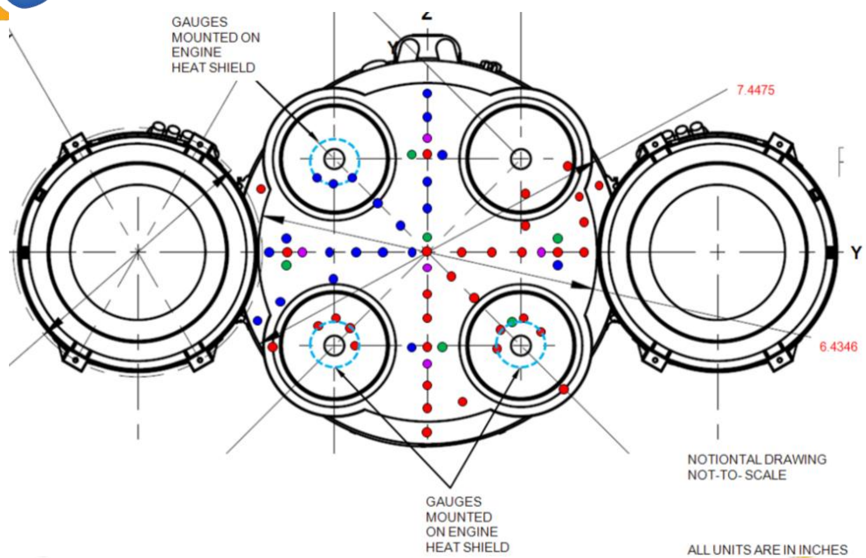
2% RS-25D

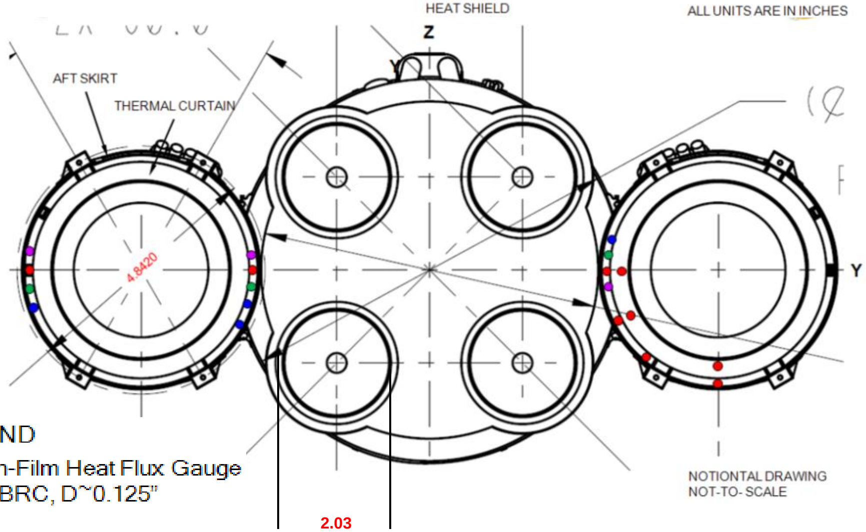


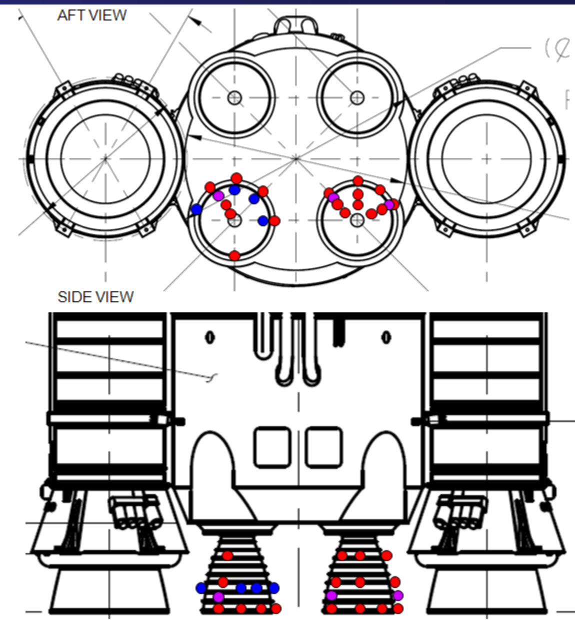
2% RSRM

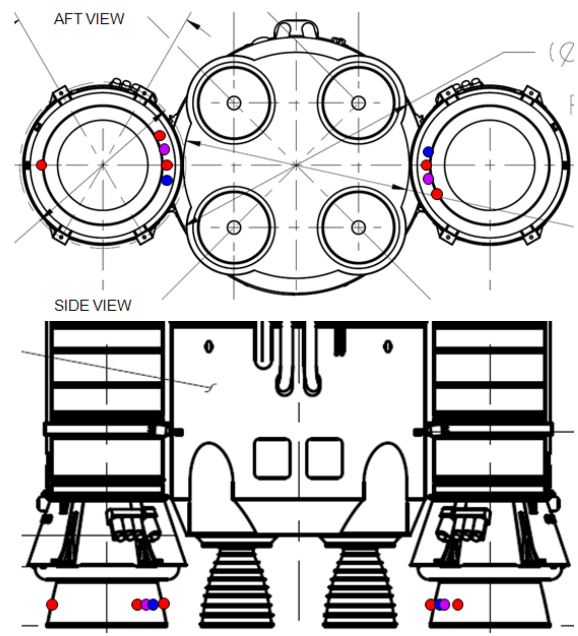


Preliminary Instrumentation Layout





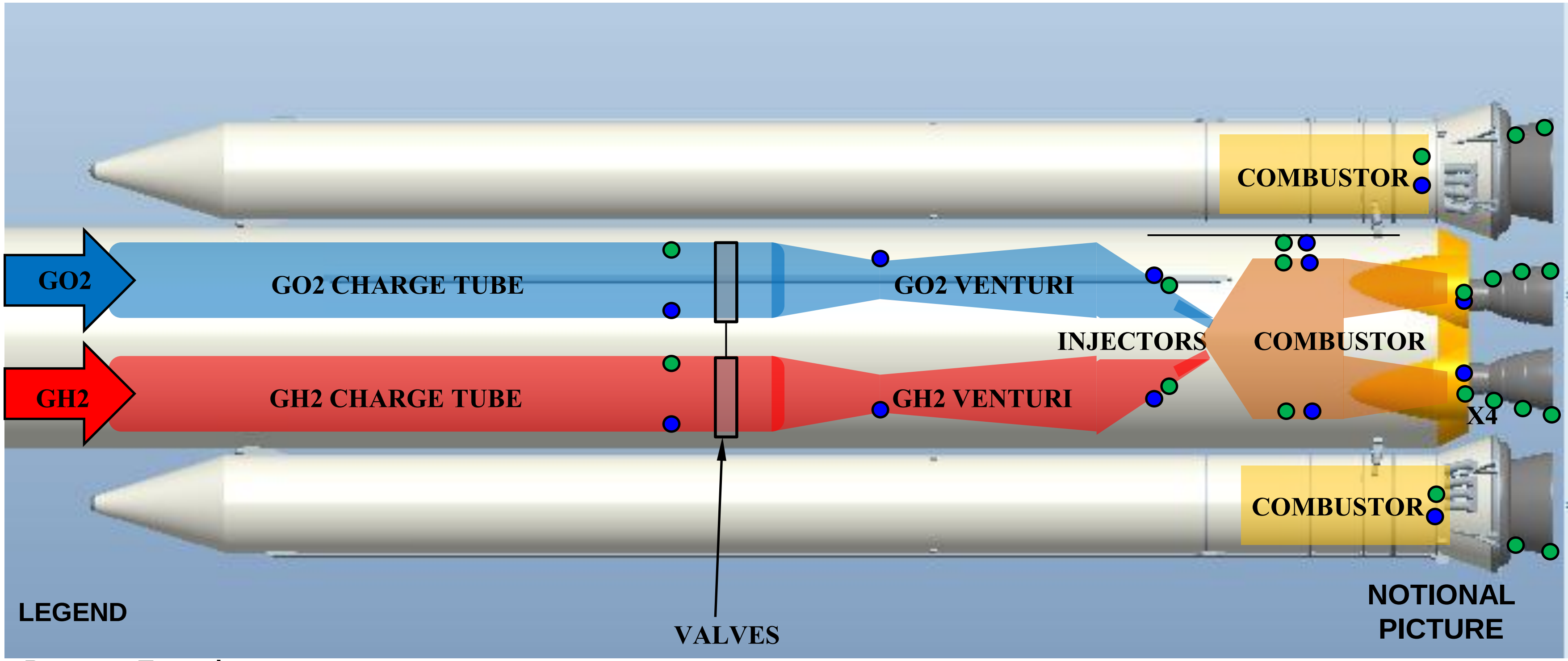




LEGEND

- Thin-Film Heat Flux Gauge CUBRC, D~0.125"
- Pressure Transducer CUBRC, D~0.16"
- Gas Temperature Probe Medtherm, D~0.15
- Radiometer Medtherm, D~0.15

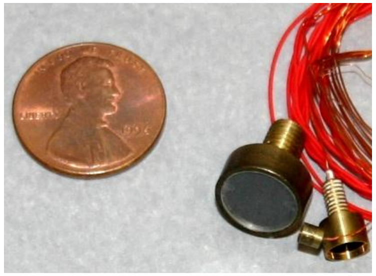
Preliminary Instrumentation Layout



Pressure Transducer

Temperature Probe

radiative heat transfer gauge



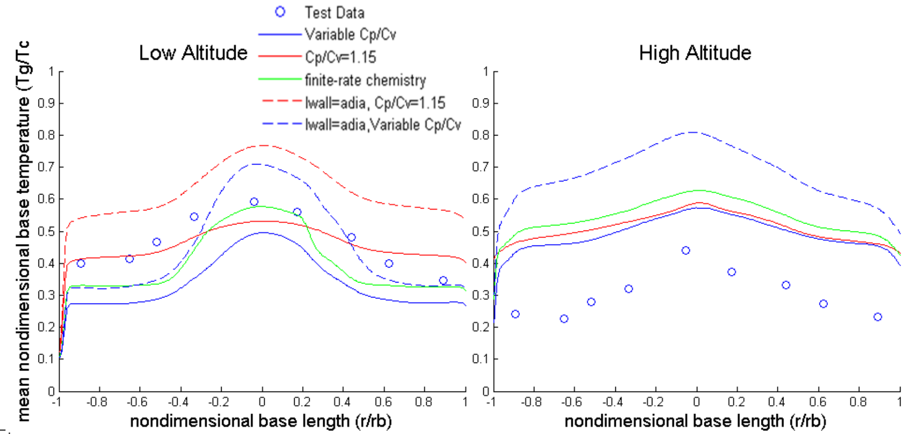
coaxial thermocouple

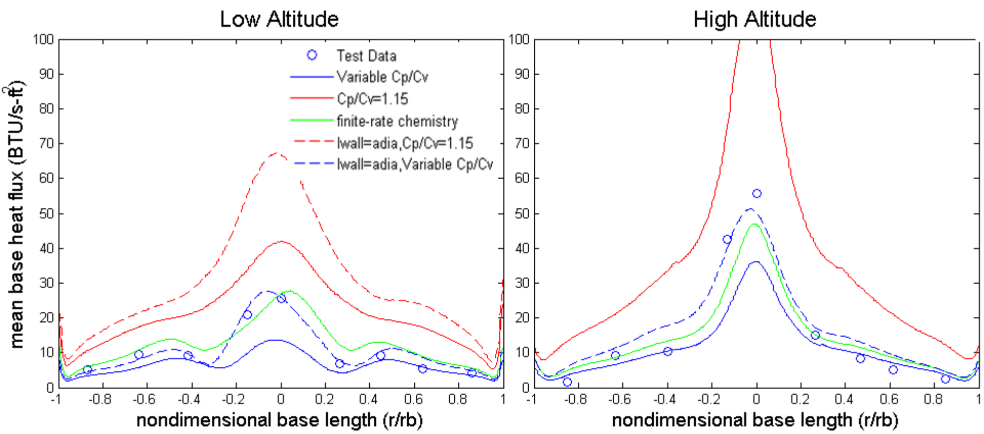
0.040” diameter thin film heat transfer gauge

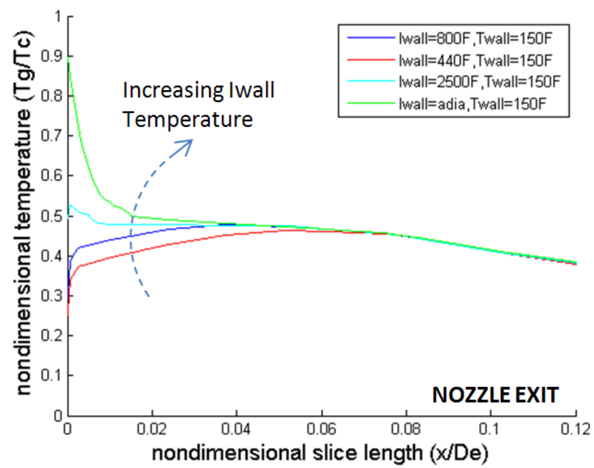
miniature Kulite pressure sensor

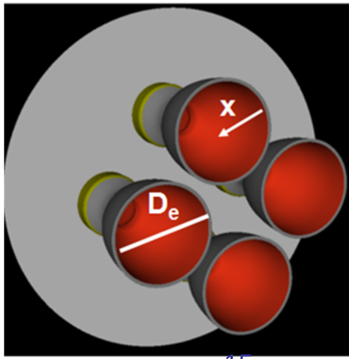
Nozzle Specific Enthalpy Flow Analysis

Both base gas temperature and heat flux are highly sensitive to the inner nozzle wall temperature. Large delta between the two different nozzle wall temperatures.







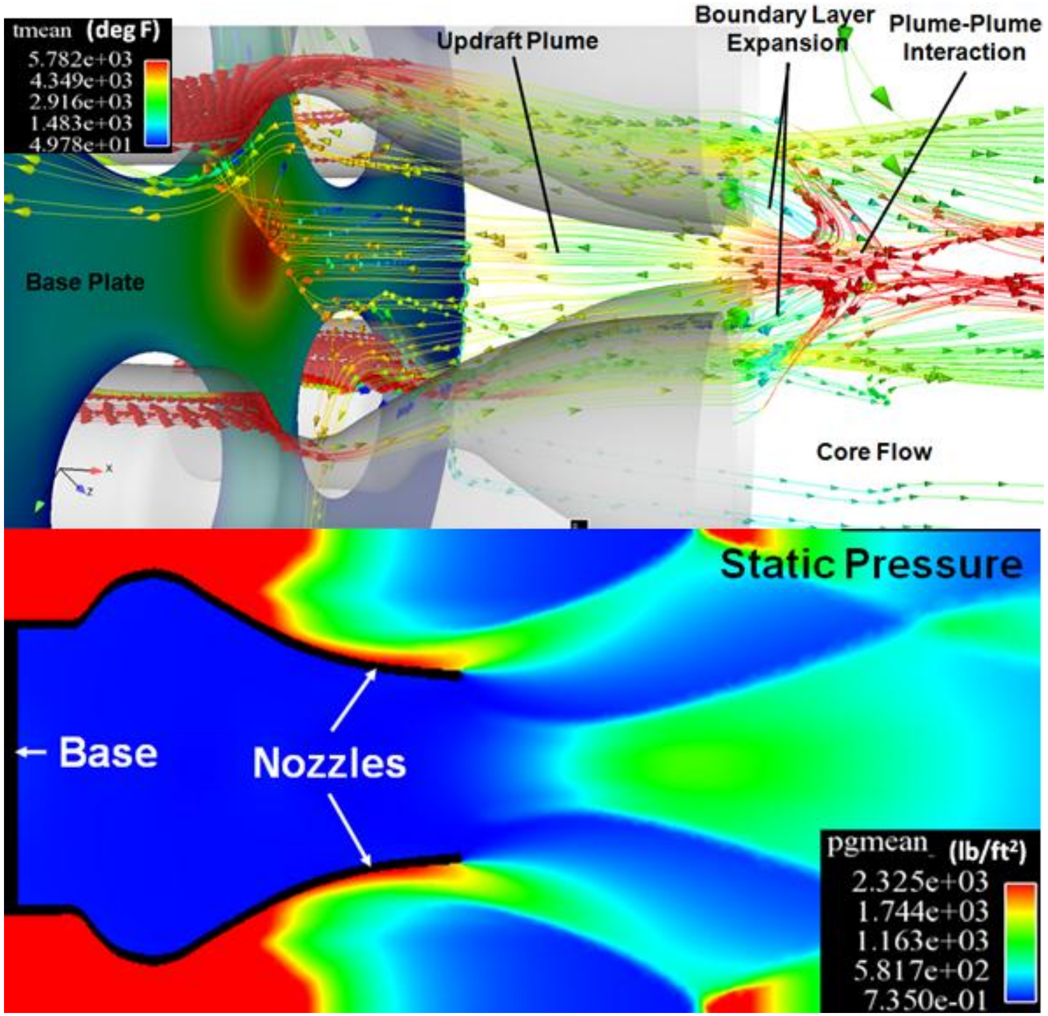


This proves that the nozzle boundary layer specific enthalpy is one of the main drivers in accurately predicting base heating

Details are provided by Mehta, M et al., AIAA JSR (2012) Numerical sensitivity study of 4 rocket engine core configuration, .

Nozzle Specific Enthalpy Flow Analysis

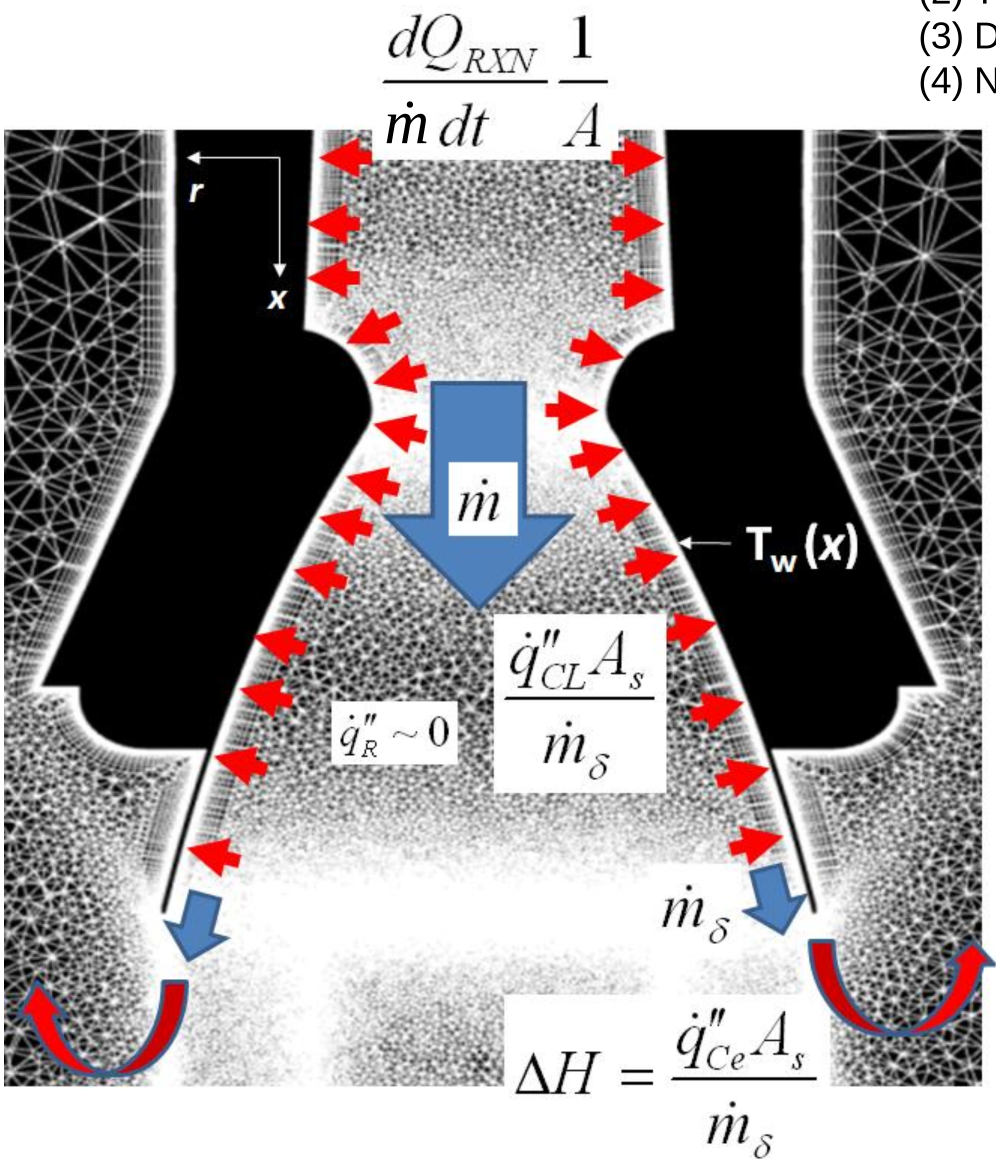
- These streamlines and pressure contours further show that the nozzle boundary layer determines the base environments especially within the recirculating regime.

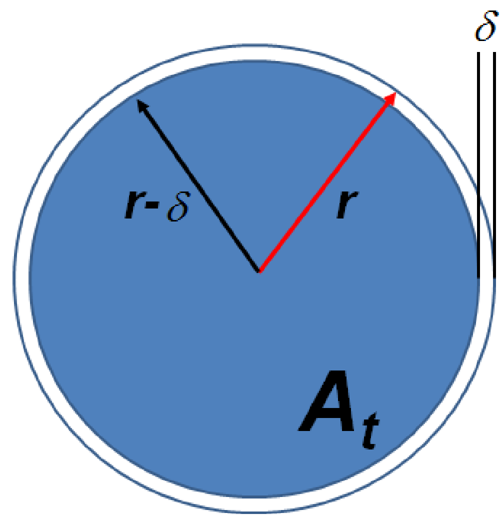


Details are provided by Mehta, M et al., AIAA JSR (2012) Numerical sensitivity study of 4 rocket engine core configuration, .

Nozzle Specific Enthalpy Flow Analysis

(1) Control volume approach
(2) Turbulent pipe-flow theory
(3) D.R. Bartz convective heat transfer theory
(4) Newton Law of Cooling





Nozzle Specific Enthalpy Flow Analysis

$$\frac{dQ_{RXN}}{dt}\left(\frac{1}{A}\right) - \dot{q}''_{CL} + \dot{q}''_{R} = \dot{q}''_{Ce} \qquad (1)$$

From Newton's Law of Cooling

$$\frac{dQ_{RXN}}{dt}\left(\frac{1}{A}\right) - h\left(T_R - T_W\right) + \dot{q}''_{R} = \dot{q}''_{Ce} \qquad (2)$$

$$\dot{q}''_{R} \sim 0$$

$$\frac{dQ_{RXN}}{dt} - hA_s\left(T_R - T_W\right) = \dot{q}''_{Ce} \qquad (3)$$

Derive $\dot{m}_{g}$

$$\dot{m}_{g} = \frac{\dot{m}}{A_t}\pi\left(2r\delta - \delta^2\right) \quad \text{(see Figure 3)} \qquad (4)$$

$$\delta = \frac{0.1794D}{G\ln(\mathrm{Re})}$$ Derived from fully-developed turbulent pipe flow; G(empirical constant) ~1.35

Divide by $\dot{m}_{g}$

$$\frac{1}{\dot{m}_{g}}\frac{dQ_{RXN}}{dt} - \frac{hA_s\left(T_R - T_W\right)}{\dot{m}_{g}} = \frac{\dot{q}''_{Ce}}{\dot{m}_{g}} = H_{Ce} \qquad (5)$$

$$\frac{1}{\dot{m}_{g}}\frac{dQ_{RXN}}{dt} = H_{RXN} \qquad (6)$$

Nozzle Specific Enthalpy Flow Analysis

$$H_{RXN} - \frac{hA_s(T_R - T_W)}{\dot{m}_g} = H_{Ce} = \Delta H \qquad (7)$$

$$H_{RXN} - H_L = H_{Ce} = \Delta H \qquad (8)$$

$$A_s = \int_{x_1}^{x_2} 2\pi\, y \sqrt{1 + \left(\frac{dy}{dx}\right)^2}\, dx \qquad (9)$$

$$\text{Nozzle Contour}: \; y = -ax^2 + bx + c \qquad (10)$$

Substitute Eqn 9 into Eqn 7

$$\Delta H = H_{RXN} - \left[\frac{(T_R - T_W)2\pi}{\dot{m}_g}\int_{x_1}^{x_2} h(x)\, y(x) \sqrt{1 + \left(\frac{dy(x)}{dx}\right)^2}\, dx\right] \qquad (11)$$

Subsitute D.R. Bartz convective heat transfer coefficient equation into Eqn 11

$$\Delta H = H_{RXN} - \left[\frac{(T_R - T_W)2\pi}{\dot{m}_g}\left[\frac{0.026}{D_*^{0.2}}\left(\frac{\mu^{0.2} c_p}{\Pr^{0.6}}\right)_0 \left(\frac{p_0}{c^*}\right)^{0.8}\left(\frac{D_*}{r_c}\right)^{0.1}\right]\left(\frac{A_*}{1}\right)^{0.9}\int_{x_1}^{x_2}\frac{\sigma_e(x)}{\pi^{0.9}(y(x))^{1.8}}\, y(x)\sqrt{1 + \left(\frac{dy(x)}{dx}\right)^2}\, dx\right] \qquad (12)$$

$$\sigma_e = f\left(\frac{T_W}{T_0}, M, \gamma\right) \qquad (13)$$

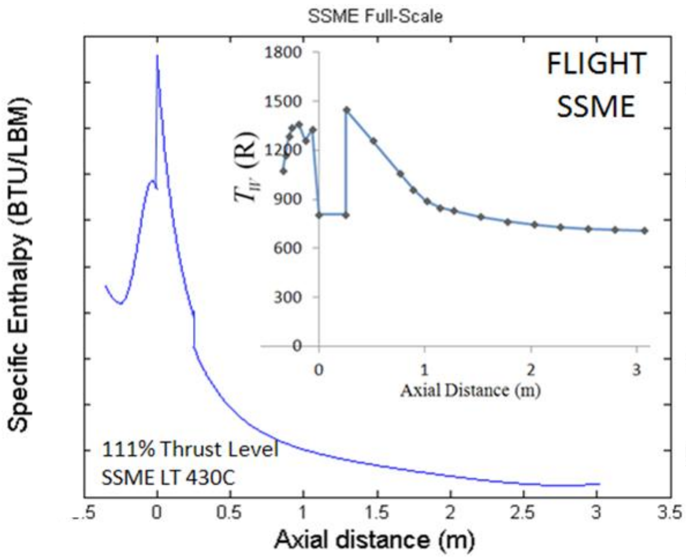
SSME and RSRM ΔH

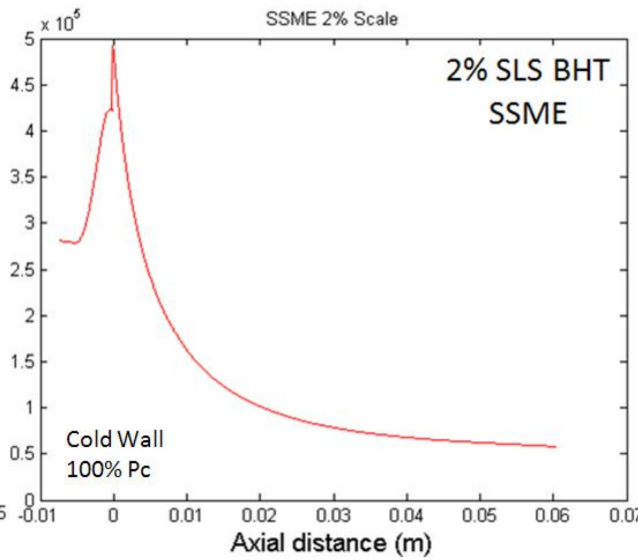
$$\Delta H = H_{RXN} - \left[\frac{2\pi}{\dot{m}_g}\left[\frac{0.026}{D_*^{0.2}}\left(\frac{\mu^{0.2} c_p}{\Pr^{0.6}}\right)_0 \left(\frac{p_0}{c^*}\right)^{0.8}\left(\frac{D_*}{r_c}\right)^{0.1}\right]\left(\frac{A_*}{1}\right)^{0.9}\int_{x_1}^{x_2}\frac{\sigma_e(x)[T_R - T_W(x)]}{\pi^{0.9}(y(x))^{1.8}}\, y(x)\sqrt{1 + \left(\frac{dy(x)}{dx}\right)^2}\, dx\right] \qquad (14)$$

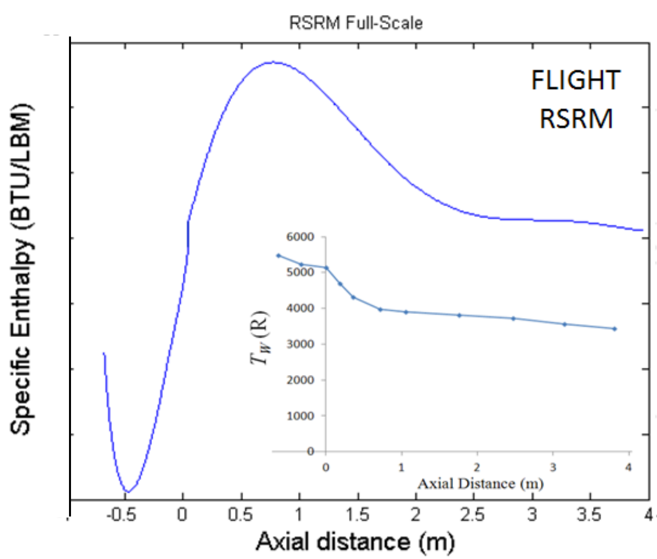
$$\Delta H' = 1 - \frac{H_L}{H_{RXN}} = \frac{H_{Ce}}{H_{RXN}} \qquad (15)$$

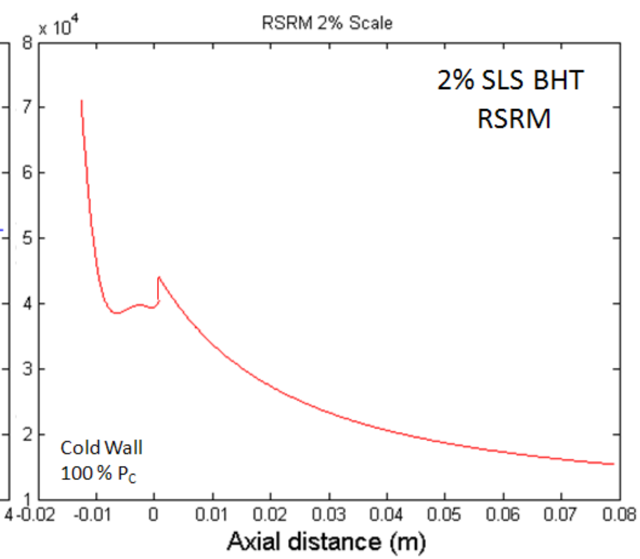
Nozzle Specific Enthalpy Flow Analysis

- Nozzle boundary layer specific enthalpy profile for the flight and 2% model RS-25D (SSME) and RSRM. SSME = Space Shuttle Main Engines





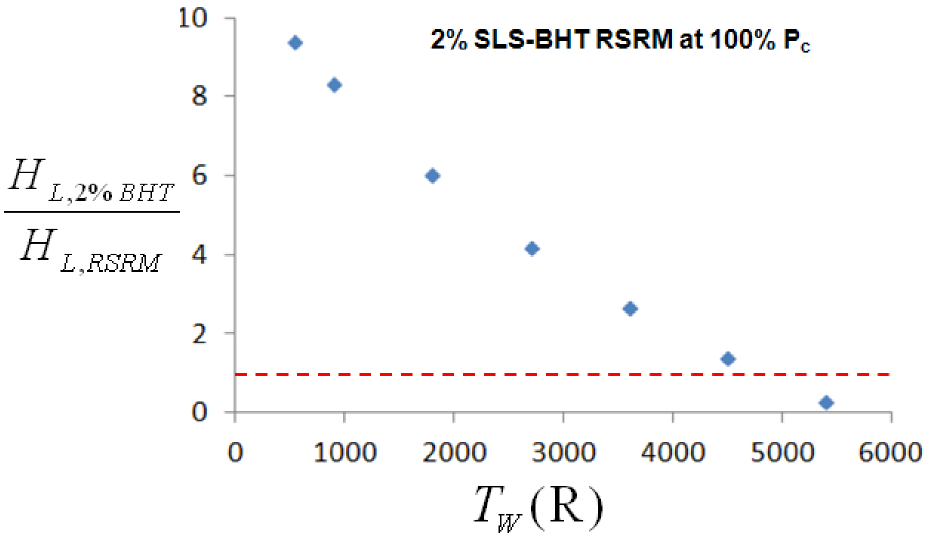


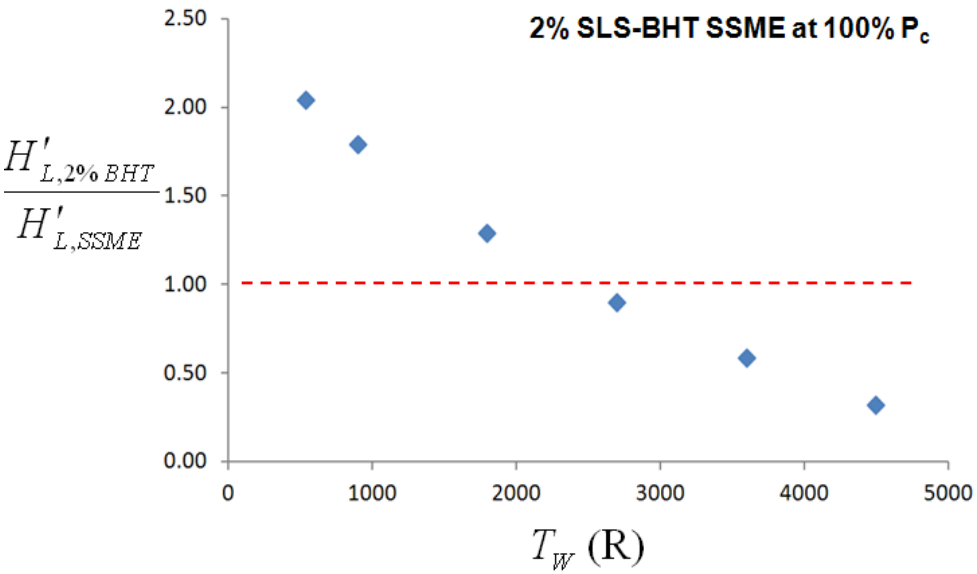


Nozzle Specific Enthalpy Flow Analysis

- Target wall temperature ~4700 deg R
  - 2% scale RSRM
  - Plume properties at t = 80 sec
  - T_W is the constant wall temperature

- Target wall temperature ~2100 deg R
  - 2% scale RS-25D
  - T_W is the constant wall temperature





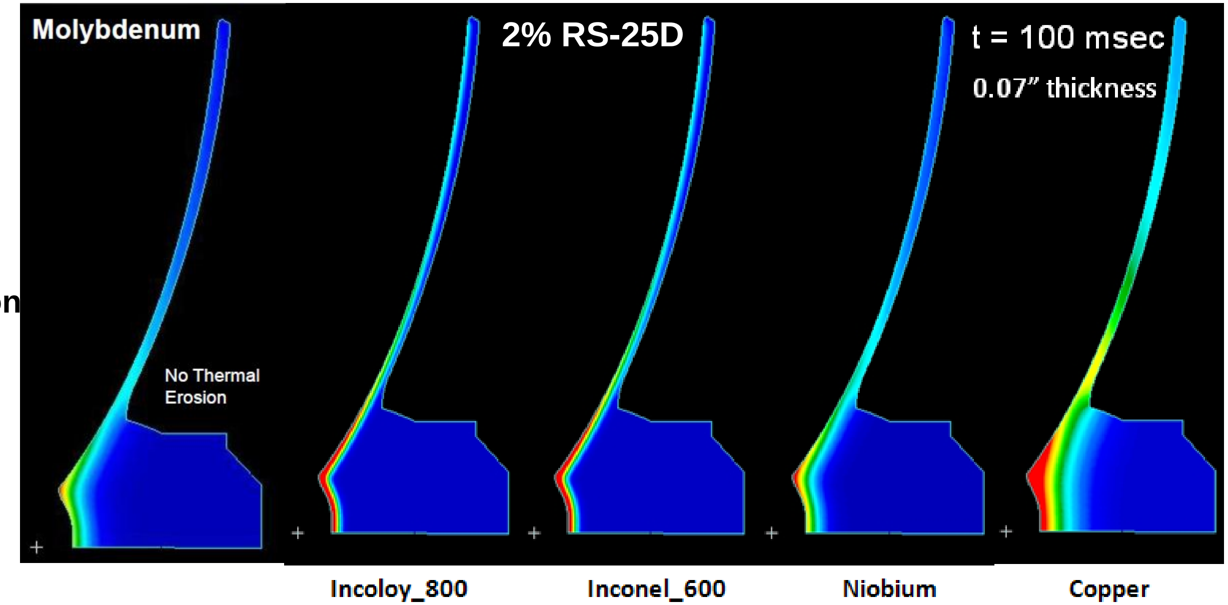
Nozzle Material and Design Selection

- Nozzle material and design selection are investigated for:
  - **Life-Cycle Study:** Which design and material can withstand the high heat rates at the nozzle throat?
  - **Specific Enthalpy Study:** Which design and material for the heat-sink methodology can provide similar nozzle exit specific enthalpy to the flight conditions?

- Nozzle material and design sensitivity study performed:
  - MSC Patran with SINDA/G thermal solver (FEA) are used to model a variety of nozzle materials and thickness for both the axisymmetric core-stage and booster nozzle elements

Nozzle Material and Design Selection

Failure = any nozzle geometric distortion due to thermal erosion

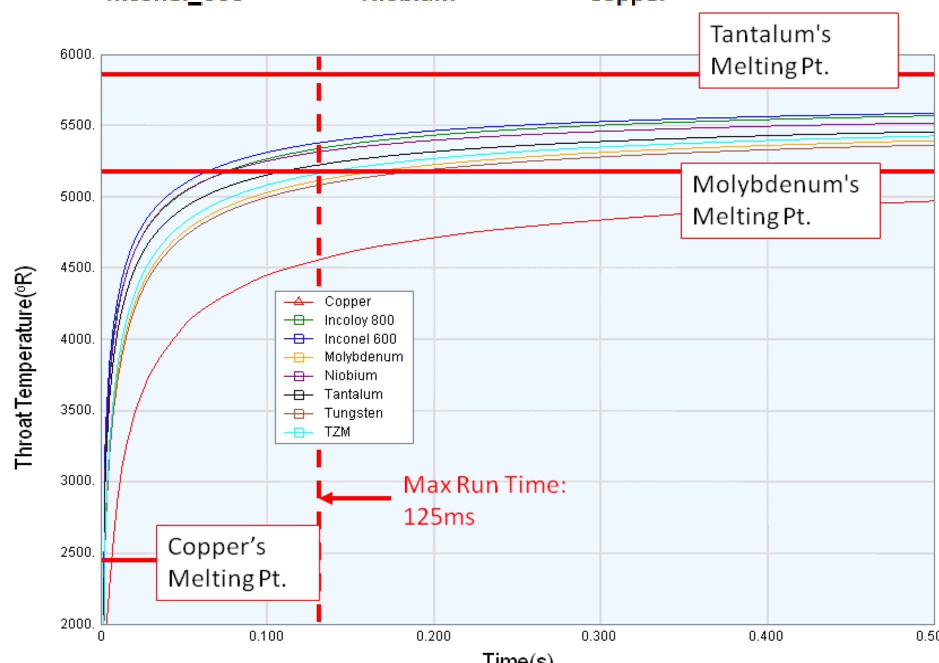


Failure Plots

Red contour is within 10% of the material's melting point

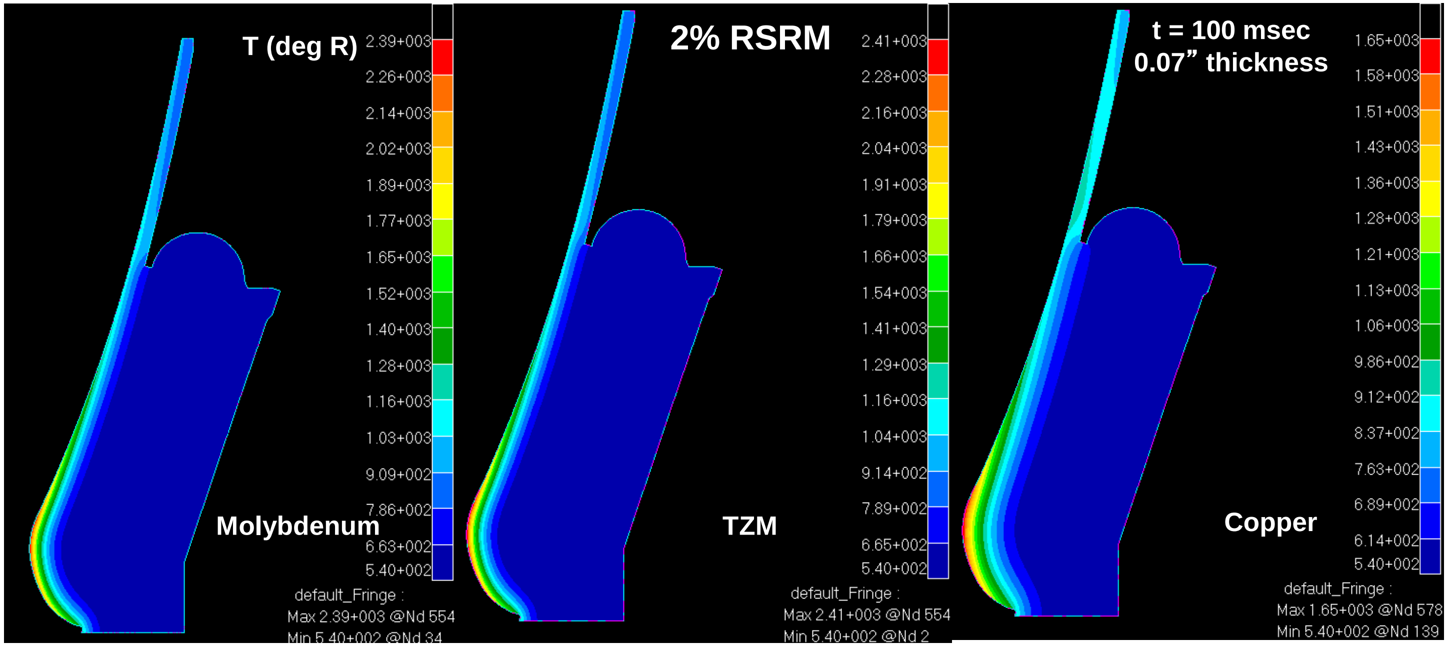
RS-25D 07" Nozzle Thickness		
Material	2% Peak Temp.(°R)	Melting Temp.(°R)
Copper	4158.44	2444.67
TZM	4942.13	5209.67
Molybdenum	4879.92	5219.67
Niobium	5145.27	4933.67
Tantalum	5029.24	5884.67
Tungsten	4832.48	6629.67
Inconel 600	5246.08	2929.67
Incoloy 800	5193.21	2934.67

Target Wall Temperature ~ 2100 deg R



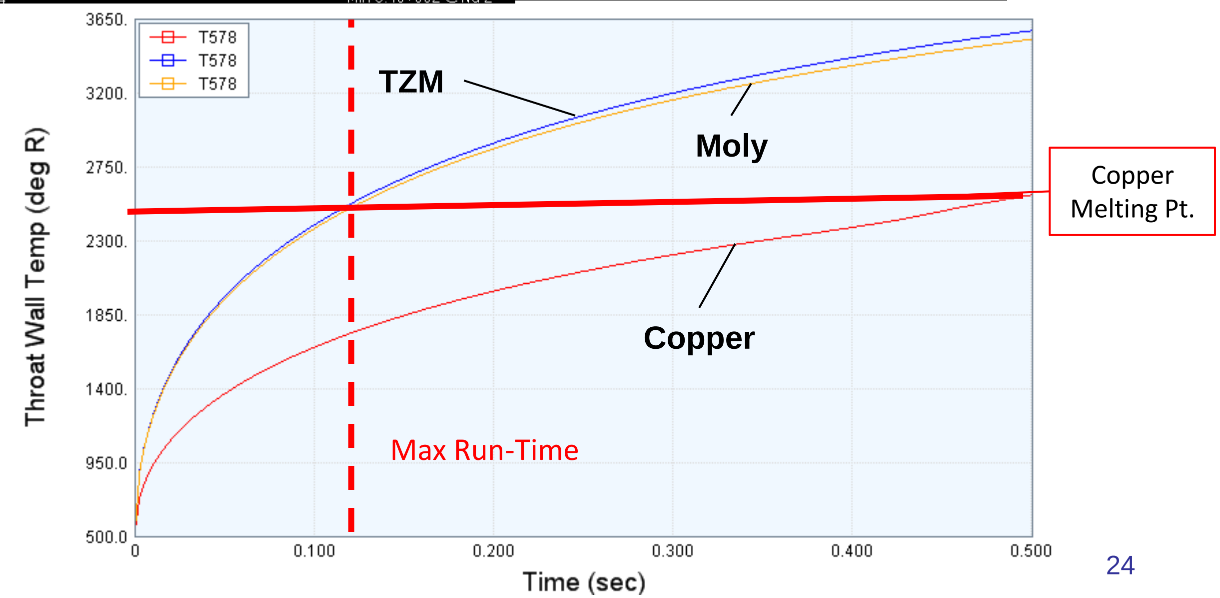
Nozzle Material and Design Selection

Material Temperature Contour (deg R)



RSRM 07" Nozzle Thickness		
Material	2% Peak Temp.(°R)	Melting Temp.(°R)
Copper	1650.00	2444.67
TZM	2410.00	5209.67
Molybdenum	2390.00	5219.67

Target Wall Temperature ~ 4700 deg R



Nozzle Material and Design Selection

- **Life-Cycle Study:**
  - Nozzle thickness of 0.07 inch should be designed
  - The following 2% RS-25D and 2% RSRM nozzle materials were successful from a life-cycle perspective:
    - Molybdenum
    - TZM
    - Tantalum
    - Tungsten (difficult to fabricate)
  - Incoloy, Inconel, Niobium and Copper showed material failure for the 2% RS-25 nozzle

- **Specific Enthalpy Study:**
  - Find that the metal and metal alloys for the RSRM are inadequate to meet the high surface wall temperature and nozzle exit specific enthalpy observed in flight
    - Need to investigate ceramic coatings, high temperature metal inserts with insulator backing, carbon graphite
  - Thermal FEA in conjunction with the specific enthalpy flow analysis will be performed
    - Axial temperature wall distributions for various time slices will be extracted from FEA and incorporated into the enthalpy flow code.
    - Determine which nozzle material and design and at what run-time will adequately simulate the nozzle exit boundary layer specific enthalpy for a short duration test to that of flight

Innovative Methods to Improve Test Fidelity

- Methodologies to increase test run-time
  - Convert LENS II facility into a Ludwieg Tube
- Matching the nozzle exit boundary layer specific enthalpy
  - Dependent on nozzle material
  - Dependent on nozzle wall thickness
  - Need Pathfinder test data to develop high fidelity analysis
  - Minimize scaling methods and improve data fidelity
- Running at 100% Pc values for both RS-25D engine and RSRM conditions
  - Dependent on facility capability
  - Dependent on propulsion component design
  - Minimize scaling methods and improve data fidelity
- Maintain steady chamber pressures for the RS-25D engine and RSRM
  - Dependent on chamber geometry and propellant properties
  - Dependent on test run-time, steady pressure needs to occur well within ~125 msec
- Thermography imaging/pyrometry techniques
  - New method to determine nozzle inner wall temperature distribution
  - Possibly provide base gas temperature measurements (?)
- Develop a more accurate gas temperature probe (GTP)
  - New innovative design method are currently being explored

Summary

- **MSFC SLS-Base Heating Test Working Group (SLS-BHT WG) has made good progress on the following:**
  - Test objectives and requirements definition
  - Test run conditions and matrix
  - Instrumentation layout (improvement)
  - Preliminary model design (improvement)
  - Nozzle boundary layer specific enthalpy flow analysis (innovative)
  - Dynamic similarity analysis
  - Nozzle material selection and design (innovative)

- **Future Added Work:**
  - Investigate pyrometry/thermography imaging techniques (innovative)
  - Select a quick-acting valve

Questions

- Acknowledgements:
  - Mark D’Agostino (MSFC Aerosciences Branch Chief)