NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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PASSIVE ROCKET DIFFUSER TESTING:
PERFORMANCE SIMILITUDE VIA NOZZLE CONTOUR MODIFICATION

By Daniel R. Jones

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WASHINGTON

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INTRODUCTION

Subscale rocket testing offers a chance to examine exhaust plume behavior and prove out auxiliary systems early in a development cycle, when the cost of design iteration remains low. Despite the utility of experimentation, perfect similitude between a rocket engine and a scaled model is unobtainable; choices must be made to prioritize the reproduction of certain parameters at the expense of others. The experimentalist's toolkit contains a variety of techniques to achieve this selective similarity. Determination of the appropriate tool is dependent on a test series' specific objectives, facility limitations, and financial considerations. Preservation of full nozzle geometry is crucial if its internal flow is of interest. However, subscale thrusters often serve as a simple plume source in the evaluation of adjacent phenomena: vehicle base pressures, retropropulsive flows, supersonic diffuser performance, refractory erosion, deflector cooling, impingement shock structures, etc. In such cases, it may be permissible to rework the nozzle geometry to compensate for discrepancies in subscale propellant pressure, temperature, or chemical composition. Thrust, pressure, Mach number, heat flux, and myriad other parameters can be used to ground the altered designs.

Subscale diffuser testing at the full chamber pressure of flight hardware is occasionally infeasible. It is hypothesized that, absent a means of backpressure reduction, comparable performance may be obtained for a prescribed motive fluid by maintaining scaled engine mass flow and nozzle exit geometry at the expense of throat geometry, exit pressure, and Mach number. The present work offers discrete empirical support for this mass-flow-based approach to contour modification.

NOMENCLATURE

P	Pressure	DI	Diffuser Inlet
0	Stagnation Condition	DT	Diffuser Throat
CC	Combustion Chamber	NE	Nozzle Exit
CELL	Test Cell	NT	Nozzle Throat

EXPERIMENTAL APPARATUS AND RESULTS

In 2015, a subscale thruster was needed to approximate the flowfield of an RS-25 throttled to 12.3 - 15.2 MPa for use in hot-fire diffuser performance evaluations [1]. The requisite H_2/O_X propellants were available at the stand but chamber pressure was constrained to a maximum of 7.9 MPa, well below the range of interest. Backpressure reduction was unavailable. The chamber pressure difference was counteracted by definition of a thrust-optimized parabolic (TOP) nozzle contour [2] to give the same mass flow, exit diameter, and exit angle as an RS-25 operating at the diffuser's geometric scale. Figure 1 shows the drift in nozzle exit conditions resulting from this approach. Full chamber pressures were considered to be 18.9 MPa and 9.9 MPa for the engine and thruster, respectively.

Extensive subscale testing was conducted using the thruster design, but the full-scale system was never constructed and hot-fire validation of the thruster's design was left unobtained. Fortunately, this approach to recontouring is first-order independent of motive fluid barring cases in which the differences between engine and model produce disparate multiphase or real-gas effects. Stated another way, the nozzles will produce similar plumes for any given ideal-gas propellant even if those plumes differ from one propellant to the next.

An opportunity for validation arose five years after the original subscale test campaign during an assessment of heated ethane as a simulant gas for rocket plume aerodynamics [3-4]. Lab-scale thruster and RS-25 nozzles were fabricated to test ethane's ability to reproduce hot-fire boundary layer separation and shock structures. A comparison of nozzle geometries is given in Figure 2. Each nozzle was coupled with an equivalently-scaled diffuser #3 from [1] and run with heated ethane and ambient nitrogen.

Normalized pumpdown curves from the lab-scale tests are plotted in Figure 3. Diffuser starting characteristics were similar between the thruster and RS-25 configurations taken on a per-gas basis, and the expected offset between propellants was also observed. Normalized steady-state wall pressures are shown in Figure 4 alongside Loci/CHEM [5-6] simulation results. Measurements were effectively identical in the test cell and diffuser inlet and similar shock structures were observed throughout the diffuser. Despite the broad similarities, the lower Mach number of the thruster flowfield produced lower stagnation pressure loss through the shock reflections and proportionally higher static pressures downstream in the diffuser.

CONCLUSION

A mass-flow-based approach to rocket nozzle contour modification has been experimentally demonstrated as an effective method of producing aerodynamically similar plumes at reduced chamber pressures. Errors incurred by this approach inevitably grow with the driving pressure discrepancy between engine and model. Nevertheless, the described diffuser testing has shown that the utility of this approach extends to a model-to-engine chamber pressure ratio of at least 0.52. The limits of its application remain unexplored.

REFERENCES

[1] Jones, D., Allgood, D. and Saunders, G. : "Passive Rocket Diffuser Testing: Reacting Flow Performance of Four Second-Throat Geometries," NASA TM 2016-219221, 2016.
[2] Rao, G.V.R. : "Approximation of Optimum Thrust Nozzle Contour," ARS Journal, Vol. 30, No. 6, 1960.
[3] Jones, D. : "Physical Simulation of Rocket Exhaust Aerodynamics Using Heated Ethane: Conceptual Foundations", NASA TM 2019-220446, 2019.
[4] Jones, D. : "Physical Simulation of Rocket Exhaust Aerodynamics Using Heated Ethane: Prototypical Experiments", NASA TM 2020-5009122, 2020.
[5] Luke, E., and George, T. : "Loci: A Rule-Based Framework for Parallel Multidisciplinary Simulation Synthesis," Journal of Functional Programming, Vol. 15, No. 03, 2005.
[6] Luke, E., Tong, X-L., Wu, J., Tang, L., and Cinnella, P. : "CHEM: A Chemically Reacting Flow Solver for Generalized Grids", AIAA 2003.

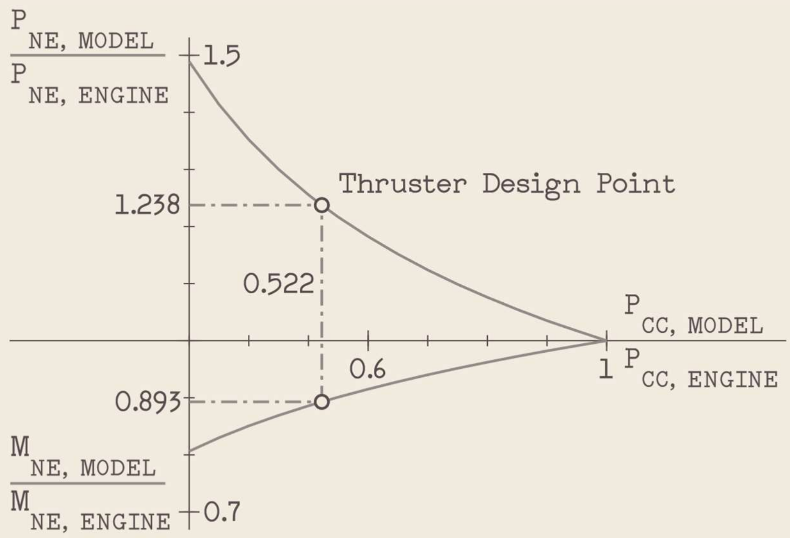

Figure 1 - Recontouring effects on quasi-1D equilibrium nozzle exit conditions.

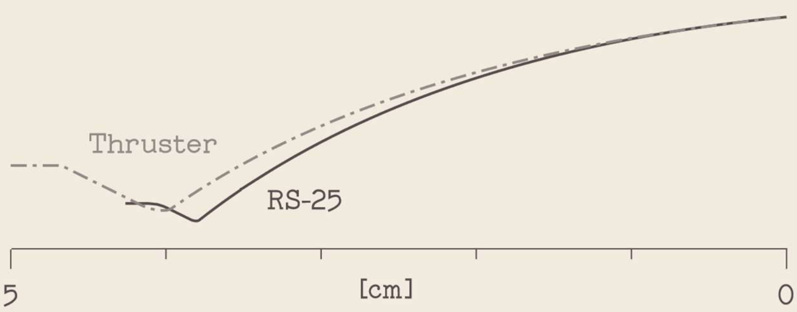

Figure 2 - Nozzle contours at lab scale.

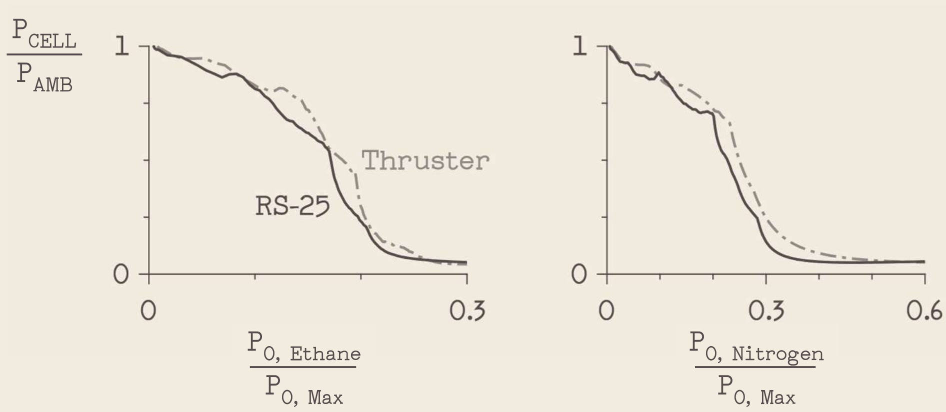

Figure 3 - Normalized ethane and nitrogen pumpdown data for each nozzle.

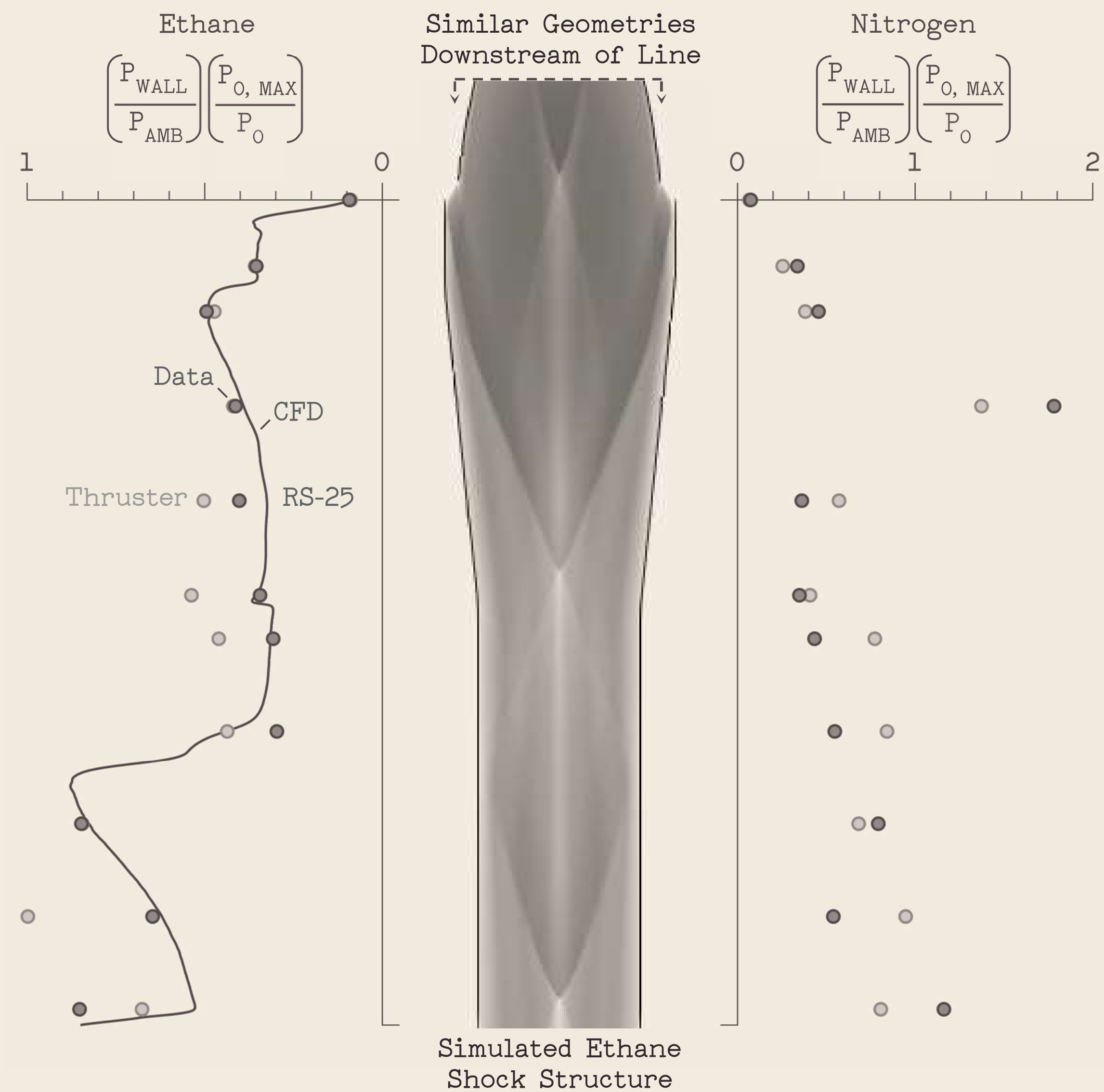

Figure 4 - Normalized steady-state wall pressure ratios.