NASA's Rotating Detonation Rocket Engine Development

Thomas Teasley[1]
NASA Marshall Space Flight Center, Huntsville, AL, 35808, United States

The Rotating Detonation Rocket Engine has maintained steady development at NASA with many staggering performance advantages demonstrated to date over the state-of-the-art (SOA). The implementation of additive manufacturing and specialized NASA developed alloys have enabled rapid maturation of the technology. Several hot fire test projects have been successfully conducted at Marshall Space Flight Center under an early career initiative project funded by NASA Space Technology Mission Directorate. In addition, a new start Technology Demonstration Mission (TDM) project has been funded to investigate challenges relating to integration of turbomachinery with an RDRE thrust chamber assembly. This engine system demonstration will leverage a methane/oxygen single shaft turbopump with fuel rich gas generator and a 10,000 lbf thrust chamber assembly. The configuration was down selected based on feedback from both US industry collaborators and power balance trades in combination with technical feasibility. To date, industry has identified several use cases for RDRE ranging from thruster to primary launch vehicle propulsion. A wide range of fuel and oxidizers were also identified including but not limited to Methane, Kerosene and other liquid hydrocarbon (LH) fuels, and hydrogen. Recent work at NASA and in partnership with NASA has investigated these major fuels of interest with oxygen, air, and hydrogen peroxide (HTP) for various applications. NASA Marshall has already investigated the use of hydrogen/oxygen, methane/oxygen, kerosene/oxygen, and has plans in partnership with industry and academia to investigate LH/air and LH/HTP. In addition to propellants, hardware geometry has been investigated with some critical lessons learned toward greater theoretical performance over the SOA. To this end, several experimental and computational activities are ongoing to further advance the RDRE towards flight missions. Given the rate of advancement, it is highly likely the technology will be flown in space mission in the coming decade. This work documents and overviews many of these investigations and overviews NASA's future plans for the technology maturation.

[1] Liquid Propulsion Systems Development Engineer, Engine Component Development and Technology Branch

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I. Introduction

NASA has continued to develop the rotating detonation rocket engine (RDRE) for use with numerous applications including launch, lander, upper stage, supersonic retro propulsion, and hypersonics. An Early Career Initiative project (ECI) has successfully tested multiple fuels and hardware scales / geometries to date. Several major lessons learned on how this technology should be designed and operated have been documented [1], [2], [3], [4], [5], [6]. A significant focus has been on understanding the heat transfer associated with detonative combustion. [1], [7], [8] Many of the pivotal tests and test projects are shown in Fig 1.

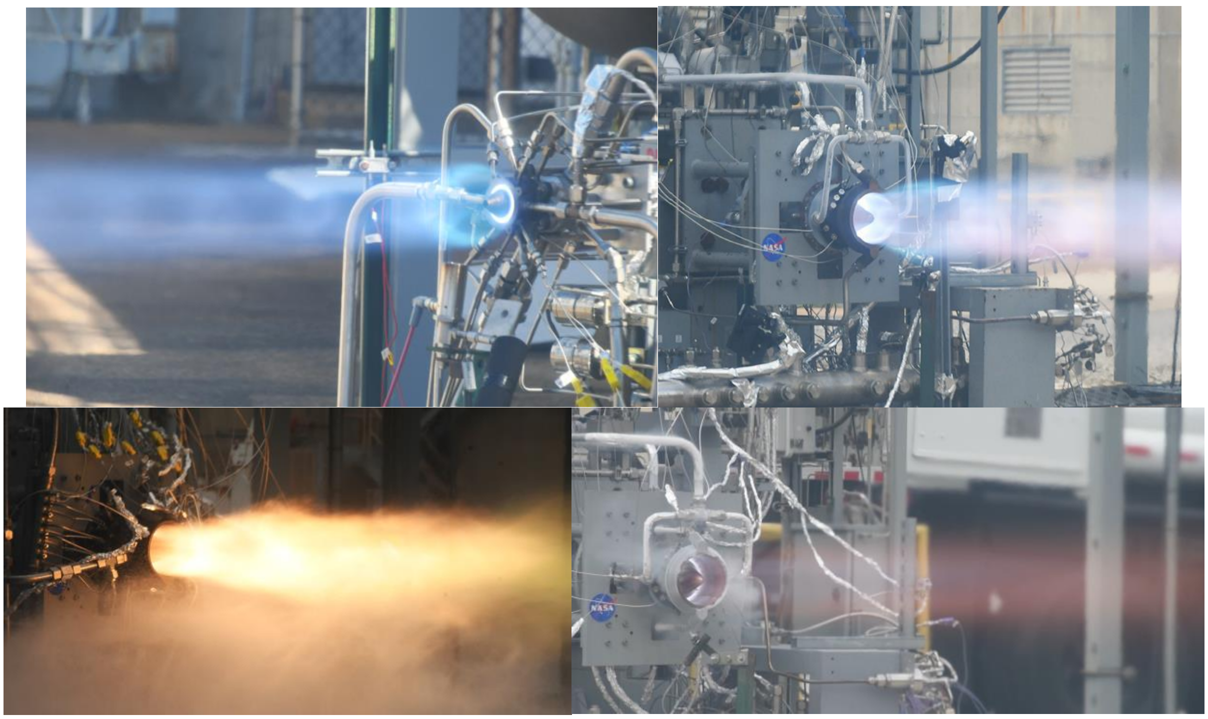

Fig 1. NASA RDRE pivotal hot fire tests; high pressure methane/oxygen at 750 psia (top left), high thrust liquid methane/oxygen test (top right), direct injection kerosene/oxygen (bottom left), and dual regenerative liquid hydrogen/liquid oxygen (bottom right).

Much of this work has been conducted with specific goals or key areas of understanding that needed to be elucidated. A great deal of the focus for NASA has been on assessing “flight realistic” operating conditions and propellants. The use of cryogenics and high-pressure operation are one example of this. The following section gives an overview of the current technology maturation approach along with critical technology elements of interest.

II. Technology Maturation Approach

A major goal of the technology maturation plan is to use rapid experimental development strategies in combination with computational approaches to methodically identify key design and operability requirements. This development strategy allows for roughly ~80+% of the theoretical performance potential for the RDRE to be obtained all the while providing industry, academia, and government partners with regular updates on successes. This ~80% mark is not necessarily a hard target but rather a rule of thumb. It is seen as high risk for US industry to develop such a radical technology when its TRL is still relatively low. This 80% goal would in theory push the technology to a reasonable TRL of potentially 5 or greater allowing industry to invest fully at lower risk.

As it stands, the RDRE has already shown many performance advantages and NASA has identified several critical design strategies, limits, and requirements that is freely open to US collaborators should they request it. Ongoing efforts are being made to rapidly iterate on varying scales of RDRE currently in operation at MSFC. Most importantly, challenges in operability have been resolved with explicit design features and this work seeks to elaborate on the technology maturation plan along with key data that will be obtained in the near future. First, an

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RDRE can be divided into subsystems / subcomponents or critical technology elements (CTEs). A general schematic of the CTEs for an RDRE are shown in Fig 2. Each CTE requires its own specific manufacturing process or sets of processes, specific alloys, post processing techniques, and test objectives.

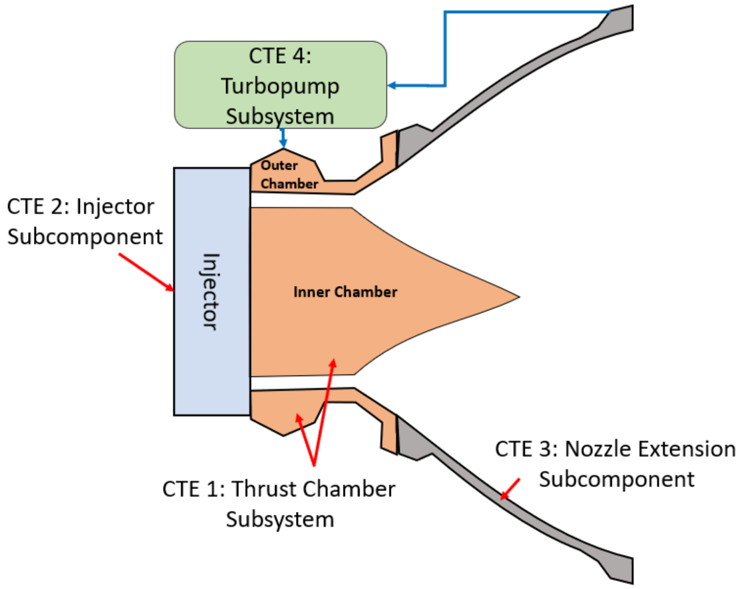

Fig 2. Schematic of RDRE critical technology elements (CTEs).

Each CTE subsystem may also include its own set of subcomponents or is, in and of itself, a subcomponent. Each CTE has its own specific description and is outlined below.

CTE-1: The thrust chamber and cooling subsystem is comprised of an outer body and inner body which are their own unique chamber geometries. Each are designed with a specific coolant channel geometry required to cool the hot wall exposed to combustion products. Each also has a unique coolant type flowing through the integrated coolant channels and are designed to a specific heat flux profile that is unique to that coolant, combustion process, and hot wall geometry. These data are iteratively obtained through multiple experimental efforts and scaling analyses. For example, fuel may be routed through the outer body. A proportion of, or all of that fuel can then be routed through the nozzle extension before being utilized elsewhere in the full system assembly. This could be a turbine, routed directly into the injector, or a proportion sent to a burner. Similarly, the inner body's oxidizer coolant passes through the chamber before being routed back towards the injector, a turbine, or a burner. Each subcomponent will require the use of laser powder bed fusion (L-PBF) GRCop-42 alloy to survive the anticipated extreme environments and manufacture the complex integrated structures. No other material, alloy, or process is known to survive these environments at the desired operating conditions. With L-PBF, the coolant channels will be produced with a high surface roughness thus they will also require post process chemical mechanical polishing (CMP). Otherwise, high pressure losses will incur. Similarly, the hot wall of the outer and inner bodies will need to be polished using conventional abrasive media. This will minimize heat transfer to the hot wall and maximize survivability of hardware.

CTE-2: The injection subcomponent will be a single piece L-PBF print using either GRCop-42 or GRX-810 depending on propellant requirements and expected pressure loads within the integrated manifolds. This will minimize development time, cost, and uncertainly of orifice discharge coefficient. Depending on the configuration and integrated design requirements, the injector may be coupled with the inner body in a single print. In this case, the orifices will need to be undersized from their desired effective flow area so that when the integrated coolant channels are polished using CMP they are opened up to their final design effective flow area. The injector face will also need to be finished using abrasive polishing to minimize heat transfer.

CTE-3: The nozzle extension subcomponent is comprised of a monolithic L-PBF or directed energy deposition (DED) printed hot wall and coupled conventionally machined inlet and/ or exit manifolds. The manufacturing process will depend on the thrust class which ultimately determines the scale of the nozzle extension. Should the

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nozzle extension exceed a specific build capacity, then DED will be the only viable production method if rapid development is desired. Other production methods such as braze welded tubes or slotted channels from stock material are not being considered due to increased cost and schedule requirements. In addition, the selected metal alloy will be required to have specific thermophysical properties to withstand the extreme heat flux environments. GRX-810 will be a desired material for thrust classes lower than 30,000 lbf. This is predominantly driven by build box limitations in existing L-PBF print platforms where GRX-810 can be produced. For thrust classes greater than this limit, alternate alloys will have to be considered such as NASA HR-1 or Inconel 625 coupled with the DED process. In all cases, a proportion of fuel will be flowed through the nozzle extension originating from either the pump exit or the outer body channel exit. That fuel is then routed to a turbine or burner depending on the engine system cycle.

CTE-4: The turbomachinery subsystem is comprised of multiple subassemblies and subcomponents. Some of these may include the pump assembly, the shaft assembly, the turbine assembly, the engine controller, and a pre or post burner assembly. For all subassemblies, the typical flow path of propellant, coolant, or drive gas is as follows; propellant is pumped up to high pressure and feeds the chambers integrated coolant channels. In the case of an RDRE, both fuel and oxidizer will need to be used due to the extreme heat fluxes. This is achieved by using two separate pumps coupled to the same or separate drive shafts. The drive shaft is driven by a turbine assembly which is often coupled to a set of bearings and various sealing interfaces. In some cases, an inter-propellant seal is used to separate fuel from oxidizer on a common shaft system. Conditioned propellant is then used to drive the turbine assembly which closes the loop. Alternatively, a pre or post burner assembly takes a portion of fuel and oxidizer, combusts it, and pushes those combustion products at low temperature through the turbine assembly. This burner assembly is divided into two subcomponents an injector and a reaction duct or combustion chamber. Various manufacturing techniques will be incorporated including conventional machining and additive manufacturing. Specific metal alloys will be required depending on the subassembly. High temperature alloys will be required for the turbine and burner assemblies, while high strength and oxygen compatible alloys will be required for the pump assembly.

As can be gleaned from the above, there have been a number of design challenges in the development of the RDRE. The integration of coolant flow paths has been uniquely challenging due to the truncated nature of the annular chamber.

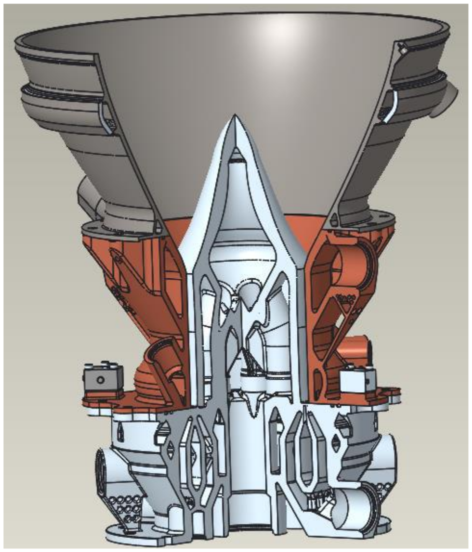

Fig 3. Example of preliminary hardware internal flow path illustrating complexity.

The following section gives an overview of just some of the experimental activities, limitations in scaling, and studies needed to understand how to optimally design an RDRE.

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III. Experimental Activities

Many of the key lessons learned have been documented and presented at appropriate conferences such as JANNAF. Several more will be documented and published over the next year. Each of these activities are intended to gain knowledge and understanding of how to ideally design an RDRE of any scale. To that end, two parametric scales of RDRE have been developed and are shown in Fig 4.

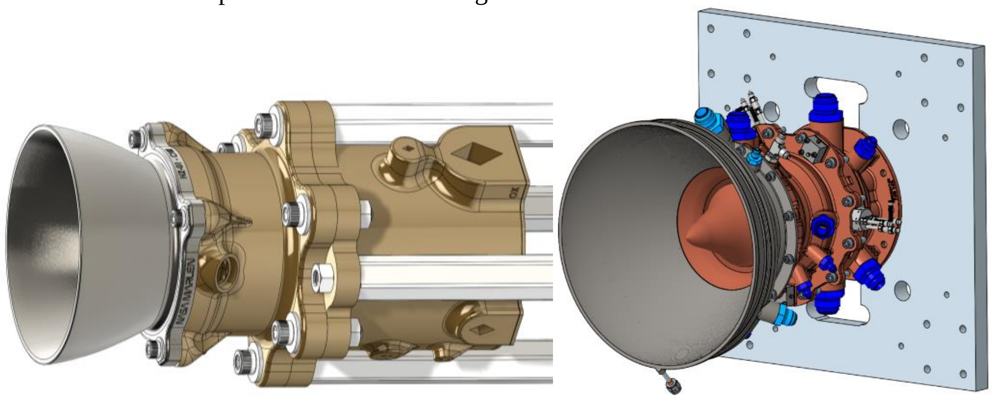

Fig 4. MARLEN subscale RDRE (left) and SWORDFISH full scale RDRE (right).

The MARLEN hardware is a cost-effective parametric platform by which NASA engineers can rapidly change out components to investigate their impacts on global performances such as wall heat flux and Isp. The SWORDFISH hardware is NASAs full scale 10K lbf platform that relays many of the key lessons learned from the subscale work and enables direct scalability comparisons to be conducted. Many experimental activities have been conducted on both platforms to date with major achievements documented in the following sections. More importantly, specific design parameters have been identified as being unimportant or critical to the operation of the RDRE. The following gives an overview of experimental activities of high importance for the technology maturation pathway of the RDRE. It explicitly evaluates impacts of different CTE design trades.

1. Thrust class and scalability.
2. Propellant and conditioning.
    a. Propellant type.
    b. Phase at injection.
    c. Temperature at injection.
    d. Pre-burned condition.
3. Annulus Geometry.
    a. Wall contouring.
    b. Annulus gap width.
    c. Area Contraction.
    d. L* and L'.
4. Injection parameters.
    a. Element scheme.
    b. Element density.
    c. Injection pressure loss or momentum balance.
    d. Orifice geometry.
    e. Manifold acoustics and coupling.

These four overarching areas of interest have been or will be further investigated to understand their roles in global engine performances including wave topology effects, heat transfer effects, and Isp.

Many aspects of the studies outlined above require the assessment of scalability such that any thrust class engine can be developed. As such, all studies included need to be validated at acceptable thrust classes to the best of the ability of the development plan. Fig 5 shows a plot of the relative initial uncertainty in scaling to higher thrust classes with validated and unvalidated cases including confidence interval.

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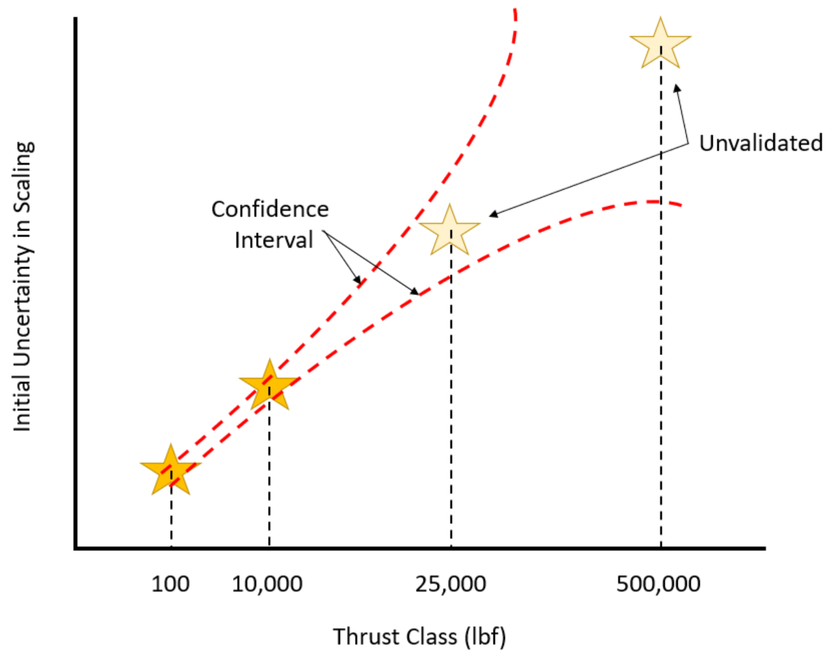

Fig 5. Representative plot of initial uncertainty in scaling with engine thrust class.

Should an additional thrust class be validated the confidence interval will collapse and reduce the gap in confidence. Many researchers also anticipate that higher overall performance gains may be achieved at larger engine thrust classes due to the relative increase in cooling capability, reduction in relative manufacturing constraints, and manufacturability of hardware. The confidence in relative manufacturability may become inverted as the thrust class becomes larger and larger. A plot representing this confidence as a function of thrust class is shown in Figure 6.

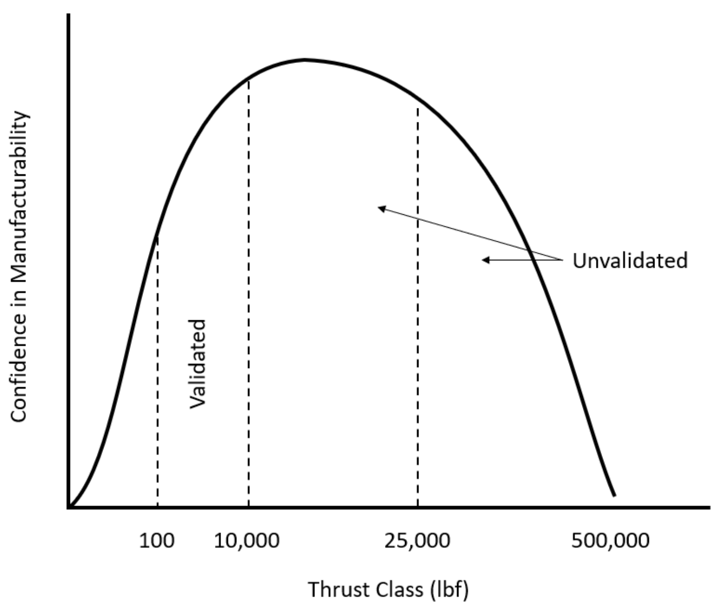

Figure 6. Confidence in manufacturability as a function of thrust class.

While something as large as a 500K design may not be necessary or useful in the near term, development and demonstration of a 25K class engine would yield a third data point for validation. This would elucidate a great number of scaling relations currently in need of validation.

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A. Major Accomplishments to Date

It has been found at both full scale and subscale that single wave operation imparts a substantial vibratory load on hardware. To illustrate this, plots of single wave and 2-wave mode thrust traces are shown in Fig 7.

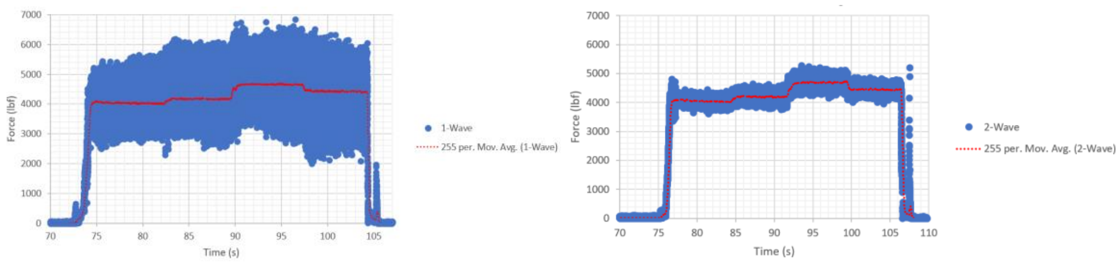

Fig 7. Single wave thrust trace (left) and 2 wave thrust trace (right).

The single wave mode imparts an unbalance force on hardware and has been shown to destroy interfaces, back out bolts, and shatter metal seals.

Fig 8. Damage of hardware from single wave operation.

This finding is critical to designers since flight hardware would not likely survive this operating point. Thus, future designs must implement strategies by which multi-mode operation is achieved even at throttled conditions.

Another major achievement includes the demonstration of detonations at high-pressures. Many applications such as launch and in-space transport may require high performance and thus high average pressure operation. However, it was previously thought that high pressure would cause deflagration to dominate, and all performance gains lost. This was found not to be the case with the subscale MARLEN hardware which was operated at 1250 psia for 5 seconds with methane/oxygen. Two strong detonations were observed in both the microphone PSD and high-speed video during this test. An image and test data are shown in Fig 9 and Fig 10, respectively.

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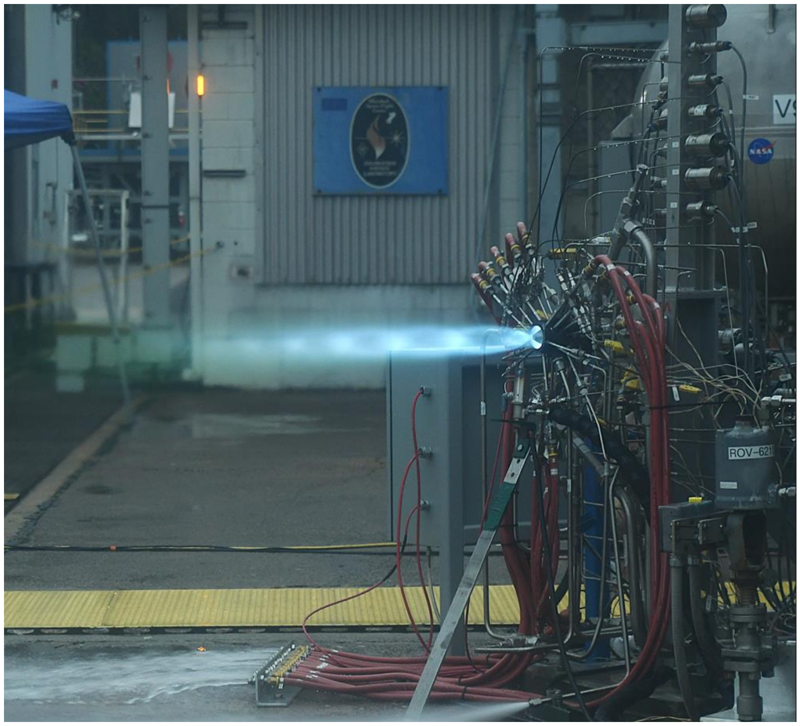

Fig 9. High pressure demonstration at 1250 psia with detonation modes.

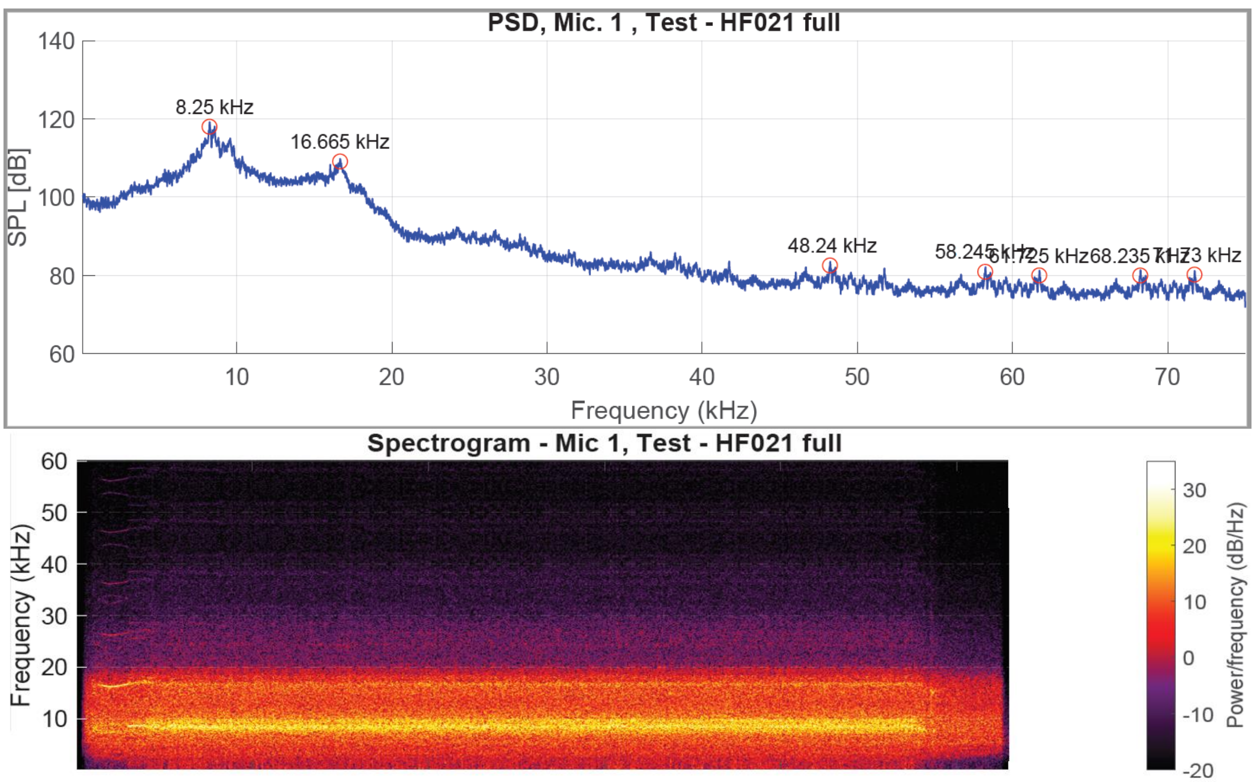

Fig 10. Power Spectral Density (top) and Spectrogram (bottom) of hot fire test 021 for MARLEN subscale pressure study.

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Furthermore, reduced injector pressure losses well below choked condition were demonstrated with wave activity observed in ranges from 700 psia to 1250 psia. These pressure losses were anywhere from 100 psid to 415 psid. There were conditions in which injector pressure losses were so low that waves were no longer observed, however chugg or other instabilities were not experienced. Further investigation into reduced injector pressure loss operation will be conducted in 2025.

Nozzle geometry dependency was also investigated at both full scale and subscale. An image of several radiatively cooled nozzle profiles are shown in Fig 11.

Fig 11. Subscale C-103 RDRE nozzle extensions.

These nozzles, and several others, have been investigated for variance of length, exit half angle, inlet half angle, and exit diameter. Length, or truncation was among the most important parameters of interest since a major feature of interest for RDRE is the overall benefit in reduced length.

Finally, a direct understanding of how different fuels (with Oxygen) impact operability has also been investigated. In general, methane compared with Kerosene yields a similar number of waves for the same injector design. However, the wave speeds are different which is likely impacted by combustion kinetics and mixing properties of the injector. Hydrogen, however, yields a vastly different number of wave at low pressure, and pure deflagration at higher pressures exceeding 250 psia CTAP or so. This is likely a design related problem where injector or annulus geometry promotes deflagrative losses. With that said, the measured combustion efficiency and Isp were similar to what would be expected with traditional steady flow rockets at the same conditions. It may be that a deflagrating hydrogen/oxygen annular rocket may benefit from the compactness of the design space in the near term while yielding reasonable performances. Further investigation is needed to assess the viability of annular deflagrating rockets and detonability of hydrogen/oxygen at elevated pressures.

IV. Summary and Conclusions

Much of the work conducted to date has been to reduce overall risk in the development of RDREs for adaptation by industry. Several findings and critical design criteria have been overviewed and will further enable the use of the RDRE by industry. A short discussion of the technology maturation plan was given with remarks of studies and investigations needed to further advance the TRL of RDRE. Each of which will be investigated going forward under the TDM program.

Experimental activities conducted in FY24 have found several critical design considerations. First, a single wave imparts a substantial vibratory environment on hardware, in some cases, an order of magnitude greater than the mean. Mode transition to 2 or more waves shows substantially reduced and more manageable environments. High pressure and reduced injector pressure loss operation has also been demonstrated. A test achieving 1250 psia at pressure drops of about 350 psid were demonstrated with detonation waves present. It cannot be understated how important of a demonstration this test was. It represents a major early milestone for the TDM project and demonstrates viability of the RDRE at high pressure and reduced injection pressure losses. Nozzle geometry was

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also investigated for compactness among other parameters. Reduced length nozzles were tested with no noticeable reduction in overall performance. Finally, different propellants have been investigated for scalability and operability. Methane and Kerosene were found to be similar in operability while hydrogen was found to be more challenging at elevated pressures as it preferred to deflagrate. Hydrogen/oxygen, while difficult to detonate at high pressure, may be viable in the near term for deflagrating annular rockets. Similar performances to traditional steady flow rockets were achieved but with the advantage of compactness that an annular RDRE affords.

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