AIAA Propulsion and Energy Forum
19-22 August 2019, Indianapolis, IN
AIAA Propulsion and Energy 2019 Forum

10.2514/6.2019-4028

Preliminary Results of a Micropump for MON-25/MMH Propulsion and Attitude Control

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The paper presents the development and protoqualification of two classes of high-power density MMH/MON-25 micropumps for spacecraft propulsion systems. The two thrust levels targeted are in the 5-7 lbf and 100-200 lbf classes, for use in both primary propulsion and attitude control systems (ACS). Micropumps can serve as enhancers for small, compact, high performance propulsion systems, replacing traditional pressure-fed systems for a decreased system mass, or increased ΔV capability. In hypergolic bipropellant systems, utilization of MON-25 reduces the freezing point of the oxidizer from -9 °C to -55 °C, such that the low freezing points of both propellants (MMH/MON-25) extends their potential use to environments with very low temperature demands. Hot fire tests with hypergolic propellants and a flight-like thruster were successfully conducted with the larger 200 lbf class pumps. 1000 ms and 3000 ms steady state burns and pulsed mode operations of the thruster were demonstrated with ambient temperature MMH/MON-3. Pump-fed steady state and pulsed mode hot fire testing with propellant temperatures of approximately -30 °C was also accomplished, thus demonstrating the technology to lower temperature operations.

I. Nomenclature

ACS	=	attitude control system
HAN	=	hydroxyl ammonium nitrate
L	=	volume, liters
ml/min	=	flow rate, milliliters per minute
MMH	=	monomethyl hydrazine
MON-3	=	3% mixed oxide of nitrogen
MON-25	=	25% mixed oxide of nitrogen
N	=	force, Newton
NTO	=	nitrogen tetroxide
psid	=	pressure differential
RPM	=	revolutions per minute
W	=	power, Watts
ΔV	=	change in velocity

II. Introduction

Micropumps can serve as either enhancers or enablers for small, compact, high performance propulsion systems. Traditional pressure-fed propulsion systems can be replaced by micropump-fed systems to provide decreased system mass and finer impulse-bit control through throttleability (without sacrificing maximum control authority). Most space engines today use hypergolic bipropellants, with MMH for fuel and MON-3 for the oxidizer, i.e. with 3% NO, which

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freezes at -15 °C. However, a MMH/MON-25 propellant combination allows operations over a wider range of temperatures, particularly in extremely cold environments as those envisioned for many future deep-space missions and to the moon. MON-25 is a mixed oxide of nitrogen containing 25 % nitric oxide (NO) with the balance being nitrogen tetroxide (NTO, N_2O_4). The additional NO reduces the freezing point of the oxidizer from the freezing point of MON-3 at -15 °C to -55 °C , closer to the MMH freezing point of approximately -51 °C [1] [2]. The reduced freezing points of both propellants extend their potential use to environments with greater temperature extremes. Missions that require prolonged exposure to extremely cold environments can be more easily achieved using MMH/MON-25without requiring the additional mass for propellant heating systems. The increased percentage of the NO in MON-25, however, significantly increases the vapor pressure of the oxidizer (~75 psia at 25 °C) [2]. As a result, a MON-25 oxidizer would be typically considered for use in pressure regulated system, which increases the need for pressurant and associated high pressure propulsion components, thus increasing total system mass. A pressurized system also requires high verification costs and adds operational risks associated with high pressures.

Miniaturization of the components in a high-pressure system can present challenges when scaled down to smaller satellites, such as CubeSats and other nanosats (tens of kg and less) [3]. These smaller spacecraft are typically extremely volume and mass limited, therefore conventional pressure-fed systems can be difficult to adapt to these platforms while maintaining the ability to conduct any significant capability to perform orbit change/control and attitude control maneuvers. Mass and volume requirements for a pressure-fed system often mean that these small satellites typically must forgo the use of a propulsion system altogether, or make use of a system with lower total impulse and thrust levels, such as a conventional cold-gas thrust system (typically < 80 seconds of specific impulse) [4]. Utilizing a micropump-fed system, the benefits of using the MMH/MON-25 bipropellants can be realized without sacrificing significant mass to the propulsion system in many applications. The micropump significantly reduces, or completely eliminates the need for a propellant pressurization system and associated components. A micropump decouples the propellant storage pressure requirement from thruster inlet feed pressure requirements, allowing for medium-pressure tanks to be used (now only limited by the MON-25 vapor pressure requirements in worst case conditions). When used with the heritage oxidizer MON-3, very low pressure systems and the potential use of conformal propellant tanks can provide significant mass savings, particularly beneficial for missions requiring large ΔVs.

A. System-Level Performance Improvements and Enablers

Previous trade studies were performed for propulsion system sizing mission analysis. Three propulsion system cases were considered: regulated pressure fed, pump-fed with pump power provided by the battery, and pump-fed with power provided by the solar array. In the pump-fed cases, the systems still include a pressure-regulated system in order to maintain the (lower) tank pressure above the MON-25 vapor pressure at the maximum expected operating temperature and the same scheme used for the fuel as well. For the pump-fed systems, the corresponding percentage increase in the payload is shown in Fig. 1. The solar-powered pump slightly outperforms the battery-powered pump since the former has slightly higher power density than the latter at the assumed 1 AU (Astronomical Unit). For example, a lunar lander requires approximately $\Delta V = 3000$ m/s; in that case, the "propulsion payload" increase is 16% for the battery-powered pump system and 19% for the solar-powered system. Here, "propulsion payload" includes all non-propulsion related spacecraft bus components as well as the scientific instruments.

For a single stage MON-25/MMH (assuming $I_{sp} = 300$ s) Mars Ascent Vehicle (MAV), which might require a total ΔV of 4000 m/s, with a vehicle wet mass of around 2000 kg, the propulsion payload mass is increased by 78% when using a battery for the pump. For a sample return mission, where the sample might account for a small fraction of that "payload", much of the additional mass could be allocated to increasing that sample size. Alternatively, for the same payload, the MAV wet mass can be reduced by 78%. For a lunar lander such as that envisioned for the candidate New Frontiers Lunar Geophysical Network (LGN) mission [5] [6], instead of using a 2-stage configuration (solid stage for deorbit and MON-25/MMH stage for landing), a single MON-25/MMH stage could be used for the total ΔV of approximately 3,100 m/s while also slightly increasing the scientific payload [7]. By comparison, as shown in Table 1, the mass budget for a pressure-fed system does not come close in a single stage configuration. Similarly, for another New Frontiers candidate mission, the Trojan Asteroid mission, replacing the pressure-fed chemical dual-mode hydrazine/MON-3 system by a battery-powered, pump-fed medium pressure system allows for an increase of up to 54% in instrument mass. This improvement reaches 76% when using low pressure tanks instead of medium pressure.

The pumps can be integrated into systems for both primary propulsion and attitude control systems (ACS). In the ACS case, the pumps feed one or more thrusters as needed during ACS pulsing by pressurizing a plenum which feeds each set of thrusters, as seen in Fig. 2. Here, the propellant storage and feed system can be designed for low pressures,

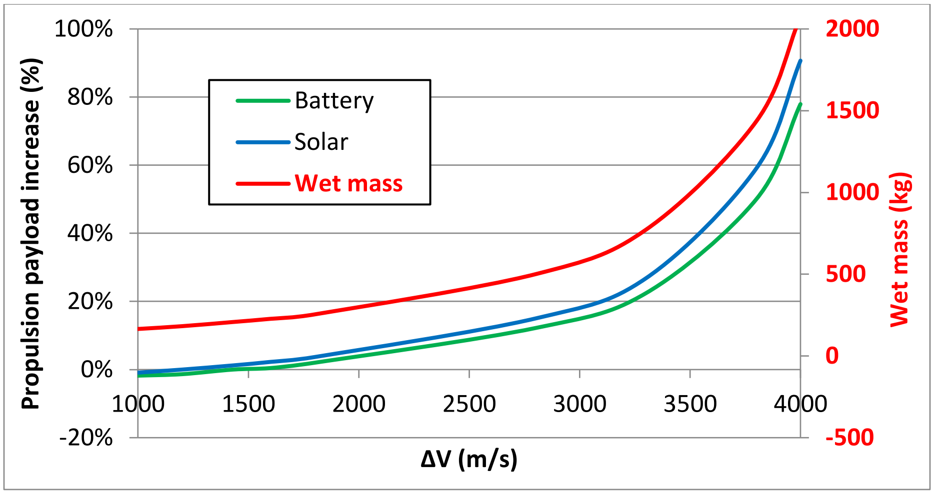

Fig. 1 Percentage increase in payload mass compared to pressure-fed system sized for 100 kg payload; the corresponding wet mass is shown in red.

Table 1 Mass study (kg) between baseline 2-stage concept and single stage pressure-fed and pump-fed (battery powered) concepts assuming the same initial mass and total ΔV of 3091 m/s.

	Baseline: Two stage concept	Single stage MON-25/MMH lander concept		
Lander masses	**Lander only**	**Pressure-fed**	**Micropump-fed**	Notes:
Payload	**26**	**-24**	**30**	**available payload mass**
Structure	28	28	28	additional structure in propulsion sys.
Propulsion Sys.	30	155	101	includes extra structure, battery, margin and residual propellant/pressurant
Other bus subsystems	94	94	94	no change; additional power (battery) in propulsion sys.
Margin	25	25	25	No change; additional margin built in propulsion sys.
Propellants	54	517	517	-
Lander Wet	**257**	**795**	**795**	-
Solid propellant deceleration stage total	**538**	-	-	-
Total per "cruise config."	**795**	**795**	**795**	held constant; assumes same wet mass as total
Payload change	**baseline**	**-192%**	**15%**	-

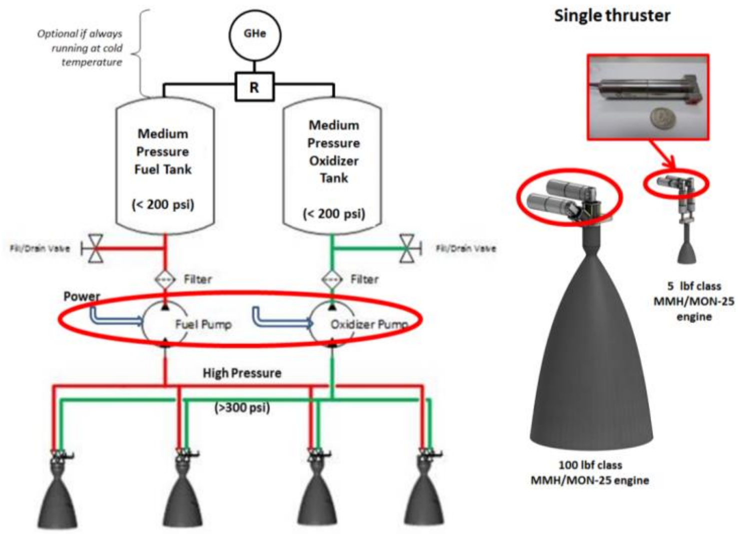

Fig. 2 Micropumps (circled) can be used to provide the pressure rise for clusters of hypergolic thrusters (left) or individual thrusters (right): >450 N (>100 lbf, left) and 22 N (5 lbf, right) class thrusters.

with lighter, cheaper, conformal tanks (for example, using additive manufacturing techniques) used instead of spherical or cylindrical pressurized tanks. This would further reduce system overall size while also reducing system verification costs and other risks for secondary payloads in a rideshare configuration.

B. Pump Technology

Microgear pump technology is based on high precision electrically driven gear pumps (Fig. 3), where the flow rate can be easily controlled via the motor RPM. Pumps can be magnetically driven (Fig. 3) such that no dynamic seal is present, extending life and reliability. The technology was first developed and validated for small pump-fed systems for a variety of propellants, starting with HAN-based propellants in 2009 [8] [9] [10]. In 2012, 5 N (1 lbf) hydrazine thruster vacuum hot fire tests were conducted, and the technology later adapted to and improved for nitrogen tetroxide (NTO). Other potential uses for the microgear pump technology include propellant transfer, coolant system circulation, and many commercial applications such as the medical and/or food and beverage fields. An overview of flow rates from previous Flight Works spacecraft pump projects are shown in Fig. 4.

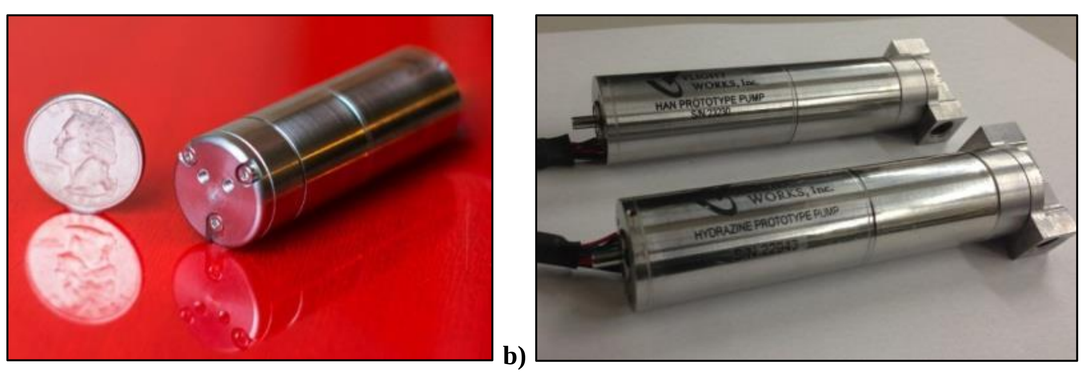

Fig. 3 (a) Sample commercial high precision micro gear pump and (b) hydrazine pump tested with a 5 N (1 lbf) thruster. Both families have the same diameter, but the hydrazine pump features a powerful magnetic coupling to reach higher pressures.

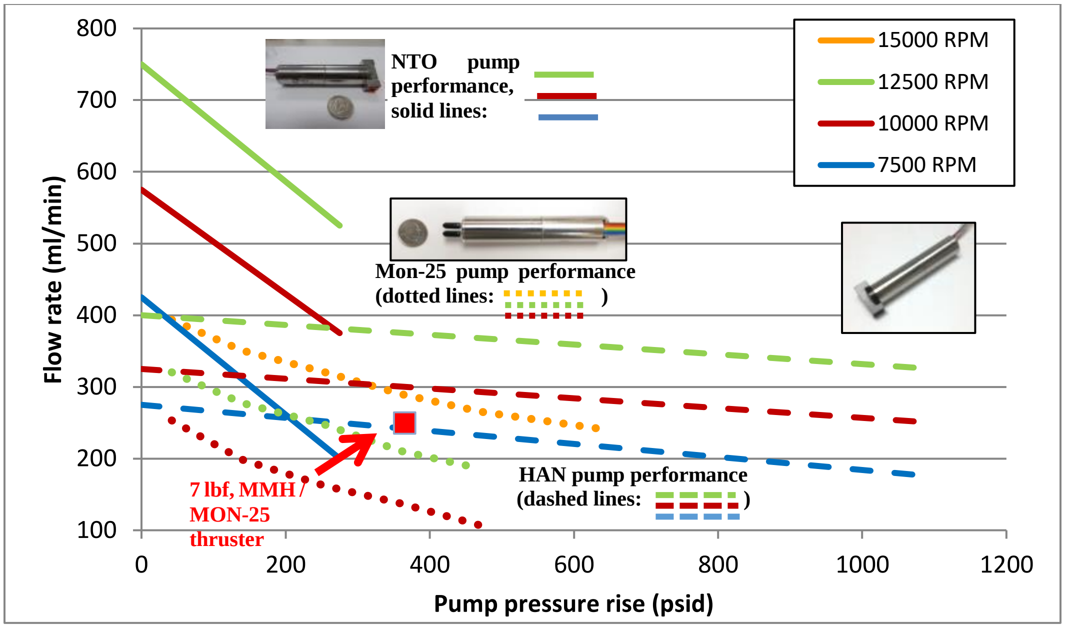

Fig. 4 Flight Works micropumps demonstrated performance with NTO (solid lines), MON-25 (dotted lines) and HAN-based propellant (dashed lines).

III. Micropump for 20-30 N (5-7 lbf) Class Thrusters

Expanding upon the family of micropump designs and previous work on MON-25 micropumps [7], Flight Works, Inc. designed and produced prototype pumps aimed at 5 lbf class thrusters, denoted the "small pump". These pumps, shown in Fig. 5, supply 250-300 ml/min flow rate for 20-30 N (5-7 lbf) class MON-25/MMH thrusters at a pump outlet/thruster inlet pressure of 250 to 365 psia. Acetone was used as the referee fluid for MON-25 (its characteristics being closer to that of MON-25 than CFC-113 which is often used as NTO referee fluid) and water for MMH. Nominal pump performance for ambient temperature MON-25 referee fluid (acetone) is shown in Fig. 5.

Protoqualification was performed on a single unit, here qualification refers to the tests needed for ensuring adequate pump operations during hot fire tests, a summary of testing is seen in Table 2. Highlights of the qualification series include: pulsed mode operation with MON-25 referee fluid (acetone), at ~365 psid pump head differential pressure, with a constant motor RPM of 12.5k, and representative endurance testing (estimated at less than 20hrs of operation in most systems). Pulsed mode testing was conducted with a 10% duty cycle (1 second period), with 20 pulses per sequence, 20 sequences total, using a fast-acting solenoid valve to simulate a thruster main valve. Example results can be seen in Fig. 6, showing 3 pulsed cycles during a 20-pulse sequence. The pump closed loop differential pressure (thruster valve not open) is 365 psid, delivering 300 ml/min flow rate. During the open thruster flow pulses, the pump delivers approximately 275 psia to the simulated thruster, which is within the targeted operating inlet pressure for a 5-7 lbf thruster.

Endurance testing serves to simulate operating conditions in excess of the total expected pump run time. Lifetime testing was run with MON-25 referee fluid (acetone), which is a thin fluid and should potentially result in greater wear of the pump interior components than would an extended run duration utilizing MMH referee (water). Post-endurance testing, shown in Fig. 7, clearly indicates that the end-of-life pump performance exceeds the target operating pressure and flow range (shown in green).

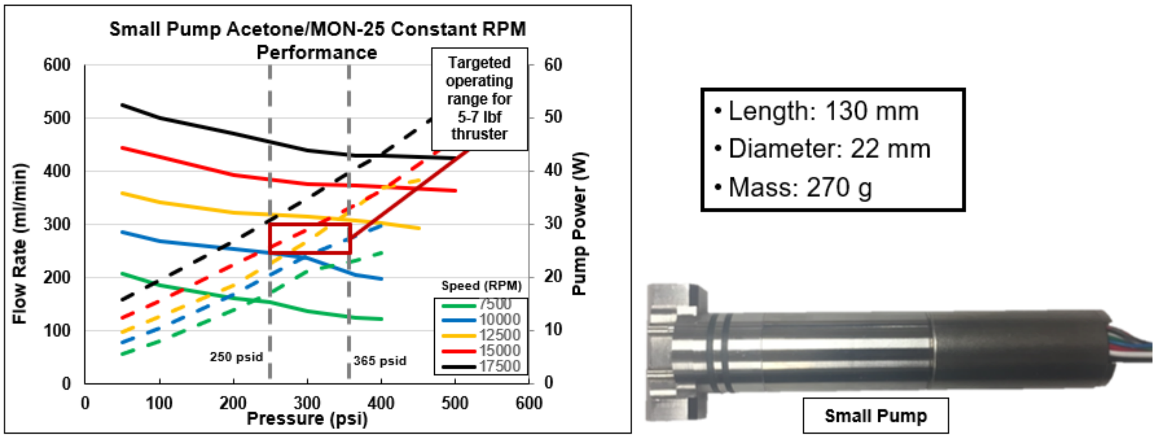

Fig. 5 Nominal ambient performance, MON-25 referee fluid, key pump characteristics, and overall dimensions of 5-7 lbf class micropump.

Table 2 Small Pump Protoqualification Testing

Qual Test	Description	Status	Notes
1. Test pump performance at ambient with referee fluids			
1a.	Characterize flow rate at nominal pressure	Complete	*Completed at various RPMs with both referee fluids*
1b.	Evaluate maximum differential pressure	Complete	*Completed at various RPMs with both referee fluids*
1c.	Characterize flow rate and nominal pressure with pulsed mode operations	Complete	*Completed at nominal operating pressure and speed with acetone referee fluid*
2. Test pump performance at cold temp. with MON-25 referee (acetone): targeted -30 °C			
2a.	Characterize flow rate at nominal pressure	Complete	*Completed at various RPMs with acetone referee fluid*
3. Test pump performance at hot temperature with referee fluids: targeted ~40 °C			
3a.	Characterize flow rate at nominal pressure	Complete	*Completed at various RPMs with both referee fluids*
3b.	Evaluate cavitation (as it affects volumetric efficiency) ata nominal flow rate and pressure	Complete	*Completed at various RPMs with both referee fluids*
4. Test pump structure integrity			
4a.	Evaluate pump at proof pressure (1.5X MEOP) while submerged (leakage)	Complete	*Completed submerged (no leaks observed); no change in pump performance or measurements following proof test*
5. Test pump in a partial thermal vacuum with referee fluid(s)			
5a.	Evaluate pump thermal profile in a partial vacuum	Complete	*Completed at nominal operating pressure and speed with water*
6. Test pump durability with acetone			
6a.	Evaluate pump life while operating at nominal pressure	Complete	*4x5 hour cycles completed at nominal operation pressure and speed with acetone*

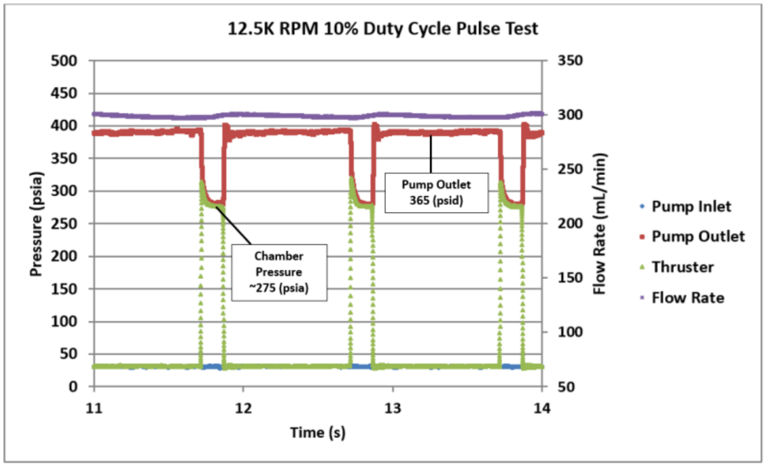

Fig. 6 Example pulsed mode operation sequence at 10% duty cycle. Pump performance at nominal 365 psid during recirculation period, 275 psia delivered to simulated thruster inlet during pulsed open flow period.

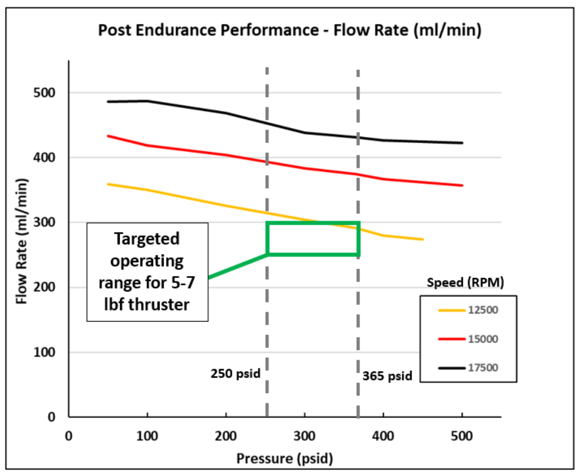

Fig. 7 Small pump pressure vs. flow rate performance with MON-25 referee fluid (acetone), post endurance testing. Targeted performance shown in green- significant margin remains available.

IV. Micropump for 445-890 N (100-200 lbf) Class Thrusters

Evolving from the 5-7 lbf small MMMH/MON-25 micropump development series, Flight Works, Inc. designed and produced prototype pumps aimed at 200 lbf class thrusters, denoted the "medium pump". These pumps, shown in Fig. 8, supply 8-10 L/min flow rate for 890 N (200 lbf) class MON-25/MMH thrusters at a pump outlet/thruster inlet pressure of 250 to 365 psia. This highly increased flow demand, as compared to that of the 5-7 lbf class micropump, translates into larger pump displacement requirements, larger power, etc. It is noted that due to contract schedule limitations, the prototype pump design seen in Fig. 8 is not optimized for flight weight and dimensions, and in particular, the pump commercial off-the-shelf (COTS) motor was not intended for space operations.

Similar to the test series for the small pump, a protoqualification was performed on the medium pump design, results are summarized in Table 3 below, with highlighted results described herein. An ambient temperature (25 °C) performance map of pump pressure rise and associated flow rates is shown in Fig. 9 for various fixed motor RPMs and propellant referee test fluids. The target hot fire test operating condition is shown as a red dot and is specific to the test hot fire test provider's (Mach Diamond) hardware. Operation of the pump design at a fixed 12500 RPM motor speed resulted in performance well into and above the target flow envelope for simulated MMH (water). Simulated MON-25 (acetone) performance is as slightly lower, as expected due to fluid viscosity differences, yet performance is within the target flow envelope at the planned hot fire operating point.

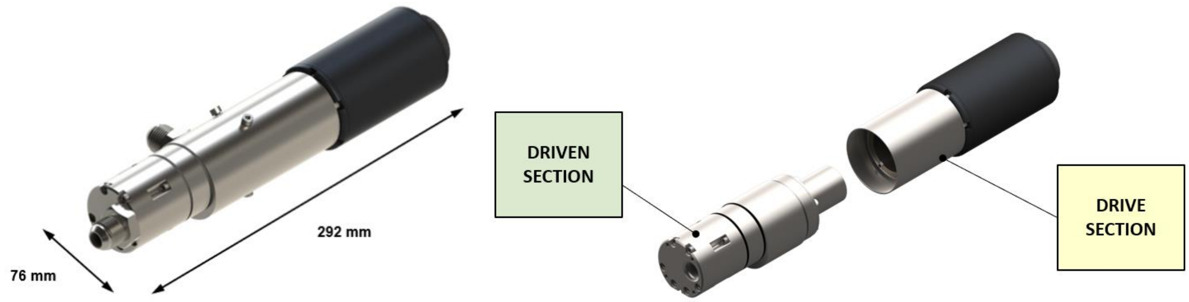

Fig. 8 200 lbf class micropump development design (not flight optimized).

Table 3 Medium Pump Protoqualification Testing

Qual Test	Description	Status	Notes
1. Test pump performance at ambient with referee fluids			
1a.	Characterize flow rate at nominal pressure	Complete	*Completed at various RPMs with both referee fluids*
1b.	Evaluate maximum differential pressure	Complete	*Completed at various RPMs with both referee fluids*
1c.	Characterize flow rate and nominal pressure with pulsed mode operations	Complete	*Completed at nominal operating pressure and speed with acetone referee fluid*
2. Test pump performance at cold temp. with MON-25 referee (acetone): targeted -30 °C			
2a.	Characterize flow rate at nominal pressure	Complete	*Completed at various RPMs with acetone referee fluid (and Jet-A referee fluid)*
3. Test pump performance at hot temperature with referee fluids: targeted ~40 °C			
3a.	Characterize flow rate at nominal pressure	Complete	*Completed at various RPMs with both referee fluids*
4. Test pump structure integrity			
4a.	Evaluate pump at proof pressure (1.5X MEOP) while submerged (leakage)	Complete	*Completed submerged (no leaks observed); no change in pump performance or measurements following proof test*

The use of MON-25 oxidizer over MON-3 greatly extends the lower limits of the propellant operation temperature range, therefore qualification testing of the medium pump at -30 °C was conducted, where Jet-A was used as a referee fluid for cold MMH. Performance maps at -30 °C temperature is shown in Fig. 10 below. As can be seen, operation at fixed a 12500 RPM motor speed is within the target flow range envelope for simulated MMH, and simulated MON-25 performance is just within the target envelope at pressure differentials less than ~200 psid.

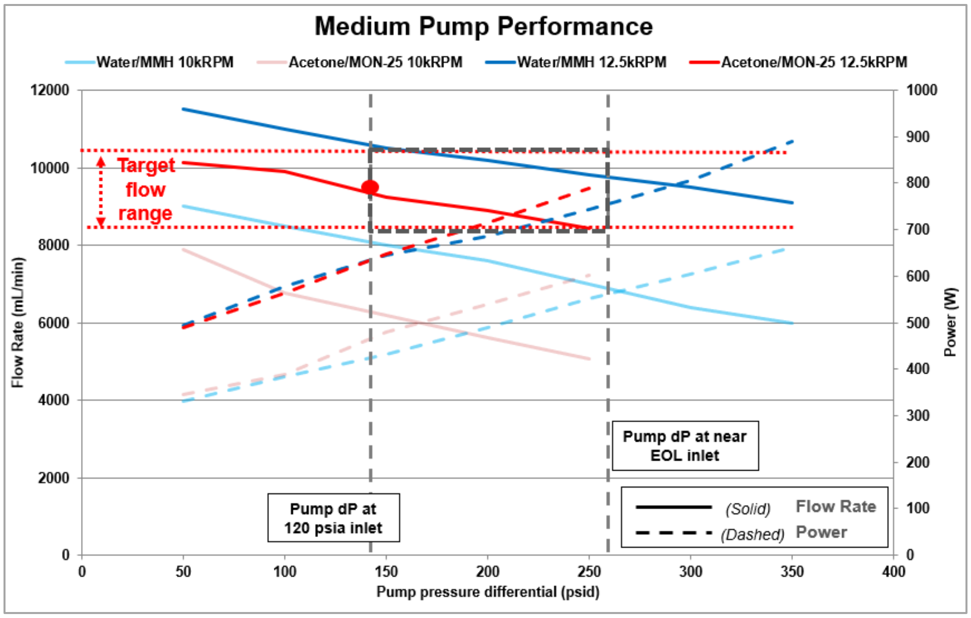

Fig. 9 200 lbf medium pump performance with water and acetone at varied fixed motor speeds- pump head pressure differential (psid) vs. flow rate (ml/min) and power (W). Hot fire test facility target conditions indicated by a red dot.

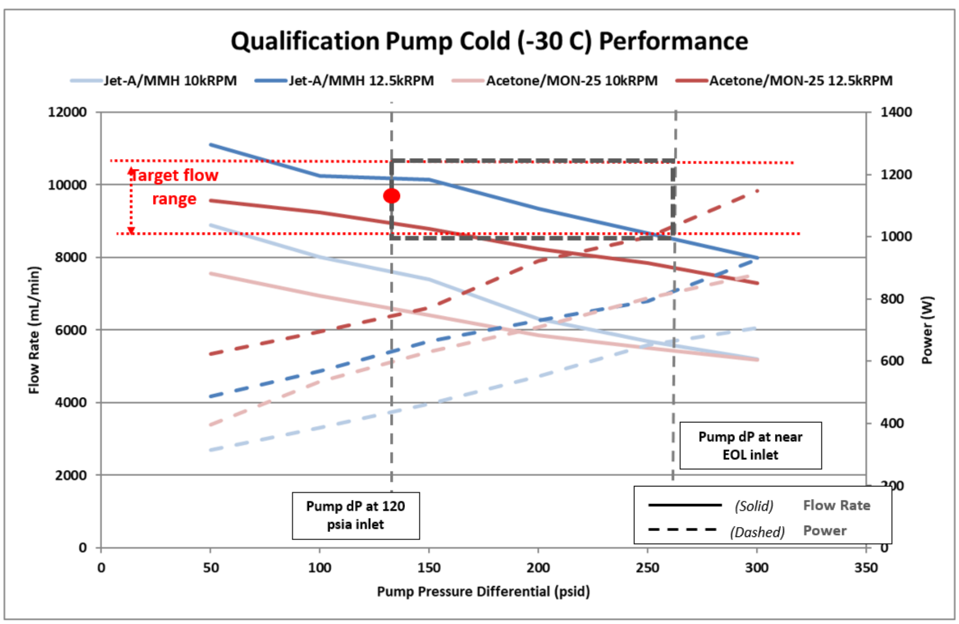

Fig. 10 200 lbf medium pump testing, -30 °C performance- pump head pressure differential (psid) vs. flow rate (ml/min) and power (W). Hot fire test facility target conditions indicated by red dot.

V. Hot Fire Testing

Following pump qualification with propellant referee fluids, the 200 lbf thrust-class medium pumps were hot fire tested with MMH and MON-3/MON-25 in representative conditions, increasing the technology readiness level. Here, representative conditions refer to temperature, from ambient to cold propellants (targeted -20/-30 °C). For hot fire testing, Flight Works subcontracted, Mach Diamond, Inc., MT, which has extensive experience in hypergolic thruster development and testing, primarily for the former Rocketdyne division of what is now AR. Under several MDA programs, Mach Diamond has developed and tested at its facilities in Montana.

The 200 lbf class-thrust technology demonstration configuration for pump-fed hot fire testing is seen in Fig. 11 below. Flight Works produced plumbing loops (one for fuel, one for oxidizer), which included the pump, relief return valve and housing, pressure and temperature instrumentation, and associated electronic support (motor controllers, DAQ, power supplies, etc.) to integrate into an existing test stand at the provider facility.

Both steady state and pulsed mode test sequences were performed, including with conditioned propellants to cold temperatures (e.g. -20/-30 °C). Initial hot fire testing was performed at Mach Diamond with ambient MMH/MON-3, in order to characterize the pump loop with actual hypergolic propellants, and perform a lower-risk test sequence with respect to the health of the thruster, since MON-25 combustion can be more unstable than that of MON-3. 3 x 1000 ms steady state burns were conducted with pump operation, and a 1 x 1000 ms burn was performed with the test stand in pressure-fed blowdown configuration. A representative 1000 ms pump-fed firing is shown in Fig. 12. As can be seen, the plume is fuel rich. This is due to the higher fuel feed pressure to the thruster, data seen in Fig. 13.

Interestingly, the data showed an anomalous flow condition which occurred during all MON-3 testing, but was not discovered, nor the culprit determined until after the MON-3 test series, when data a video became available for review. As can be seen in Fig. 13, pump outlet pressures began to rise in incremental steps as the motor RPM was increased. However, at a certain pressure for each propellant (~ 275 psig outlet on the fuel and ~430 psig outlet on the oxidizer), the pump outlet roughly equalizes with the pump inlet pressure. Once data, and especially video, were reviewed it was determined that the pump magnetic drives had in fact decoupled from the fluid pump head, which would explain the sudden pressure drop. The decoupling coincides with a brief increase of pressure as the pumps are commanded to a higher motor RPM. To further support the theory of decoupling events, shown in Fig. 14 is an overlay of pump loop pressure with motor current signal (motor current signal does not represent Amps). Coinciding with the sudden outlet pressure decrease event is the drop in the motor current signal. Since current is directly related to required torque of the motor, this indicates the motor (and drive section) were now spinning freely and the pump had decoupled. The root cause of the decoupling is thought to be due to the geometrical configuration of the pump loop in the integrated test stand at Mach Diamond, which left the inlet of the pumps non-wetted, depending on the system propellant tank load levels. When testing at Flight Works with the pump loops on the bench, the pumps were able to self-prime. However, perhaps there were issues with priming when the loop was configured upright, as was the orientation during hot fire testing. Nonetheless, the thruster survived combustion at low chamber pressures, and testing was moved forward to the planned MON-25 tests.

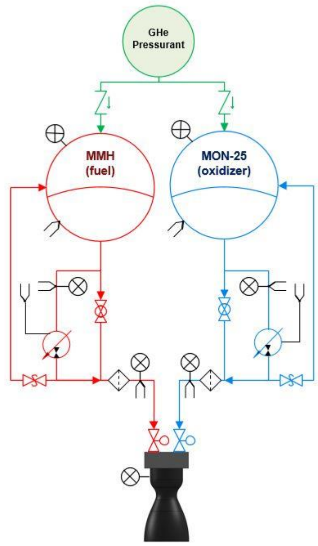

Fig. 11 Hot fire test schematic.

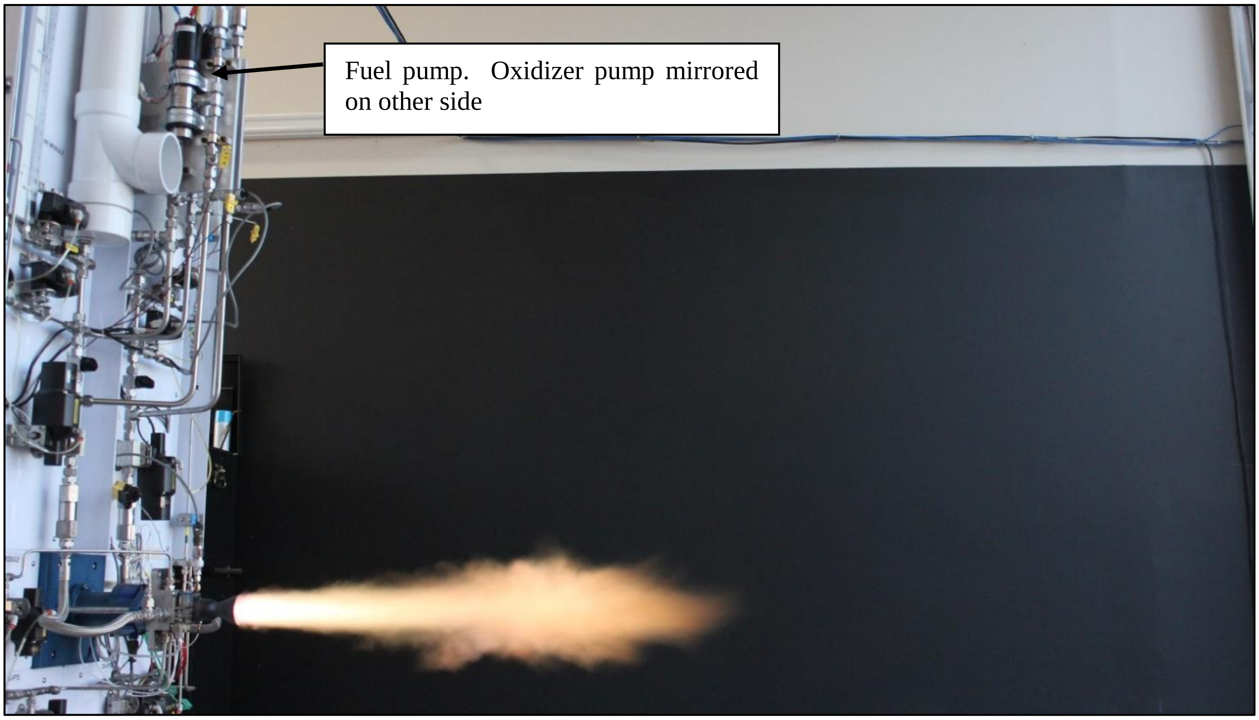

Fig. 12 Pump fed MMH/MON-3 1000 ms hot fire test.

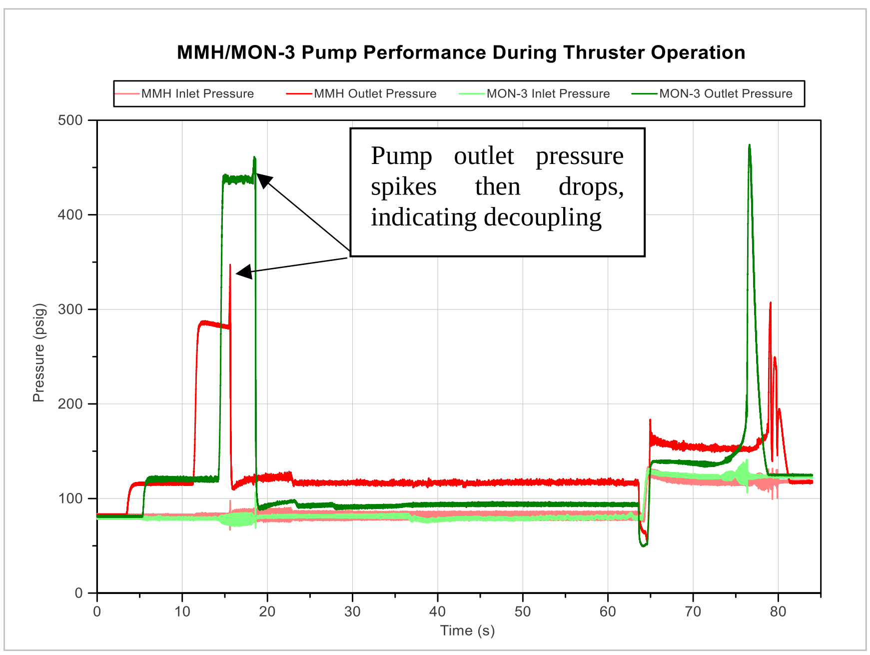

Fig. 13 Pump fed MMH/MON-3 1000 ms hot fire test pump loop data.

Operational corrections were successfully implemented for the MON-25 test series to mitigate decoupling of the pump magnetic drive. During closed pump loop testing, the pump motor RPMs were increased until decoupling was observed, and decouple RPM noted (see Fig. 15). The motor RPM of each pump (usually different for fuel/ox pump, and dependent on pump inlet pressure) was then reduced until the pump re-coupled, then RPM increased again to just below the decouple threshold. Based on the decoupling points, one of the pump RPMs was usually further decreased to match the oxidizer to fuel ratio (O/F) for thruster hot fire testing.

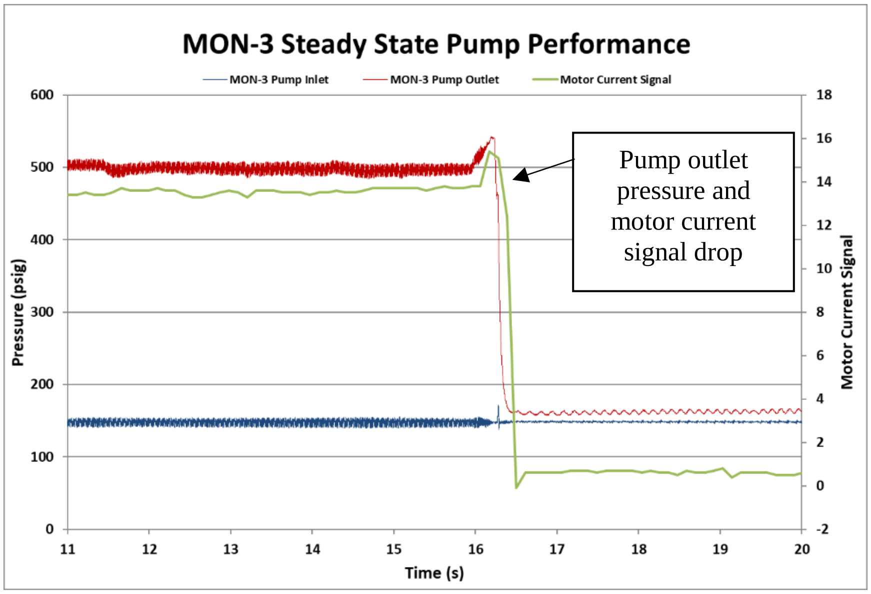

Fig. 14   Pump fed MMH/MON-3 hot fire test pump loop data.

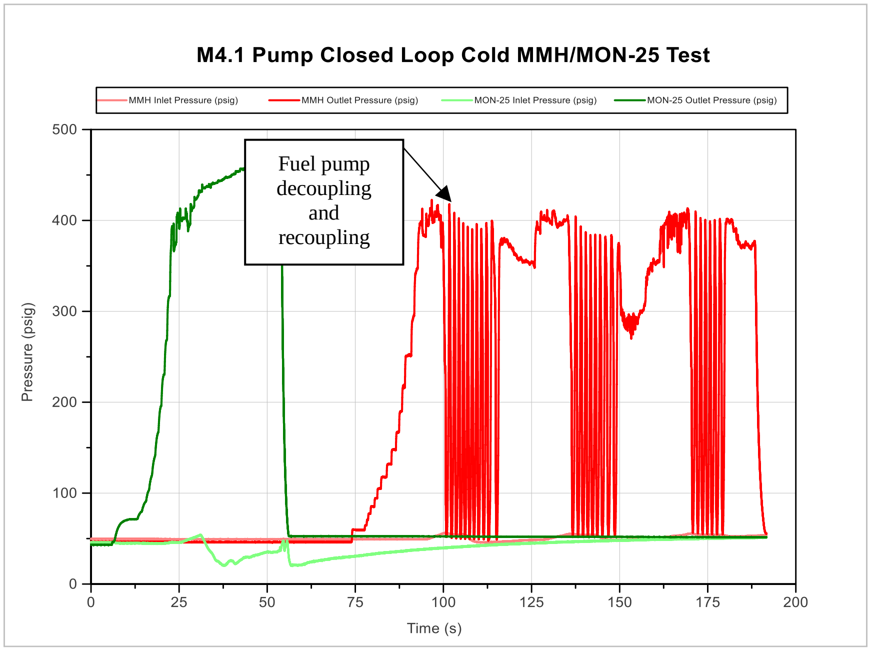

Fig. 15 Pump fed MMH/MON-25 closed loop flow pump data.

Since the decouple threshold was lower (approximately half) than the planned pump operation regime, system flow to the thruster was greatly reduced, and as a result pump outlet pressure was off target due to fixed restriction of the injector.

A hot fire test series with thermally conditioned MMH/MON-25 was successfully completed, an example firing is shown in Fig. 16**,** where ice that had condensed on the exterior of the propellant tank housing, and can be seen falling down toward the thruster. The following test series with cold (-20/-30 °C) MMH/MON-25 was successfully completed:

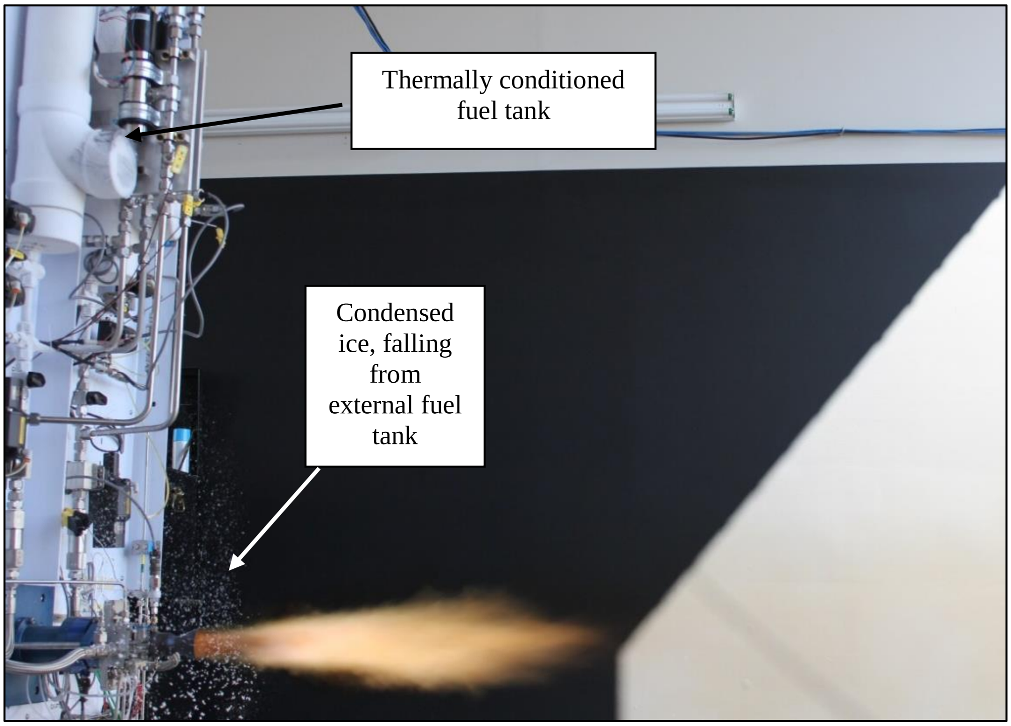

Fig. 16 Pump fed MMH/MON-25 1000 ms hot fire test. Ice from thermally conditioned propellant tanks (-20 to -15 °C) can be seen falling off near the thruster.

- 1 x 1000 ms trim test
- 1 x 1000 ms
- 1 x 3000 ms
- 10 x 100 ms ON 200 ms OFF pulsed test

Target propellant temperatures were -30 °C, while tank external skin temperatures reached as low as -65 °C. Propellant temperatures in the pump loop were measured at -20 °C for the fuel pump inlet and -15 °C on the oxidizer pump outlet before thermocouples failed due to freezing of the electrical harness. Temperatures are assumed to have further decreased in the propellant lines due to thermal soak from the propellant conditioning system.

During the 1000 ms steady-state burn (after trim test), the fuel pump and oxidizer pump provided an initial 220 psid and 350 psid pressure rise, respectively, but upon commencement of the thruster flow, those pressures were reduced to approximately 65 psid and 25 psid, due to the need to keep motor speeds below the decouple threshold. Chamber pressure and load cell instrumentation data are seen in Fig. 17, and despite the low thruster inlet pressure, combustion is fairly stable, while ~40-45 lbf thrust was generated. The pump loop data from the 3000 ms burn is shown in Fig. 18 below, with similar pressure generation from the pumps, again due to required motor operation in a low RPM regime.

Data from a pulse-train test series with cold (-20/-30 °C) MMH/MON-25 is shown in Fig. 19 (Flight Works pump loop data) and Fig. 20 (Mach Diamond chamber pressure and load cell data) below. The fuel and oxidizer pumps each generate 100 psid and 175 psid respectively, and is characteristic of a pump fed pulse test and was demonstrated during qualification testing, the pump outlet pressure drops when the thruster valve opens to demand flow, recovering when the valve has closed. The difference here is that the flow, and hence the pressure, provided to the hot fire thruster was lower than targeted due to pump RPM limitations. The thrust generated (Fig. 20) reaches over 100 lbf. It can be noted that this is slightly higher than thrust in the immediately preceding tests. This is due to the fact that the previous test runs (3 seconds) consumed propellant, dropping the liquid level in the tank, and as a result, the liquid level was below the height of the pump intake, which was thought to contribute to the geometrical configurations issues which resulted in pump drive decoupling. Therefore, the propellant tanks were pressurized to reduce the ullage volume and raise the liquid level to wet the pump inlets. Although the pressurant was then isolated from the test stand, the initial higher pump inlet pressure, plus the pressure differential generated by the pump, resulted in higher pressure to the thruster, and therefore higher generated thrust.

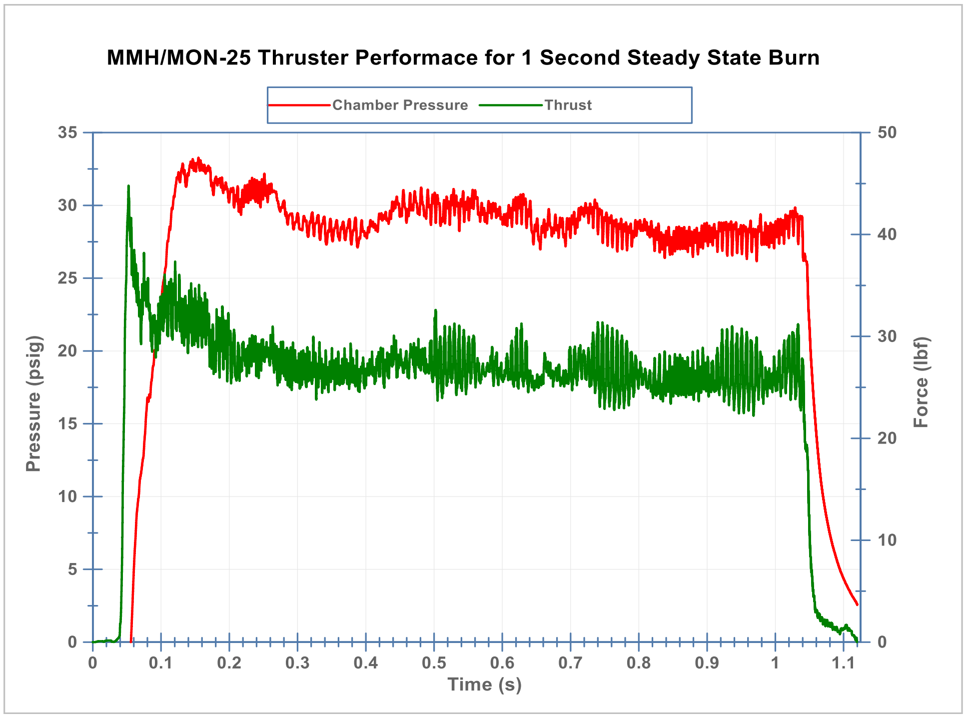

Fig. 17 Pump fed MMH/MON-25 1000 ms hot fire test chamber pressure (psig) and thrust (lbf).

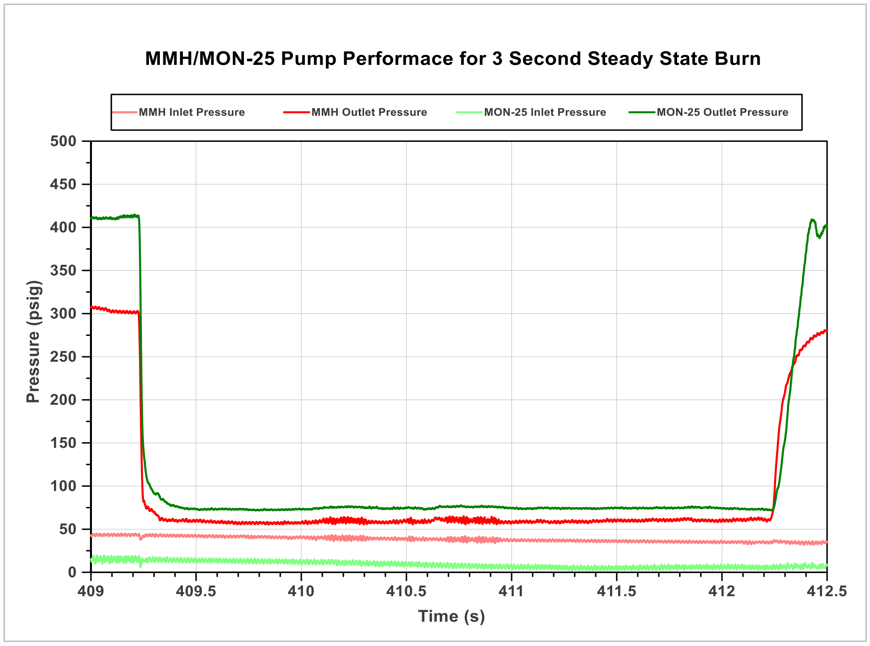

Fig. 18 Pump fed MMH/MON-25 3000 ms hot fire test pump loop data.

Overall, the hot fire test series data pointed to opportunities for improvement in pump design and pump system integration, which could be developed and implemented as part of a future contract effort.

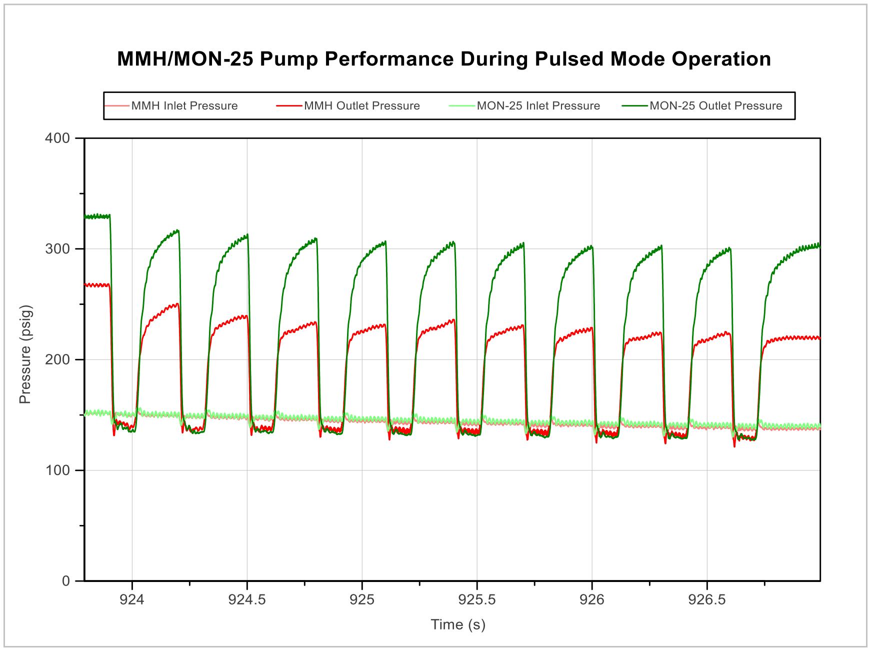

Fig. 19 Pump fed MMH/MON-25 10 x 100 ms ON 200 ms OFF pulsed hot fire test pump loop data.

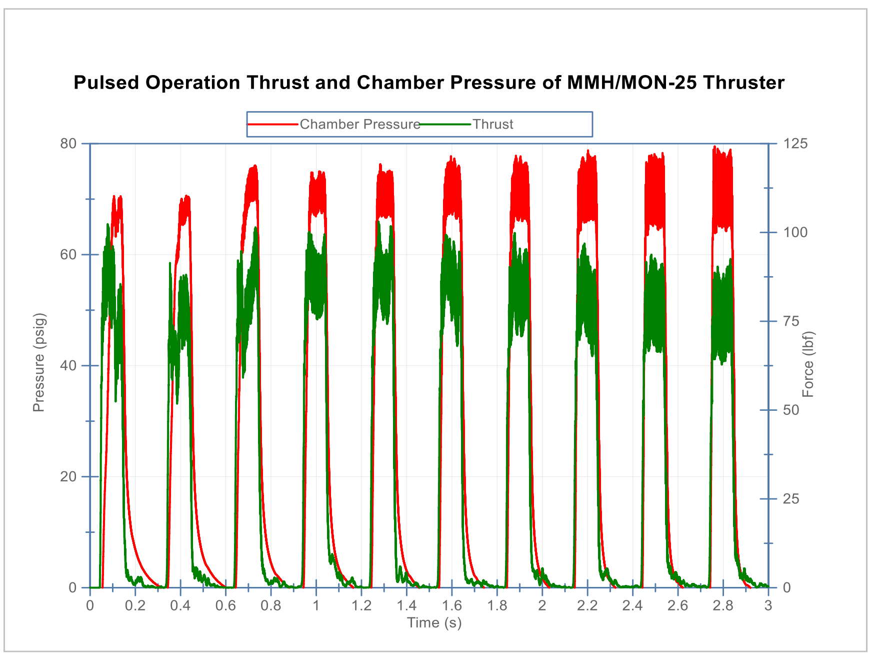

Fig. 20 Pump fed MMH/MON-25 10 x 100 ms ON 200 ms OFF pulsed hot fire test chamber pressure (psig) and thrust (lbf).

VI. Conclusion

Discussed here was the development and demonstration of two classes of high-power density MMH/MON-25 micropumps for spacecraft propulsion systems, developed by Flight Works, Inc. Micropumps can serve as enhancers for small, compact, high performance propulsion systems, replacing traditional pressure-fed systems for a decreased system mass. Utilization of MON-25, as opposed to MON-3 or NO, reduces the freezing point of the oxidizer from -9 °C to -55 °C, such that the low freezing points of both propellants (MMH/MON-25) extends their potential use to environments very low temperature demands.

The two thrust levels targeted were in the 5-7 lbf and 100-200 lbf classes. Development focused on designing and qualifying the pumps to meet the required thruster inlet flow rates and pressures with both propellants for the targeted thruster classes, for use in both primary propulsion and attitude control systems (ACS). Flight Works has developed the micropump designs to be compatible with hypergolic propellants, and testing with referee fluids successfully demonstrated the ability to provide adequate flow rates for both thrust levels (5-7 lbf and 100-200 lbf) at pressure increases larger than 350 psid.

Hot fire tests with hypergolic propellants and a flight-like thruster were successfully conducted with the larger 200 lbf class pumps. 1000 ms and 3000 ms steady state burns and pulsed mode operations of the thruster were demonstrated with ambient temperature MMH/MON-3 for the purposes of risk reduction. Demonstrating the thermal envelope advantages of MON-25 as the oxidizer, steady state and pulsed-mode hot fire testing with propellant temperatures of approximately -30 °C were accomplished. Test data pointed to opportunities for improvement in pump design and pump system integration, which could be developed and implemented as part of a future contract effort.

Acknowledgments

This work was supported in part by a Phase I SBIR from NASA Glenn Research Center, contract #NNX15CC40P, and by a Phase II SBIR from NASA Glenn Research Center, contract # NNX16CC12C (F. David Koci [GRC-LTF0] acts/acted as COR for both contracts). Flight Works acknowledges the participation of Phase II subcontractor Mach Diamond.

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