NITROUS OXIDE MONOPROPELLANT THRUSTERS - RELOADED

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ABSTRACT

Nitrous oxide (N_2O) is a low-cost, non-toxic propellant with potential to balance performance and handleability. It offers an attractive option for New Space companies needing to include propulsion in their spacecraft but lacking resources to purchase traditionally space-qualified products. N_2O as a rocket propellant was first researched over 20 years ago; despite considerable effort, R&D has not been able to realise its many potential benefits. Newton Launch Systems and Rocket Engineering Ltd, with UKSA grant support, are developing a small, low cost, re-startable monopropellant thruster targeting a density impulse and power specific thrust competitive with other low-cost small spacecraft propulsion thrusters.

1. BENEFITS & DRAWBACKS OF N2O

The current state-of-the-art chemical mono-propellant widely used on large spacecraft is hydrazine. This is increasingly unpopular with small spacecraft and their launch service providers. It is being aggressively targeted for phase out in European and UK space usage due to health & safety concerns and handling costs. Alternatives have started to gain flight heritage, in particular small HPGP (High Performance Green Propellant) thrusters, however the shortcomings include a need for exotic refractory alloys to withstand the decomposition temperature, and the high cost of the propellant itself.

High Test Peroxide (HTP) and Nitrous Oxide (N_2O) have been considered as alternatives to hydrazine, both undergoing a highly exothermic decomposition reaction when heated or in the presence of catalysts, making them viable monopropellants. HTP is a strong oxidiser and naturally decomposes over time or in contact with many common propellant tank materials, notably Titanium, posing handling and storability challenges. N_2O is considered long-term storable over a wide temperature range, and is catalytically reactive with far fewer structural materials than HTP. However, it is a liquefied gas at standard temperature and pressure, requiring careful consideration of pressurised gas systems and pressure vessel safety to meet launch service provider requirements. Consequently, neither propellant is considered an ideal candidate for hydrazine replacement, despite being theoretically able to provide a comparable specific impulse (Isp).

The view of the authors is that N_2O has the greater potential for realising a low-cost alternative to legacy hydrazine propulsion systems, targeting small satellites. In particular, the authors consider N_2O may have a unique selling point in an era when propulsion systems for end-of-life de-orbiting and collision avoidance are becoming increasingly necessary, driven by regulatory requirements for sustainable space operations; and in the cost-constrained era of 'New Space'.

N_2O's low toxicity, being widely used as both a food additive and a low-grade medical anaesthetic, makes it very easy to use with only basic safety training. Being long-term stable in storage and self-pressurising to ~5MPa at room temperature, makes it well-suited to use in multi-year space missions. Its self-pressurised nature also permits a highly flexible operation to be offered by a single propellant tank and thruster, including cold-gas mode (prioritising low minimum impulse bit) all the way up to high impulse hot-fire burns with a vacuum Isp up to ~180 seconds, without the need for separate pump or blow-down systems. While liquefied gas propellants presents some challenges to satisfying launch service provider safety checks, both N_2O and traditional blow-down systems present similar shortcomings, and both have been commercially deployed recently

Previous commercial and academic research and development work on the use of N_2O as a monopropellant has failed to produce a commercially viable system (See Section 2). This has largely been due to N_2O requiring a relatively high temperature of 800℃ to initiate decomposition, stemming from the 250kJ/mol activation energy to initiate decomposition [1], but then releasing 82kJ/mol, allowing a considerably higher decomposition temperature to be reached. The presence of a catalyst can significantly reduce the decomposition temperature [1], but it still presents a significant power draw and thermal management challenge to operate.

It is worth noting that these are not insurmountable problems; HPGP also requires a preheat temperature (200-350℃) to trigger its decomposition [2], and thrusters using HPGP have many years of space heritage [2]. However, the decomposition of N_2O liberates oxygen, leaving catalyst beds, heater elements and combustion chamber/thruster nozzle exposed to a high temperature exhaust rich in oxygen radicals. While appropriate coatings and materials selections can address this issue for most thruster components, catalysts need to be exposed. Significant and rapid catalyst erosion was observed in a variety of past research work [3]. In addition to the added complexity of ensuring pressurised systems safety satisfied launch service provider safety requirements, these were sufficient barriers to block commercialisation using this approach. A comparison of common monopropellants available in Europe is shown below.

Table 1. Comparison of high-level parameters of popular monopropellants or candidates. Density is given at room temperature.

	Vac Isp (s)	**Density (g/cc)**	**Adiabatic Flame Temp (℃)**	**Toxicity**	**Cost (£/kg)**
Hydrazine	220-230	1.00	600	High	130
ADN / HPGP	205-235	1.36	1650-1900	Mid	1,030
90% HTP	180	1.35	750	Low-Mid	17-32
N_2O	At least 180	0.8	1336	Low	5.50

2. PREVIOUS N_2O SPACEFLIGHT HERITAGE AND LESSONS LEARNED

The most fundamental and comprehensive work with N_2O which led directly to a flight demonstration was carried out in the 1990s by Lawrence as PhD research at the University of Surrey Space Centre [4], considering a resistojet application. A third generation (Mk III) resistojet developed during the research was flown on the Surrey Satellite Technology UoSat-12 mission launched in 1999. Another flight model was built for a USAF mission which was not launched. A significant achievement was the demonstration of self-sustaining, zero power decomposition for 18 hours [5, 6] but although the decomposition was possibly incomplete since the maximum Isp measured was 148s, considerably less than the theoretical vacuum Isp of the monopropellant. More significantly, the power demand to initiate decomposition at around 100W was considered too high for early microsatellites and SSTL moved to a lower power resistojet optimised for other non-toxic propellants including Butane, Xenon and most recently water [7].

In Europe, significant N_2O flight heritage has since been obtained for 1N and 20N class bipropellant thrusters, with Netherlands-based Dawn Aerospace claiming 76 thrusters in orbit on board 16 spacecraft in 2024 [8]. Dawn's thruster technology, available in 1N, 20N and in future 200-300N classes, uses N_2O and Propene C_2H_4 propellants, fed in blowdown mode, spark ignited and with regeneratively cooled thruster chambers. A significant claimed benefit is up to 50% lower system weight and 36% lower system volume than LMP-103S, a key hydrazine replacement system, and compared to other blow-down alternatives [8].

Although the heritage achieved since 2018 is laudable, a publication in 2022 [9] notes that although thruster qualification achieved 11000 ignitions and 94000Ns of impulse, steady state performance could not be achieved before damaging the Inconel 718 chambers. A maximum 10s burn, chamber wall temperatures exceeding 980℃ and long cool down periods between firings were reported. This problem is expected to worsen for the smaller 1N class thruster, which has a much smaller flow rate of N_2O available for cooling. Design and potentially material improvements to increase lifetime were reported [9].

3. PROGRESS IN N_2O MONOPROPELLANT THRUSTERS

Work by Zakirov et al [5,6] starting in 2001 at the Surrey Space Centre continued over the next decade in multiple other groups, including Wallbank et al [3] in the university of Surrey chemistry department 2004, and Zhu [10] at the Chinese Academy of Sciences in 2007, researching use of Ir / Al_2O_3 based catalysts such as the "Shell405" catalyst used commercially for hydrazine, and more thermally robust compounds based on Barium-Iridium-Iron-Alumina, respectively.

These groups focused on use of a heated catalyst bed, comparable to legacy hydrazine systems; however all groups encountered issues with catalyst degradation (for widely available catalysts) [3] and difficulties synthesising and shaping catalyst beds for more exotic materials which remained stable in high temperature oxidising conditions [10]. Catalyst performance was observed to decrease rapidly as temperatures rose, with later testbed systems finding that ignition failure began to occur after 2-5 restarts [3]. Mitigation without addressing robustness would require an excess of catalyst material, adding significantly to both system mass and power requirements for heating (and leaving unresolved questions over whether this would significantly benefit restart capability).

Although most studies thus showed that a N_2O monopropellant thruster system was technically feasible, the findings cast considerable doubt over the practical and commercial feasibility. However, prior to Dawn Aerospace' commercial success (see previous section), there was little apparent market pressure to move away from legacy hydrazine systems, so there was less urgency to identifying suitable green propellant systems than there is today; this lack of pull factors was a major driver in effectively halting this line of research for many years. Renewed interest has occurred in the last 5 years, with examples of recent work in Italy [11] in 2021 and in China [12] 2022, but in each case, the aim is still to replicate a hydrazine monopropellant system design which may not be well suited to the high temperature oxidising conditions of N_2O decomposition. The complexity, and potential cost of a cooled bipropellant approach such as the Dawn Aerospace Cubedrive, especially when considering the published pulse impulse limits [9], seem difficult to justify for many low cost missions.

After the initial setbacks of catalyst approaches alternative approaches began to be examined which might be better suited for the unique challenges of N_2O; in particular researchers at the University of Maryland (UoM) have trialled a number of such alternatives, which have included a dielectric discharge barrier [13] which sought to reduce the activation temperature of the decomposition reaction; and the use of an induction heater rather than a standard resistive heater [14] to trigger thermal, rather than catalytic, decomposition.

The UoM research demonstrated that such a system appeared feasible, igniting N_2O gas flow successfully, however the test was purely a proof-of-concept and did not address issues such as survivability of materials, optimal heat exchanger arrangements, and system-level thermal management or power optimisation (amongst others).

4. WHY RELOADED IN 2024?

Following on from the research at the University of Maryland noted above [14], Newton Launch Systems Ltd (NLS) noted the potential for an induction heater approach to address some of the previous issues with the use of N_2O as a monopropellant.

Induction heating is a method in which eddy currents are induced in a conductive 'workpiece' by a nearby conductive coil, through which a high frequency alternating current is passed. The workpiece is electrically and thermally isolated from the coil, with heating occurring due to interactions with the electromagnetic field. This design therefore does not require the same thin wire heating elements as more standard resistive heaters and allows direct heating of larger and more resilient structures such as a chemically inert heat exchanger. Such a system is expected to be far more resilient due to physical separation of active elements from high temperature oxygen liberated during decomposition. The approach taken by the University of Surrey's resistojet work by Lawrence [4] to achieve a thermal balance between heat radiation, and heat generation, with a self-sustaining zero power decomposition, after initial powered activation, is being looked at closely.

The increased potential efficiency of this approach, compared to the energy required to heat a large catalyst bed, appears to offset the increased power requirements of the high temperature needed to trigger thermal decomposition. By removing the need for an exposed catalyst, such a system should have significantly higher restart capability and burn duration. Once triggered, the reaction should remain self-sustaining, so the heater's power draw is not required throughout the burn.

In effect, the core problem changes from one of catalyst material design, to one of thermal and power management, where many more options and proven solutions exist to achieve and maintain adequate operating conditions for a host satellite.

In particular, the existence and commercial attractiveness of HPGP systems shows that such thermal management and power draw issues for relatively high temperature monopropellant systems are not an inherent blocker to successful adoption, providing suitable mitigation such as suitable high temperature materials and oxidation resistant coatings can be provided. NLS and Rocket Engineering Ltd (REL) are addressing these problems with a view to creating a commercially attractive N_2O monopropellant system.

NLS first conducted tests during 2021-22, under a UK Space Agency (UKSA) National Space Innovation Programme (NSTP) grant, aiming to produce an induction heated decomposition system to demonstrate feasibility of a self-sustaining decomposition reaction. This study also aimed to trial multiple different arrangements of heat exchangers to gather data for further analysis. This series of tests were deemed successful, but highlighted unexpected challenges in electronics design, thermal management, and materials resilience (See Section 6)

REL was then approached as a specialist in materials and thruster design with staff having worked on fuel film cooled bipropellant engines operating in the 1400-1600°C range.

A second project was recently begun under the UKSA Enabling Technologies (Call 3) Programme. This work is currently ongoing, aiming to produce a viable thruster core system, and examine system-level implications more broadly, aiming to bring this technology to market if a suitable approach can be identified, balancing the limitations of such a system against meeting commercial needs at as low a cost as possible. The requirements we are working towards are detailed in the next section.

5. APPLICATIONS AND CUSTOMER REQUIREMENTS

Simplicity, coupled with versatile operation in multiple modes, are key technical benefits of an N_2O monopropellant propulsion system. Research is ongoing into a range of potential use cases and broad mission requirements, which can then be used to inform design (and if necessary, define any mission profiles that cannot be met with this approach).

Desk-based research in late 2023 identified broad classes of mission level propulsion requirements for small, low cost spacecraft, as shown in the table below.

Further research into propulsion specific requirements and metrics is leading towards derivation of a specification for an N_2O thruster, and in due course an N_2O monopropellant system.

Table 2. Summary table outlining requirements for typical low cost propulsion system use cases

	Orbit Change	**Docking**	**RCS**
Longest Burn	~30 mins	<15 mins	<0.2s
Lifetime Thermal Cycles	<1,000	<9,000	<18,000
Cold Gas Required	No	Yes	Yes
Min Impulse Bit	Not Required	<0.1Ns	<0.1Ns

In a small spacecraft, particularly cubesats, propulsion system cost (not thruster cost alone) is the primary design driver. Work by Sellers at the Surrey Space Centre in the mid-1990s [15] evaluated the dimensions of cost, which were then applied to the UoSat-12 mission by Lawrence for his PhD research [4]. Six metrics were identified for small to medium spacecraft flying in Low Earth Orbits (LEO) and seeking to control their orbits, to operate as constellations and to prolong life against the perturbations such as drag. Today, the ability to conduct an end-of-life manoeuvre to comply with UK, US and other regulations relating to sustainability is a further mission performance and cost driver.

It was determined [4] that overall system cost, power, volume, mass, integration complexity and thrust were the key propulsion system metrics for small spacecraft, and assessed in detail <u>specific thrust</u> (input power divided by thrust), where a low value is targeted; and <u>density specific impulse</u>, which is a measure of both mass and compactness of the system able to achieve a given mission. Aside from preheat power to prolong catalyst life, hydrazine and cold gas nitrogen have lower specific power than any competing system, and outperform an N_2O resistojet. However, the low storage density and high tank mass of pressurised nitrogen make its density Isp, or system compactness and mass to deliver a given Isp, uncompetitive. The cost of hydrazine components, hydrazine handling, and the minimum size of Commercial off-the-shelf (COTS) thrusters being around 1N, challenging the limited ACS of cubesats and microsatellites, also make it uncompetitive. Furthermore, many small launchers and in-space transport providers will not fly hydrazine systems, citing health & safety and cost concerns. The table below shows research [4] that N_2O can be competitive from a specific power perspective but is less so from a compactness viewpoint.

Table 3. Comparison of two major performance metrics for chemical and electric propulsion systems in small spacecraft [4, 15]

System	**Input power / thrust** (W/mN)	**Density Isp** (s), density defined in terms of specific gravity
Nitrogen cold gas	0	7
Hydrazine	0	222
N2O resistojet	0.6	105
Hydrazine resistojet	1.9	304
Water resistojet	2.7	182
Ammonia arcjet	6.5	372
Hydrazine arcjet	9	507
Ammonia resistojet	15	228
Hall effect thruster	16	695
Xe gridded ion thruster	26	982
PPT Pulsed Plasma thruster	27	2000
FEEP	60	11000

It was also determined [4] that a thrust of 50-100mN was sufficient to meet UoSat-12 (a 300kg class spacecraft) manoeuvring requirements, at a power level of 100W electrical. Subsequent development by SSTL has found a thrust of 30mN, but at a lower power of 15-30W [7], to be adequate for manoeuvring a range of modern small spacecraft such as SSTL's 130kg Carbonite platform [16].

As a result of the above exercise and findings, some idealised design targets were identified as aims for the current project. An ability to support spacecraft modifying their orbits ('deltaV' manoeuvres) and carrying out proximity operations plus attitude control ('impulse bit' operations) is desirable to maximise customer interest. Thrust is of particular importance, high values reducing required manoeuvring time for orbit changes, lower thrust and precise on / off values, or small 'impulse bits' offering an ability to carry out precise Rendezvous and Proximity OperationS (RPOS).

This work is at an early stage and the technology's performance envelope and many aspects of its operation remain under investigation; it is therefore unclear whether these aims can be met, and they are included as idealised outcomes that are being pursued. Thruster and system level objectives are summarised below. Deriving realistic requirements that consider the achievable density specific impulse of N_2O monopropellant and consider the unknowns about inductive heating (see section 6), is receiving considerable attention by the NLS and REL design teams.

Table 4. Summary of idealised aims for N_2O monopropellant propulsion system design

Thruster-level	
Aim	**Description**
1	At least 100mN to be competitive with existing small spacecraft manoeuvring systems. Justification for targeting 500 or 1000mN To Be Confirmed TBC.
2	Specific Impulse of 180s for long duration (deltaV) burns. A reduced Isp for short pulses during RPOS manoeuvres would be accepted.
3	Target spacecraft mass: applicable to a range between 5 and 150kg.
System-level	
4	Envelope: 150x100x100mm (excluding nozzle), or 1.5U equivalent.
5	Deliverable impulse of at least 1000Ns, offering a potential deltaV approaching 200m/s on a 6U cubesat, which would allow for deOrbit from typical LEOs

The above requirements are subject to ongoing customer engagement, both in the UK and Europe; and a watching brief on competitors, where a significant number of entities are operating and entering the market. Changes in future to the above requirements based on technical solutions feasibility and customer 'ask' are not ruled out.

6. DESIGN CHALLENGES (THRUSTER)

Consistent repeatable decomposition of N_2O creates a challenging environment for operation, with gas flow at a maximum temperature around 1300ºC containing a high proportion of oxygen radicals. In addition to this, ignition requires an extended warm-up period with at least parts of the thruster core reaching close to 800ºC to trigger a self-sustaining reaction. Added to this, the thruster core requires an induction heater coil in close proximity. These devices generate a powerful localised magnetic field as part of their operation, and will thus need shielding and careful design to ensure they do not cause electromagnetic interference for the satellite payload.

A coaxial design, taking into consideration previous work on directly heated resistojets such as shown below, is being developed.

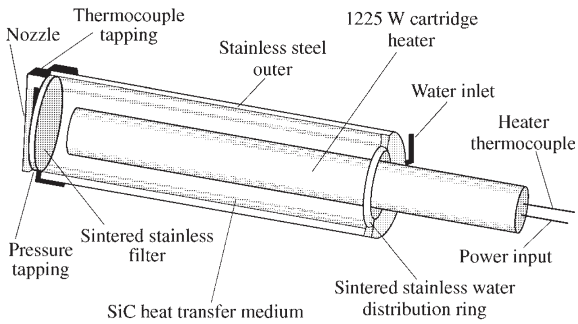

Figure 1. Early University of Surrey Space Centre resistojet design using coaxial, resistance heating [17]

Broadly, the main issues for design relate to the design of the thruster core - the induction heater and heat exchanger - with other challenges, such as thermal and power management, driven by the precise architecture of these core components.

COTS induction heater systems, used by NLS for its 2021 study [18], were found to be poorly suited to space applications. In particular, the frequency at which the induction coil alternates its current is highly specific to particular materials to be heated, with some frequencies working very well with some materials, and not others. As a result, COTS induction heaters are usually designed to work with specific workpieces and materials, and thus do not work efficiently with the novel geometry and materials required for a small satellite thruster.

COTS components are also designed to have the size of the workpiece limited to the central half of the coil radius, with a large gap between the workpiece and coil; this is less time and energy efficient, but draws less peak power and reduces peak swings in current. This results in a more straightforward electronics arrangement that does not need especially resilient components, but limits how quickly heat is induced, thus allowing more heat leakage and is not ideal for this system's high temperature requirement. Arranging the heat exchanger/element in a more time-optimal configuration results in a system where even small alterations to internal geometry or position, produce significant variability in the induction heater's current draw, pushing beyond the rating of many COTS power electronic components.

Taken together, the two previous issues highlight the need for a highly customised induction heater to be designed around the selected heat exchanger geometry and material, with the challenges of operating in a vacuum taken into account.

Even with such a customised induction heater system, there remains a significant thermal management challenge: to prevent excess leakage of heat from the chamber, before self-sustaining decomposition is triggered, but which also then allows heat from decomposition to be passed to the flow, without soakback to induction heater electronics, sensitive fluid feed system components such a valves, or the host spacecraft. Electrically conductive materials suitable for the induction coil are also thermally conductive, and can quickly channel heat to the more delicate electronic components if not managed adequately.

The project is currently evaluating material optimisation for a coaxial design, with three specific challenges to solve for the heated core where N_2O decomposition is initiated, requiring suitable materials for (1) an inductively heated material, (2) a heat exchanger medium to efficiently transfer heat from the induction heated material to the gas flow, and (3) a material protecting the induction coil itself from direct thermal soak back from the heated material and decomposing gas. These materials must have a combination of magnetic, thermal and oxidation resistant properties. Other thruster elements including a downstream chamber and nozzle, and an upstream thermal standoff and mounting flange are also being considered.

7. SUMMARY OF WORK TO DATE

Work to date has focused on two areas of research:

1. The induction heater/element
2. The heat exchanger

The induction heater development is a combination of electronics and materials selection. A bespoke induction heater circuit has been designed and built to act as a resilient and flexible test-bed for the development programme. This will ultimately be refined for use in the proposed thruster. Heater elements have been built of varying geometries and materials to test how quickly and efficiently the target temperature of 800 ℃ can be achieved. This not only depends on the rate at which heat is supplied to the heater element, but also the rate at which it leaks into the surrounding material and environment. Holding the heat within the heater element is a significant challenge.

Once the heater element has reached the desired temperature, the heat must be passed into the N_2O in order to trigger decomposition. Several designs of heat exchanger have been trialled, including stacked plates and stainless steel wire wool. By far the most promising in terms of heat transfer and resilience is the pebble bed style heat exchanger, in which the N_2O is passed through several layers of granular ceramic material. Initial tests of this arrangement (heated in air) have demonstrated the ability to generate temperatures sufficient to initiate N_2O decomposition, with the heat rapidly conducting from the element through the ceramic bed. A photograph of the pebble bed heat exchanger under test is shown in Figure 2. Tests have been carried out in which oxygen was blown through the heated pebble bed to evaluate its survivability. The next step will be to replace the oxygen with N_2O to ensure that the ceramic material can survive the much higher temperature (1600 K) associated with decomposition.

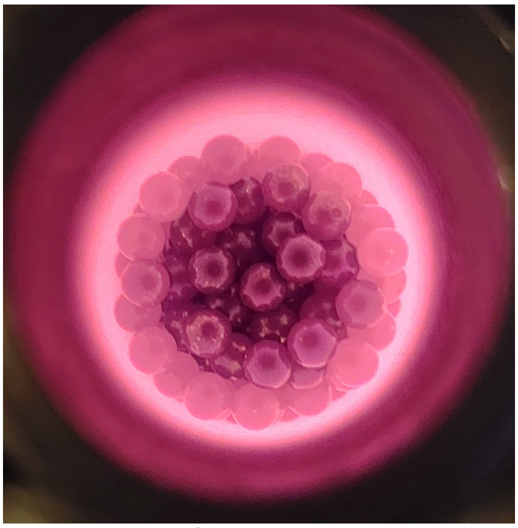

Figure 2. Axial view of 'pebble bed' heat exchanger test, showing temperature of >700°C. The workpiece (in context, the source of heat) is the inner chamber wall.

Figure 3 shows the heating curves for alternative heat exchanger materials. These test data demonstrate the promising performance of the ceramic pebble bed approach, but also highlight that there is still considerable optimisation required, especially if the proposed thruster is to be used for applications that need a rapid response.

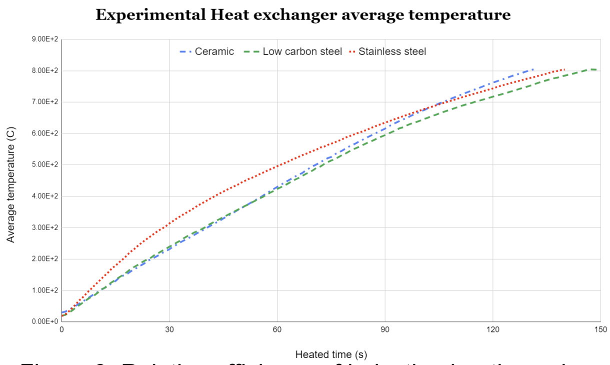

Figure 3. Relative efficiency of induction heating using different heat exchanger/ workpiece materials.

8. CONCLUSIONS AND NEXT STEPS

Newton Launch Systems and Rocket Engineering Ltd, with UKSA grant support, are developing a small, low cost, re-startable monopropellant thruster targeting a density impulse and power specific thrust competitive with other low-cost small spacecraft propulsion thrusters.

The key goal for the project at present is advancement of the heat exchanger design, following which a prototype decomposition chamber will be manufactured for hot-fire testing with N_2O. A custom induction heater optimised for the selected heat exchanger geometry and materials is being constructed. Work is progressing on materials selection addressing the unique requirements of the thruster, with ongoing supplier discussions.

The team will then consider the system-level design and supporting components, leading to a breadboard prototype of a complete thruster. Pending positive results from prototype testing, a breadboard induction heated N_2O monopropellant thruster, tested in a representative environment, meeting the requirements summarised earlier project plus a reference design for a propulsion system are anticipated in early 2025.

(see next page for references)

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