An overview of development model testing for the LEROS 4 High Thrust Apogee Engine

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***Abstract*—The LEROS 4 High Thrust Apogee Engine is a European Space Agency funded 1100 N storable propellant spacecraft main engine undergoing development by Moog. The engine will uniquely support the agencies future interplanetary exploration missions by reducing the mass of spacecraft propellant required for orbit insertion manoeuvres and so allow increased scientific payload to be accommodated on these missions.**

The first phase of the project has been completed with a formal design review of an extensive range of tested development model engine hardware and a proposed flight engine design.

An overview of this first development phase is given and covers the status of the flight engine design, the supporting materials investigation, the challenges faced in narrowing down a broad design parameter space for development model engine testing, the hardware manufactured to support test campaigns, a summary of cold-flow and sea-level hot-fire testing, and the design development plan for the next phase of the project.

Keywords*—liquid apogee engine, storable propellant, development model, experiment design, testing.

I. Introduction

THE High Thrust Apogee Engine (HTAE) is a European Space Agency (ESA) technology development in response to the needs of future European space exploration missions. It is part of the Mars Robotic Exploration Programme (MREP) and was one of a number of technology pre-developments initiated to establish technical credibility and maturity sufficiently early in future mission planning and so remove these technologies from the critical path [1] [2].

Studies conducted by ESA compared the use of an 1000 N engine to that of a 500 N engine and showed that, even with a specific impulse of 320 s, significant propellant savings could be made when working against the Martian gravitational field during orbit insertion e.g. for a spacecraft arrival mass of 4000 kg and hyperbolic arrival velocity of 3 km/s, approximately 200 kg of propellant savings were estimated. This reduction in the propellant required can be directly translated to an increase in bus and payload mass, enabling better science return on these missions.

TABLE I. Main LEROS 4 technical specifications

Specification	Value	Units
Fuel	MMH	-
Oxidant	MON-3	-
Propellant feed pressure	1.54	MPaA
Propellant feed temperature	293	K
Vacuum specific impulse (minimum)	320	s
Vacuum specific impulse (target)	323	s
Vacuum thrust (nominal)	1100	N
Vacuum thrust range	900 to 1300	N
Overall O/F mixture ratio (nominal)	1.65	mass fraction
Overall O/F mixture ratio range	1.5 to 1.8	mass fraction

The LEROS 4 technical specification given in Table I derives from these mission studies. The engine uses Mono Methyl Hydrazine (MMH) as a fuel and Mixed Oxides of Nitrogen 3 (MON-3) as an oxidant i.e. dinitrogen tetroxide with 3 wt% nitrogen monoxide. Given a propellant feed pressure of 1.54 MPaA and temperature of 293 K at Normal Design Point (NDP), it is being designed to produce a vacuum thrust of 1100 N and a vacuum Specific Impulse (Isp) of 323 s at a Mixture Ratio (MR) of 1.65. The *LEROS* designation is given to the HTAE as its design draws from the heritage of production LEROS apogee engines such as the LEROS 1, 1c, 1b and 2b [3].

In 2011, Moog ISP (formerly known as AMPAC-ISP) won the development contract to design the engine. The overall engine design and development work is led from its Westcott Operations in the United Kingdom, with valve design and development taking place at its Dublin Operations in Ireland.

Phase 1A was essentially a paper study which was conducted from October 2011 to October 2012 [4] [5]. It produced a conceptual flight engine design and a conceptual Development Model (DM) engine design, both of which successfully passed an ESA Baseline Design Review (BDR).

In Phase 1B, which was conducted from November 2012 to December 2013, the DM engine was designed in detail, built and successfully sea-level hot-fire tested. The flight engine design was iterated based on these test results as well as results from a parallel laboratory-scale materials test campaign.

Phase 1 was completed with the engine development having progressed to Technology Readiness Level (TRL) 4 on the European Cooperation for Space Standardization (ECSS) scale.

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II. Overview

The availability of trustworthy rocket engine performance data is essential for the design and optimisation of new engines.

A typical liquid apogee engine scheme could be defined as an engine with: pressure-regulated hypergolic liquid bi-propellant feed, thermally isolated solenoid-actuated valves, injector assembly containing (though dependent on the injector) central oxidant gallery and outer fuel gallery, radiative and film cooled combustion chamber, and Characteristic Velocity (C^*) limited by combustion chamber material thermal capability.

Apogee engine schema have not changed significantly over the past few decades, yet there is a lack of accessible experimental data to aid efforts to evolve such engines for new applications.

The goal of Phase 1B was to front-end load the project with flight-representative hardware and test data in order to inform a higher fidelity flight engine design and de-risk later development phases.

The main activities which took place in Phase 1B were:

1) Materials investigation. A materials test campaign was conducted to study candidate high temperature capable combustion chamber materials. This testing activity involved the National Physical Laboratory (NPL), with contributions from The Welding Institute (TWI), both in the United Kingdom. A coating development program was also conducted to investigate a novel higher temperature capable coating for an industry-standard combustion chamber alloy. This coating activity involved Archer Technicoat Limited (ATL) in the United Kingdom, with contributions from Enbio in Ireland.
2) DM hardware design and build. Detailed design work was performed to fully define the DM engine. This included DM valves as well as a large number of injector and combustion chamber variants made from industry-standard materials. A full-scale combustion chamber made from one candidate higher temperature capable material was also designed and built. This latter activity involved the Japanese Aerospace Exploration Agency (JAXA) and Mitsubishi Heavy Industries (MHI) in Japan.
3) DM engine testing. An extensive test plan was defined for the DM engine which allowed the design parameter space to be surveyed under a range of engine operating conditions. Engine-level test campaigns were conducted at Moog's Westcott Operations using cold-flow and hot-fire test facilities which were custom upgraded to meet the unique requirements of the DM engine test plan.
4) Flight engine design iteration. All test data accumulated from laboratory-scale materials testing and DM engine testing was used to inform flight engine design concepts which met ESA technical specifications.

III. Materials investigation

The three major engine components in the materials selection challenge were the combustion chamber, expansion nozzle and the injector assembly. The choice of the combustion chamber material is the most important. Its selection limited the choices available for the adjacent expansion nozzle and injector assembly.

At the start of the investigation the best material candidates for the combustion chamber were:

1) Pt alloys sourced from Heraeus Materials Technology in Germany.
2) The refractory ceramic SN282 sourced from Kyocera Coporation in Japan.
3) The refractory composite C/SiC sourced from Herakles in France.
4) A novel Ir/W coated Nb alloy to be developed by ATL in the United Kingdom.

The trade-off performed between these materials could not be based solely on thermal and mechanical capability considerations. There were a number of other import factors which complicated the materials choice, such as: availability of the material, raw material dimensional limitations, manufacturing constraints, ease of joining to other materials and overall engine cost.

At the end of the investigation, the leaders in the trade-off were Pt alloy, SN282, and Ir/W coated Nb alloy. All required compromises in the flight engine design and, to further complicate the trade, *different* compromises.

A. Platinum alloy

As Pt alloys were a flight proven and extensively used material, they were considered a conservative choice and were selected as a tentative baseline.

Structural-thermal analysis was performed [6] using a standard set of geometries and thermal profiles to down select and prioritise Pt alloys for material property investigation at NPL. These analyses used temperature extrapolations of material property data made available through a literature study, however results at the operating temperature profiles considered for the engine showed no clear leader among the alloys. The following candidates were taken through to laboratory-scale testing of specimens: Pt10Rh (Pt with 10 wt% Rh), Pt20Rh, Pt30Rh and Pt20Ir. The goals of this test campaign were to confirm literature data, fill in gaps in literature data and extend literature data to the higher temperatures. The tests produced data on elastic modulus, thermal diffusivity, coefficient of thermal expansion, Poisson's rato, emissivity, and oxidation-related changes to some of these properties. A materials test campaign enabler was NPL's Electro Thermal Mechanical Testing (ETMT) system which was essential for gathering high temperature stress-strain data.

Based on the test results, Pt alloys were re-ranked with Pt20Ir and Pt20Rh highest. However it was noted that material properties obtained were very sensitive to the condition of the original specimens, i.e. the specifics of how the specimens were manufactured.

There were quite a few constraints which drove the design towards segmentation of the combustion chamber and expansion nozzle shown in Figure 1.

The combustion chamber is made up of a cylindrical section near the injector face (chamber segment 1), joined to a

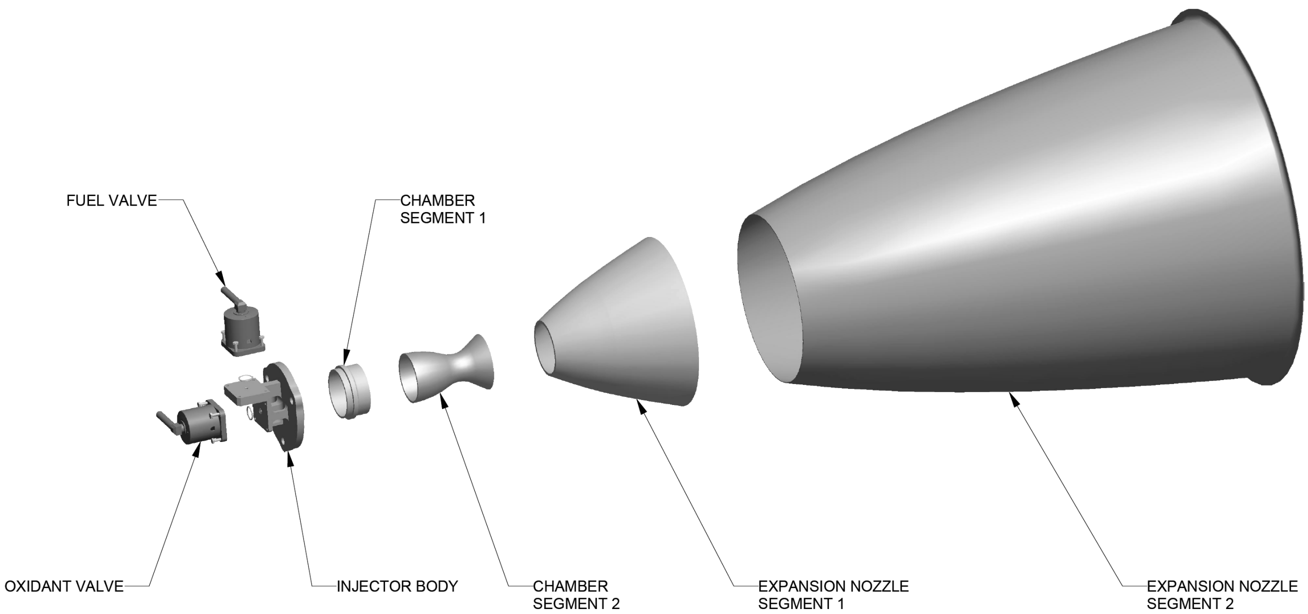

Fig. 1. The Phase 1B baseline flight engine design. The components shown are [left to right] : solenoid-actuated oxidant and fuel valves, injector assembly made from Inconel 718, cylindrical combustion chamber segment made from Pt20Rh, convergent-divergent combustion chamber segment made from Pt20Ir, first expansion nozzle segment made from Pt20Rh and second expansion nozzle segment made from Inconel 718. Due to the manufacturing limitations of Pt20Rh and Pt20Ir, the engine would require segmentation of the combustion chamber and expansion nozzle. Though considered a conservative baseline due to use of flight-proven Pt alloys, the required segmentation introduces engine-level complexity. In assembled form, the engine bounding cylindrical envelope is approximately 1 m long and 0.5 m in diameter.

convergent-divergent section (chamber segment 2). The Pt20Ir alloy has higher yield strength than Pt20Rh at room temperature and this property is of benefit when considering random vibration loading. It was the preferred chamber material as the dominant vibration mode for the engine is a bending of the expansion nozzle about the throat. Pt20Ir however can only be cast to limited cylindrical dimensions so this meant it could only be used for a limited section of the combustion chamber. It was positioned in the highest temperature region near the throat as chamber segment 2, but then required joints both upstream and downstream of this segment.

Chamber segment 1 could have been made from Pt20Ir, however Pt20Rh was selected as it had proven weldibility to the chosen injector assembly material Inconel 718. The joint in the high temperature region downstream of chamber segment 2 could not be made to another Pt20Ir segment, as Pt20Ir cannot be easily shear formed to the contour required for the expansion nozzle. This joint was instead made to Pt20Rh. Though cast Pt20Rh billets are also dimension limited, the alloy can be shear formed, is high temperature capable and can be easily joined to Pt20Ir.

To limit the cost of the expansion nozzle, the Pt20Rh expansion nozzle segment 1 was deliberately limited in size. The full expansion nozzle area ratio of 293:1 was created by joining to expansion nozzle segment 2. Inconel 718 was tentatively selected for this second expansion nozzle segment due to its weld compatibility with Pt20Rh. It is not an ideal choice given the size of the expansion nozzle, the material density and the dominant vibration mode. The position of the joint between expansion nozzle segments was chosen based on wall temperature profile and this determined the size of the Pt20Rh expansion nozzle segment 1.

Pt alloy does have high thermal conductivity and this would increase thermal loading on the valve, further complicating the engine design. However, this engine development benefits from a custom designed valve.

Given the complexity introduced in the segmentation of the thrust chamber assembly, and the additional challenges for the valve design, it is difficult to regard Pt alloy as a true *conservative* baseline when one considers the engine as a whole.

B. Silicon nitride

Silicon nitride in the form of SN282 does offer an attractive alternative to these Pt alloys. In comparison, the ceramic is also flight-proven, has sufficiently high thermal capability to allow for sufficiently high C* efficiency, has much lower thermal conductivity so will offer minimal thermal loading on the valves, also requires no internal wall coating, and it is less expensive.

NPL conducted testing on two different silicon nitride materials; SN282 specimens supplied by MHI, in parallel to S10 specimens supplied by Dynamic Ceramic in the United Kingdom. The aim of the laboratory-scale testing conducted was broadly similar to that of Pt alloy testing and produced data on: flexural strength, thermal diffusivity, coefficient of thermal expansion, and oxidation-related changes to some of these properties. The results showed that SN282 was overwhelmingly superior to S10 for the high temperature operating conditions under consideration, and this is believed to be partly due to the thermal capability of the powder sintering agents used.

The low reactivity of silicon nitride aids in its high temperature survivability, but this also means that it is difficult to join to other materials.

SN282 has its own manufacturing limitations and can only be manufactured to a size that would allow a monolithic thrust chamber assembly with an exit-to-throat area ratio of approximately 100:1 to be constructed. Though very compact, such an engine would be Thrust Coefficient (C_f) limited and unable to meet the 323 s Isp target. To increase Isp further requires an expansion nozzle segment (likely metallic) to be joined.

The engine design is therefore driven towards the use of bolted joints. A bolted joint at the injector end has been flight-proven for this material and may be acceptable, but such a joint at the expansion nozzle end may prove challenging.

The need to develop a reasonably high operating temperature SN282-to-metallic joint for the expansion nozzle region is essential to keep silicon nitride as an option in later development phases. While from a materials properties point of view the material is a good Pt alloy alternative, it creates development challenges at engine-level.

C. Iridium and tungsten coated niobium alloy

The Ir/W coating under development by ATL was based on the promising results from a limited investigation by Ultramet and Aerojet. The initial coating development work by ATL was funded through a United Kingdom Space Agency (UKSA) 'Pathfinder' study, however the results led Moog to fund additional development work. Given the engine-level challenges of using Pt alloy and SN282, this coating may allow the industry-standard niobium alloy, C103, sufficiently long life at high temperature than it is presently limited to by its industry-standard oxidation resistant coating, R512E.

Ir coupled with an appropriate interlayer such as W can:

1) Prevent oxygen diffusion into the underlying material.
2) Offer a compliant layer to prevent any thermal expansion mismatch between coating and substrate.
3) Discourage the formation of brittle inter-metallics between substrate and coating.

A chemical vapor deposition reactor was built by NPL for use with particular Ir and W precursor materials, and coating trials were performed on flat plate and rounded test specimens of C103. This niobium alloy was chosen as a reference base material with which to compare coatings trials on other Nb alloys in later development phases.

The use of Ir/W on Nb alloy would not deviate too far from heritage LEROS engines and therefore represents a lower development risk evolutionary change, rather than a revolutionary change. However, it is at the lowest TRL from all three combustion chamber material candidates and still requires significant development effort.

IV. Development model design and build

In Phase 1 a bolt-up DM engine approximating the flight engine was designed and built for testing. The testing was performed at sea-level altitude instead of simulated high altitude

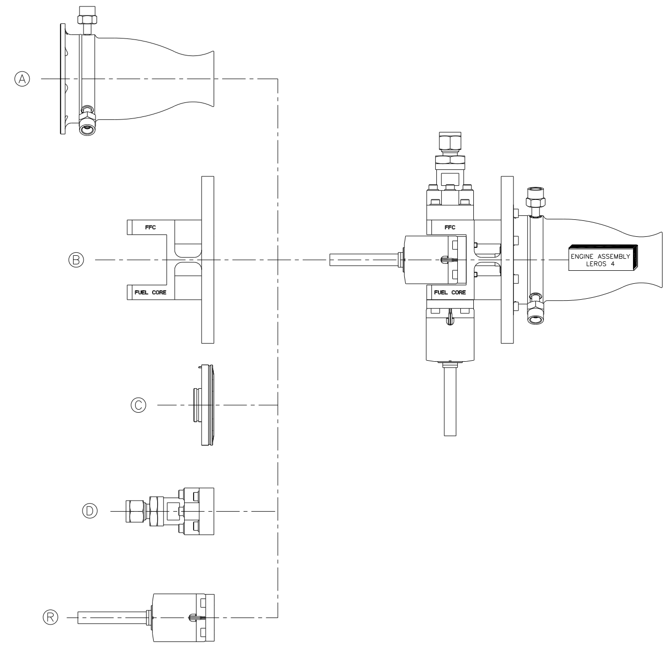

Fig. 2. Development Model (DM) engine. The *main components* used in a typical configuration of the DM engine are: [A] the combustion chamber (approximately 120 mm long), [B] the injector body, [C] the injector plate, [D] the Ground Test Valve Outlet Assembly (GTVOA) and/or [E] the DM valve. The DM engine on the right uses DM valves for core propellant feed and a GTVOA for the fuel film cooling feed. For the majority of the test campaign these GTVOA's, which simulated the flow profile and pressure drop of the DM valves, were used for all three propellant feeds. Design parameters were studied under test by using many variations of the combustion chamber and injector plate shown.

conditions. The use of a sea-level facility decoupled the engine development testing problem at the sonic throat, allowing the valve-injector-chamber development to be satisfactorily advanced before needing to include the expansion nozzle - a design, build and test activity for the next development phase. Given the extensive test plan that was implemented, the choice of a sea-level facility allowed for quick DM engine reconfiguration between tests and resulted in a significant reduction in development testing cost and duration.

The reconfiguration focused on the alteration of propellant injector and combustion chamber flow-related geometry. In the flight engine the propellant injector is a "plate" installed within the injector assembly. The injector assembly is manifolded on the oxidant side to channel oxidant from the oxidant valve to a central gallery feeding all Oxidant Core (OC) injector elements. It is manifolded on the fuel side to channel fuel from a centerline off-set fuel valve to a single annular fuel gallery which feeds both Fuel Core (FC) and Fuel Film Cooling (FFC) injector elements. The injector has unlike impinging jet core elements arranged in a circular layout, and jet film cooling elements arranged in a circular layout close to the combustion chamber inner wall. The combined flow from the core elements forms a reacting spray cone within the combustion chamber. The flow from FFC elements is directed toward the wall to create a uniform fuel film along the chamber wall.

Figure 2 gives an overview of a typical configuration of the DM engine built. The design approach used was to:

1) Maintain the design as close to the flight engine as possible so as to be thermally representative.
2) Include pressure taps in the combustion chamber to allow pressure measurement.
3) Allow for bolt-on *workhorse* chambers to be easily removed and replaced so that different chamber geometries could be investigated.
4) Allow for injectors to be easily removed and replaced so that different injector element parameters could be investigated.
5) Use seal interfaces instead of weld interfaces to allow components to be easily removed and replaced.
6) Split the annular fuel gallery into two separated annuli such that FFC and FC flow rates may be manipulated independently.

The material selected for workhorse chambers was the industry-standard C103 alloy with oxidation resistant R512E coating. It was cost effective, well-understood, and offered the high reliability. The chambers were conventionally turned from bar stock C103 from ATI Wah Chang and coated with R512E at Hitemco. Chamber pressure was tapped at two places in the uncooled region of the wall near the injector face, and between FFC jets to avoid disturbing the FFC layer. The chambers were truncated designs with an 1.8:1 exit-to-throat area ratio which was more than sufficient to allow full-flow of combustion gases under sea-level testing.

Following a similar justification, the material chosen for the injector assembly was Ti6Al4V, the industry-standard companion to R512E/C103. The injector plate as well as the injector body within which the injector plate is housed were conventionally turned from bar stock. The injector element holes were conventionally drilled on most injectors at Moog Westcott Operations however Electro-Discharge Machining (EDM) by SARIX in Switzerland was used for a few injector plates. Unlike micro drilling, the EDM technique is unconstrained by length-to-diameter ratio and allowed for more design freedom. However, performance sensitivity related to manufacturing repeatability and hole taper still need to be thoroughly investigated in the next phase of development.

The use of seals on all interfaces consumed valuable space within the injector assembly and made the requirement of flight representation challenging to achieve. The annular fuel gallery division posed a particularly difficult challenge to incorporate into the design. Should the dividing seal here be damaged during assembly, fuel would leak between the FC gallery and the FFC gallery, shifting core MR and changing the fuel allocation to boundary layer cooling. Appropriate modifications to the hot-fire test site allowed the mounted DM engine to be cold-flow tested and the integrity of this inter-gallery seal to be checked ahead of testing. Such a gallery division is expected to pose a greater challenge to incorporate into smaller engines. With regular replacement, Ethylene propylene (fuel side) and Viton (oxidant side) seals were found to be sufficient for the DM testing conducted with no need to resort to the higher temperature capable Kalrez seal.

The injector body also had to accommodate three different chamber interfaces. There were two diameter variations of the R512E/C103 chambers and an SN282 chamber. This ceramic chamber was designed in conjunction with MHI, and was manufactured in Japan from powdered raw material through a process of forming, sintering and machining. Compared to R512E/C103, this chamber was considered more sensitive to the engine thermal profile and its wall thickness profile underwent rigorous analysis before the design was frozen for manufacture. One of the challenges of using a monolithic ceramic combustion chamber was in incorporating an appropriate chamber flange clamping assembly into the design. The need to clamp this chamber using a clamping ring, rather than bolt through the chamber flange as with the metallic workhorse chambers, is due to three particular design rules which needed to be followed when designing for ceramics. Firstly, designing for compression rather than tension; in general ceramics are stronger under compression and this is why the compressive loading of a chamber flange clamping ring is more acceptable. Secondly, designing to accommodate differential expansion; the low thermal expansion coefficient of SN282 compared to the adjacent Ti6Al4V injector assembly required detailed design work to ensure stress on the chamber flange due to differential expansion during a test firing was within allowable limits. Lastly, designing to avoid features that introduce surface scratches or regions of stress concentration; the introduction of through-bolts would introduce sharp corners, surface scratches and differential expansion stress. Surface condition is carefully controlled during the manufacturing process and monitored during the handing of these chambers.

DM solenoid valves, based on the ESA-funded and qualified Apogee Engine Valve (AEV) were designed and built by Moog Dublin Operations [7] for use with this DM engine. The use of a 500 N class apogee engine valve would incur too high a pressure loss and force margin penalty at the flow rates of an 1100 N engine. No known valves exist that can meet these two headline requirements for this engine. The requirement for low pressure drop is due to propulsion feed system pressure budget constraints and the requirement for high force margin originates with ECSS guidelines. Low pressure drop, high force margin, low mass and small footprint are all competing constraints that were investigated. The High Thrust AEV (HTAEV), into which the DM valves would eventually evolve, would therefore be unique to this class of engine. Most of the early DM engine testing was conduced with Ground Test Valve Outlet Assemblies (GTVOAs) as shown in Figure 5. These were static bolt-up assemblies which mimicked the DM valves in flow profile and pressure drop. They were part of a larger assembly called the Ground Test Valve (GTV) that interfaced the DM engine with the test facility.

In essence, the DM design allowed for easy reconfiguration of the hardware: valves, injectors and chambers.

V. Development model testing

The full design parameter space translates into a number of hardware configurations that should ideally have been tested. These hardware configurations mainly consist of a particular combination of chamber geometry and injector geometry.

In full factorial testing, a particular design parameter, such as the net momentum angle of an unlike impinging doublet injector element, can be studied by varying it whilst keeping all others fixed. In such testing the number of hardware configurations is proportional to the number of variations (or levels) to be studied raised to the power of the number of parameters. Given that a minimum of three variations are required to find an optimum, this can translate to a large number of configurations.

With both time and cost aspects to consider for a hot-fire test campaign, the number of configurations must be reduced significantly below this ideal i.e. fractional factorial testing. The engine design was parametrised guided by the literature for such engines, and the parameters were screened categorising them according to those which:

1) could be designed based on threshold criteria e.g. jet free stream length
2) have no known design criteria but are known to have some influence on the performance e.g. chamber cross-sectional mass flux distribution
3) primarily have a local mixing influence e.g. core injector element diameter ratio
4) primarily have a bulk mixing influence e.g. core spray net momentum angle

The thrust and MR range specified for the engine spans 900 N to 1300 N, and 1.5 to 1.8 respectively. This is a very broad thrust range for a liquid apogee engine and poses a particularly difficult thermo-acoustic challenge. The design approach in this case was to design for stability. The intent was to obtain good hot-fire test data on how parameter variation affects performance, avoiding conditions where test runs would need to be aborted early, rather than attempting to design for optimum performance from the outset. To this end there were actually injectors which were deliberately designed for low performance in order to understand the extremes of parameter variation.

The main hardware components manufactured to study parameter variation were:

1) 39 injectors (among these were nine EDM injectors, three of which were duplicates to study manufacturing variability)
2) 6 chambers (among these were two R512E/C103 duplicates)

A. *Cold-flow testing*

It was assumed that good cold-flow injector performance would be no guarantee good hot-fire performance. It was considered more valuable to bias efforts towards coupled injector-chamber parametric studies under hot-fire testing. Cold flow testing was used to clear injectors for hot-fire testing by detecting obvious manufacturing defects, and to provide some level of information on the relative behaviour of injectors that could aid better understanding of the hot-fire testing results.

The cold-flow propellant injector spray studies performed were basic checks and though there are much more advanced techniques to characterise propellant sprays, such as laser doppler anemometry, these basic checks were cost effective, easy to implement, and provided a sufficient level of fidelity.

Cold-flow testing was performed using a custom built rig shown in Figure 3. Water was used as a propellant simulant, and the testing was conducted by flowing through the DM engine to atmospheric pressure.

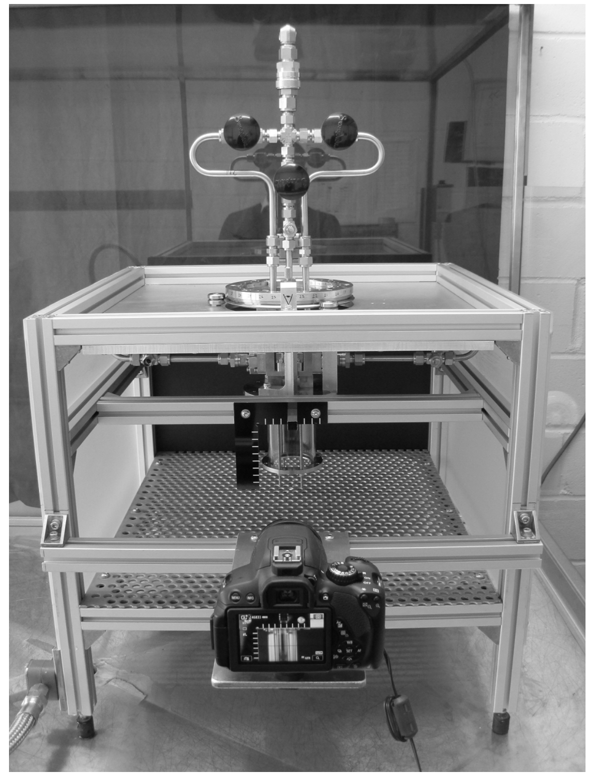

Fig. 3. The custom built cold-flow test rig used for propellant injector characterisation with water as a propellant simulant. The rig allows water flow to each of the three propellant manifolds in the development model engine to be controlled sufficiently accurately to simulate different engine thrust levels, core mixture ratios, and fuel film cooling ratios. The rotating engine mounting plate (indexable to each injector element), camera mount, imaging reference frame, blackened background and diffuse lighting panels allowed for propellant injector spray imaging studies to be conducted in a repeatable manner.

The rig allows water flow to each propellant manifold to be controlled sufficiently accurately to simulate different engine thrust levels, core MR's, and FFC %'s (with appropriate adjustment of mass flow rates based on the density difference with the actual propellants).

By configuring the rig in different ways:

1) FC, OC and FFC manifold pressure drops were directly measured
2) Core spray, and FFC spread along the combustion chamber wall, were directly imaged using a digital camera with appropriate lighting
3) Core spray mass flux distribution was directly measured using a ”mechanical patternator” approach

Repeatable measurements were possible as the relative distances between camera, lighting, propellant injector, and spray collection matrix could be controlled. Further, the injector assembly was mounted on a rotating head which allowed each injector element to be numbered and indexed individually with respect to the observer.

A typical core spray image is shown in Figure 4. The image was taken at a target thrust level of 200 N and 1.65 MR. The lower flow rate was selected as with increasing flow rate (or thrust) the impingement point becomes increasingly obstructed

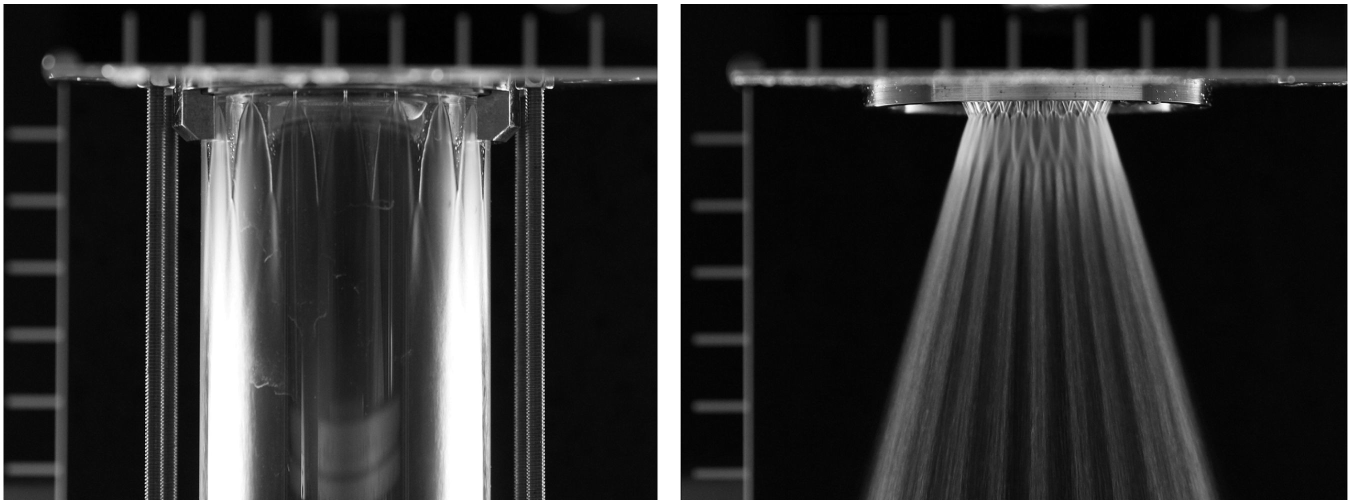

Fig. 4. Typical cold-flow test images used for characterisation of propellant injectors. [Right] Core impingement point imaging. This is a typical picture taken for an injector when studying core injector element jet-jet impingement. The test was performed at a water equivalent flow rate corresponding to an engine operating point of approximately 200 N and 1.65 mixture ratio. [Left] Fuel film cooling spread imaging. This is a typical picture taken for an injector when studying fuel film cooling of the combustion chamber wall. A transparent glass cylinder was used as a chamber proxy. The testing was performed at water equivalent flow rates corresponding to different levels of fuel film cooling.

by atomised spray. As the feed conditions were measured, the achieved thrust level and MR could be accurately determined. By rotating the injector assembly and indexing to each injector element, close-up images allowed the major symptoms of manufacturing defects to be checked, such as jet breakup ahead of impingement, and jet-jet misalignment at the impingement point.

Information on the relative level of spray generated at higher flow rates by different injector designs could also be evaluated, providing an understanding of the degree to which errant core spray was likely to affect the liquid FFC region under hot-fire test and, after image processing, an indication of whether the achieved net momentum angle for the core spray was in line with the designed value.

At high flowrates, collection of the core spray in a matrix of collection tubes positioned directly below the injector allowed the mass flux distribution to be determined. Whereas the imaging provided mainly qualitative information, and focused on the external structure of the core spray, the *volumetric* study allowed the internal structure to be observed and provided quantitative information on the spray.

A typical FFC image taken is shown in Figure 4. In this test a transparent glass cylinder was used as a proxy for the combustion chamber cylindrical section, where the FFC is expected to be in liquid phase. Images such as these were taken at various FFC mass flow rates corresponding to different FFC %'s at NDP. By rotating the injector assembly and indexing to different FFC injector elements, the images taken could be processed to understand jet-wall impingement and spreading behaviour, bearing in mind differences in surface roughness and surface tension with respect to the actual chamber and propellant. In particular, the lowest FFC % at which complete wall coverage occurs and the distance at which merger occurs could be determined for different chamber diameters and varying FFC injector element parameters.

B. Hot-fire testing

There are many design rules available in the literature, however the quantitative influence of parameter variation on performance is not always clear, especially when considering the details of operating conditions and hardware geometry. With the possibility of the flight engine requirements evolving in the face of changing mission requirements, it was also considered prudent to obtain a good understanding of how the variation of key design parameters affected performance i.e. a design parameter survey.

Hot-fire testing was performed in a custom upgraded sea-level hot-fire test facility [8] built at Moog's Westcott Operations. The test facility interfaced with the DM engine through a Ground Test Valve (GTV) shown in Figure 5. The GTV is a rack-and-pinion mounted three-way ball valve system that controlled propellant feed (OC, FC and FFC) into the engine, nitrogen feed for engine purge and forced convection cooling, as well as water feed for engine decontamination.

To reduce test campaign costs, propellant feed pressure was regulated using nitrogen and propellant feed temperature was not deliberately controlled. The lack of temperature conditioning was noticeable in performance data obtained in the latter part of the test campaign as seasonal change caused propellant temperatures to fall. This was correctable using data from a reference configuration repeatedly tested throughout the campaign. The application of helium regulation and saturation as well as thermal conditioning was left for the next phase of development.

Thrust was not a measured parameter for testing, and performance was instead based on C^*, which incorporated: chamber pressure data, throat diameter dilation from chamber temperature data and propellant mass flow rate data. This reduced the cost of the test campaign but required the use of an assumed C_f in order to estimate engine Isp.

The facility, part of which is shown in Figure 6, was equipped with the following sensors: a roof mounted infra-

Fig. 6. A typical Development Model (DM) engine configuration mounted on the test stand at the Moog Westcott Operations sea-level hot-fire test facility. [Left] A view in front of the mounting plate showing a R512E/C103 workhorse combustion chamber with pressure taps and engine mounted thermocouples clearly visible. [Right] A view behind the mounting plate showing the DM valves connected to the engine.

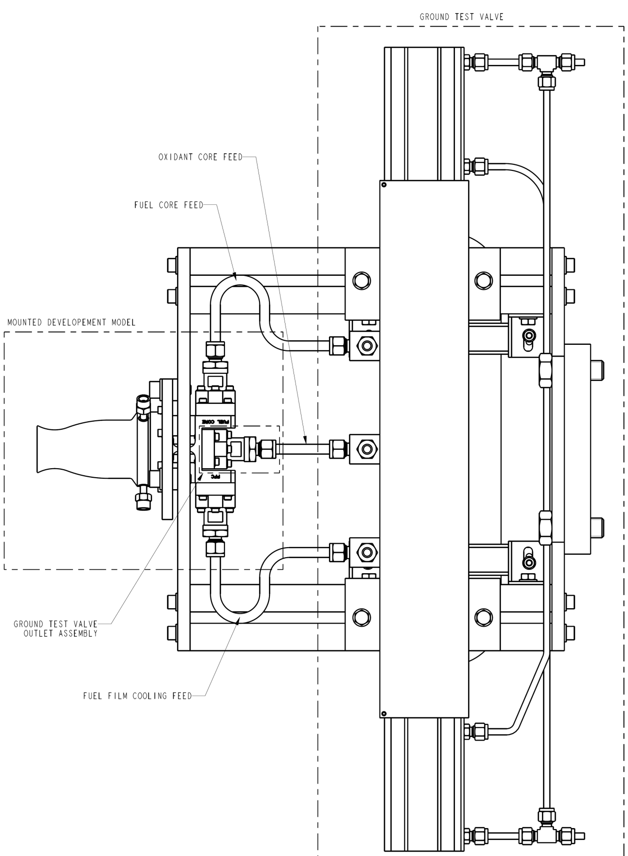

Fig. 5. A diagramatic representation of a typical Development Model (DM) engine configuration mounted on the thrust stand. This is a top view of the horizontal-firing setup used throughout the test campaign. The propellant feed lines are connected to a Ground Test Valve (GTV) on the right. In the majority of the testing conducted, the static GTV Outlet Assemblies (GTVOA)'s shown on the engine were used in place of the DM valves.

red camera with a field of view encompassing the engine, a roof mounted video camera with the same field of view, engine mounted thermocouples, combustion chamber mounted pressure sensors, propellant feed thermocouples, propellant feed mass flow meters and propellant feed pressure sensors. There were many more sensors installed which were feed system related. Pressure transducer data was recorded at 1 kHz and all other data at 100 Hz. For the majority of the testing performed, the engine was stable and the 1 kHz limit proved more than sufficient. In the next phase of development, higher frequency data will be required to deal with potential thermo-acoustic problems, especially when off-nominal propellant characterisation is performed using helium saturated and warmed propellants.

As the development engine allowed FFC and FC to be independently manipulated, the facility was designed to be capable of targeting a particular thrust and *overall* propellant mixture ratio at varying FFC %. This was a tremendous saver of time and hardware as many more operating points could be achieved with the same injector. Although the FC and FFC jet flow properties were altered with varying FFC % at fixed overall MR (changing parameters such as the core spray net momentum angle and the FFC jet-wall impingement behaviour slightly), these changes were quantifiable.

A typical firing is shown in Figure 7, and in a typical day of testing 14 of these test firings were performed. Each operating point in the performance box was first attempted with a 2 second "trim" run. Based on the targeting accuracy, feed system parameters could be adjusted before a longer 20 second test run was performed. The plotted data reveals the exceptional accuracy of operating point targeting using the test facility. The duration of the test run chosen was a compromise made to manage the cost of the test campaign, given the number of hardware configurations that were to be tested. From experience with previous LEROS engine developments, a 20 second run duration was considered sufficient to stabilise C* in a metallic chamber. Figure 8 shows the regions of the

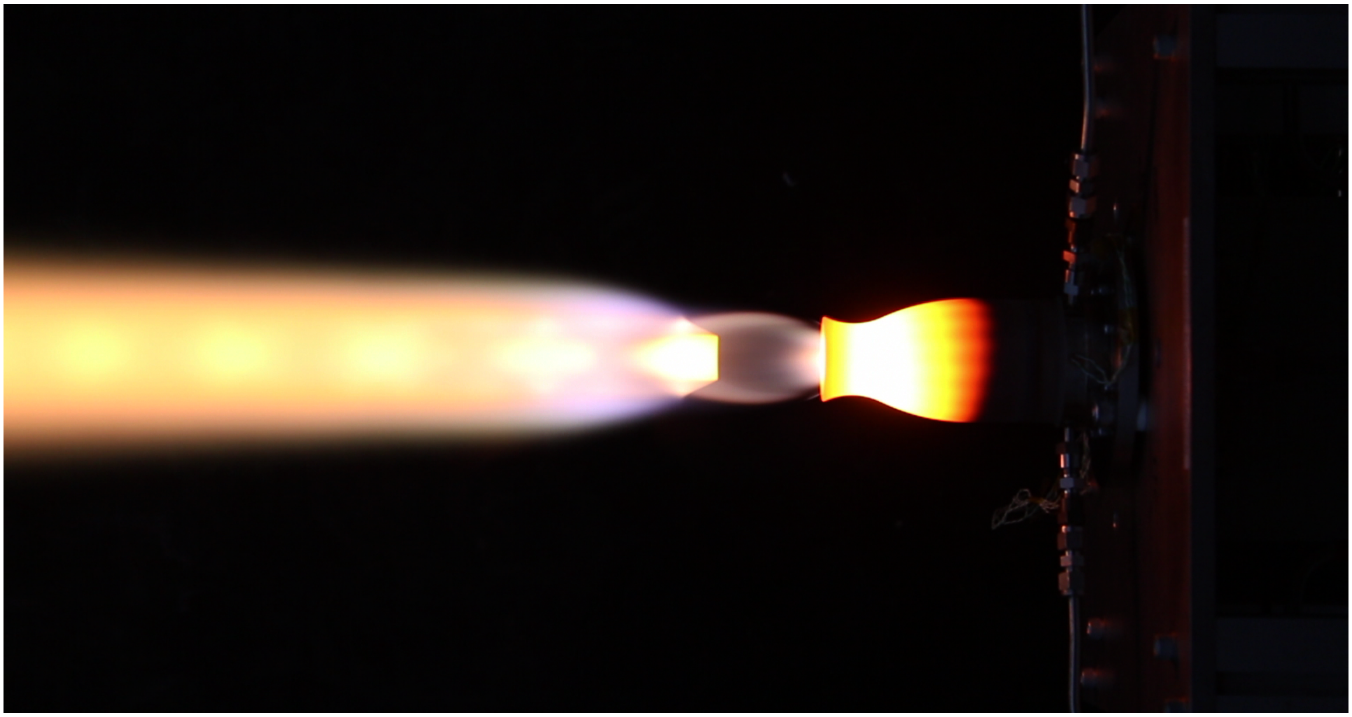

Fig. 7. An overhead optical camera view of a typical Phase 1B development model engine firing. The image shows one of the R512E/C103 workhorse chambers glowing at approximately 1400°C during testing at the Moog Westcott Operations sea-level hot-fire test facility.

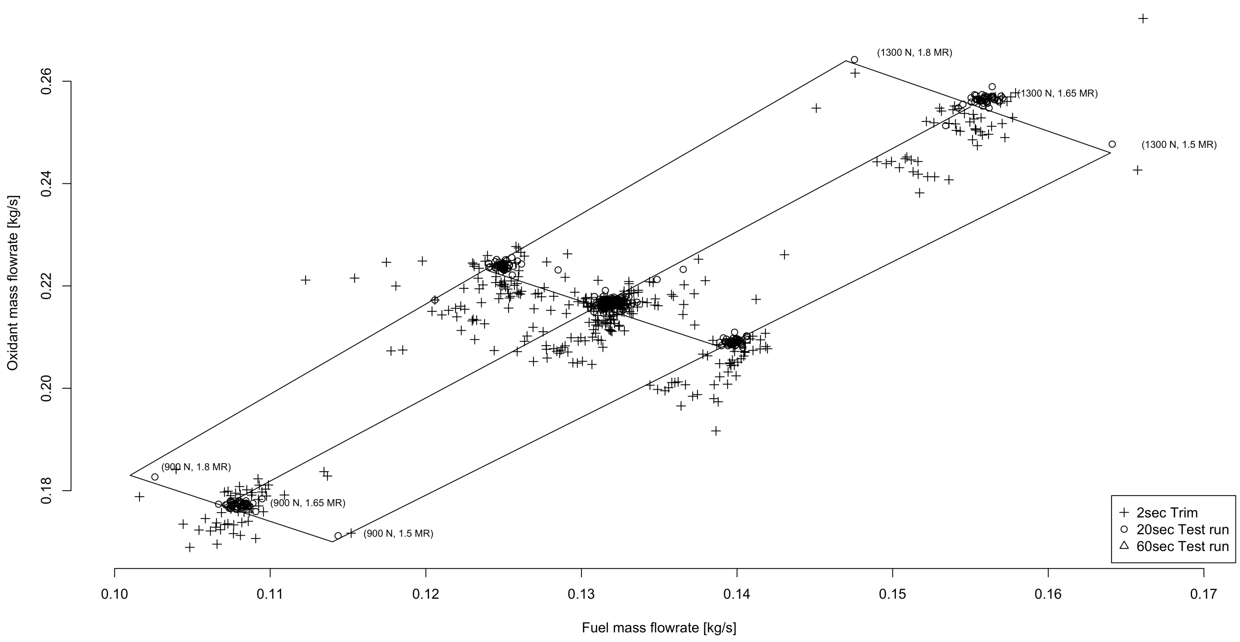

Fig. 8. The Phase 1B sea-level hot-fire testing performance box showing over 700 firings of the engine conducted over a 3 month period in 2013. Solid lines mark thrust and overall Mixture Ratio (MR) operating boundaries for the engine and plotted points show regions explored during the test campaign. The benefits of 'trimming' before a test run are clearly shown by the spread of 2 second trim markers relative to 20 second test run markers. The 60 second test run markers are obscured in the central nominal thrust and mixture ratio region.

performance box which were the focus of testing for a typical hardware configuration i.e. 900 N / 1.65 MR, 1100 N / 1.65 MR, 1300 N / 1.65 MR, 1100 N / 1.5 MR, 1100 N / 1.8 MR. To close out the test campaign, one good performing injector-chamber combination was selected for testing with the DM valves, across the entire performance box, and for longer duration at NDP.

Figure 9 shows 6 chambers that were hot-fire tested. Injectors used with these chambers were drawn from a pool

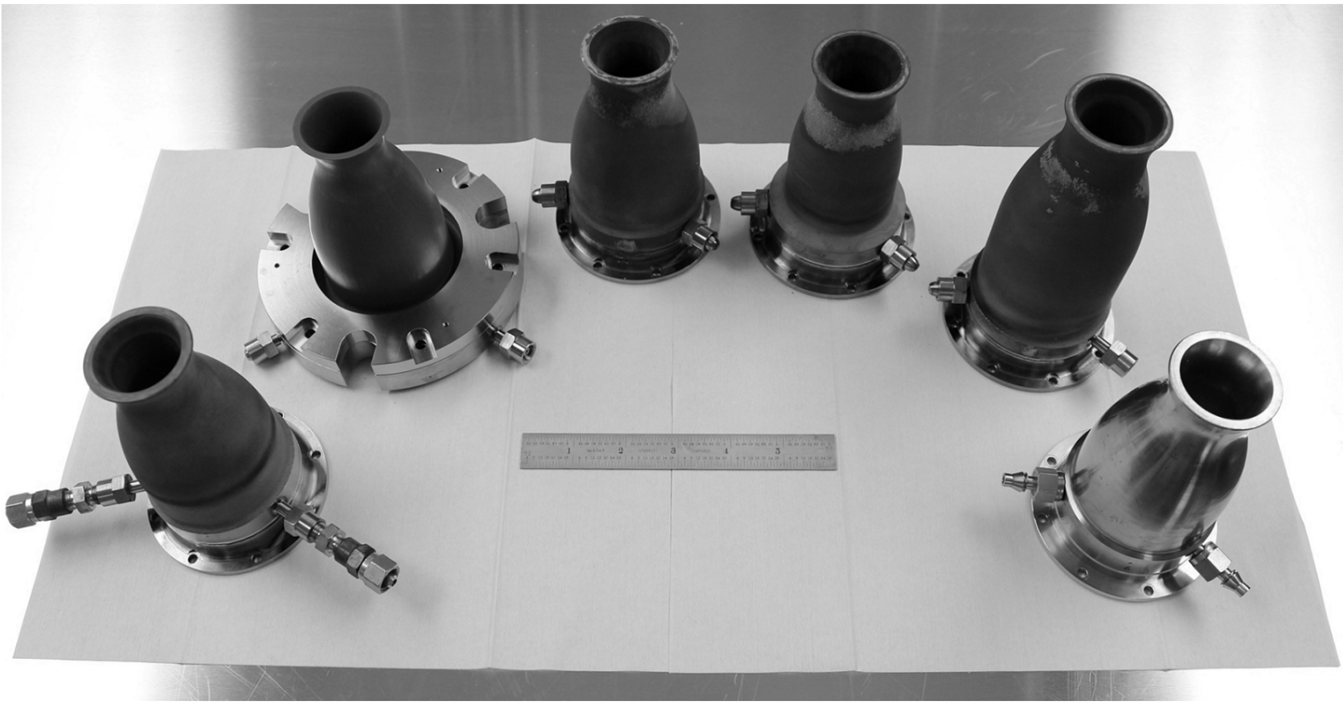

Fig. 9. Combustion chambers tested during the Phase 1B sea-level hot-fire test campaign. [Anticlockwise] Stainless steel chamber, Long R512E/C103 workhorse chamber, Narrow R512E/C103 workhorse chamber, Baseline R512E/C103 workhorse chamber, SN282 chamber with clamping assembly, Spare baseline R512E/C103 workhorse chamber. The R512E/C103 chambers show external coating damage at the throat due to thermal cycling in a higher oxygen concentration environment (sea-level) than these engines are normally operated.

of 37 different designs. In total over 50 unique configurations of valve-injector-chamber were tested in over 700 test firings.

The sensor dataset collected consisted of: thermocouple data, mass flowmeter data, pressure transducer data, infrared camera data and high definition video. In addition there was logged data for the firing command pulse, GTV opening and closing pulses. There was externally sourced data such as ambient temperature and pressure history. More qualitative data included post-firing hardware inspection images for injectors and chambers, as well as the firing site operator log which had a record of anomalies experienced for each test day and a log of maintenance activities which could have influenced testing.

There were two primary performance metrics to be considered in Phase 1B:

1) Predicted vacuum Isp (based on the C_f agreed with ESA). The target is 323 s at NDP conditions, and the minimum acceptable value is 320 s.
2) Peak-to-peak chamber pressure roughness. The requirement is a maximum value of 11 % at NDP conditions.

Based on preliminary analysis of the data, both the minimum Isp and the pressure roughness requirements have been met. The target Isp has not been met with this first iteration of the design, however it should be noted that the intent of Phase 1B was to perform a broad design parameter study rather than design for optimum performance from the outset. Given the wealth and the quality of data gathered, this development objective has certainly been achieved.

Specific impulse must be considered together with peak chamber temperature at thermal equilibrium, and preliminary analysis of test data indicates that if Phase 1B hardware geometries are used, without optimisation, to attempt to achieve this target Isp, then the use of a higher temperature capable chamber material would be required i.e. a thermal capability beyond that of the R512E/C103 material used for workhorse chambers.

In the next phase of development, the DM engine will be optimised and tested further under sea-level hot-fire testing. Major design changes will be the use of a unified fuel gallery and a propellant feed manifold designed for optimised oxidant lead. The DM valves will be exclusively used for all testing and will ensure propellant lead times into the combustion chamber are more flight representative.

The workhorse chamber design will be optimised based on Phase 1B test results, with particular emphasis on the operating pressure. Other areas in which to improve performance will include: optimisation of fuel distribution to injector elements, optimisation of injector elements, optimisation of FFC %, helium saturated propellant performance characterisation, thermally conditioned propellant performance characterisation, bubble ingestion performance characterisation, hot restart performance characterisation, and acoustic cavity design if necessary. The sea-level hot-fire test facility will be further upgraded to permit such off-nominal characterisation of the engine.

VI. Conclusion

Phase 1 involved a design study of a flight engine as well as the design, build and test of a bolt-up development model engine inclusive of development model valves.

The improvements in computing capability since the last LEROS apogee engine was qualified in 1998 has allowed for a tremendous amount of test data to be collected during this development model test campaign. Combined with the ambitious approach to testing, that had as its focus a survey of the design parameter space, this has resulted in an avalanche of data on engine performance across different propellant injector geometries, injector element manufacturing techniques, combustion chamber geometries, chamber materials and engine operating points.

The cold-flow and hot-fire test campaigns were conducted at Moog Westcott Operations in the United Kingdom. Data generated in the cold-flow test campaign was focused on injector spray imaging and flow collection across a number of different operating points in order to provide insight on hot-fire test results. Data generated in the hot-fire test campaign included infrared video, high definition optical video, thermocouple data, propellant flow rate data, chamber pressure data, as well as post-firing imaging of injectors and chambers.

Six chambers were combined with thirty seven injectors to create over fifty unique hardware configurations for testing. This represents just the first iteration of the engine hardware. In total, over seven hundred test firings were conducted on these hardware configurations, across a three month period, and at a typical rate of fourteen firings per day.

The chamber materials tested were the widely used silicide coated niobium alloy as a workhorse chamber, and a high temperature capable silicon nitride material. The injector element manufacturing techniques tested were electro-discharge machining and conventional drilling.

Phase 1 concluded in 2013 with an in depth investigation of the design drivers and sensitivities behind the baseline engine design. The geometric design of injectors and chambers in this phase has been focused on bounding the optimisation problem and the designs provided a sufficiently varying parameter range over which to collect cold-flow and hot-fire testing data. Preliminary analysis of the test data collected shows that the design and testing methodology was sufficiently capable of delivering the hardware and test data necessary to advance to the next stage of development.

In Phase 2A, which lasts approximately two years, and is formally completed with a Preliminary Design Review (PDR), the development model will be optimised and subjected to further sea-level hot-fire testing, detailed design work will be performed to define the expansion nozzle contour, and the materials for the flight engine will be finalised.

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