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Aerospike Engines for Nanosat and Small Launch Vehicles (NLV/SLV)

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The paper discusses advantages inherent to the use of an aerospike engine in the first stage of a two-stage pressure-fed Nanosat Launch Vehicle (NLV). If all other vehicle parameters are fixed, the increase in specific impulse provided by the aerospike engine allows for either using aluminum tanks instead of cryogenic composite tanks or increasing the payload capability by 23%. In order to advance aerospike research to a point where it might be baselined for such operational vehicles, the team has embarked on a flight test program to collect data over a wide range of operating conditions and flight regimes characteristic of typical launch vehicles. The first step towards this goal consisted in developing and flying a LOX/ethanol 4450 N (1,000 lbf) ablative annular aerospike engine. The first two flight tests demonstrated the basic aerospike concept and flight data collection capability. They also showed that such development, static fire tests and flight tests can take place over a relatively short period with minimal funding. Work is now underway to develop a next-generation, high-expansion ratio engine which could be used to get in-flight data to help validate aerospike performance. Although the team is pursuing this development as a means of improving the performance of the Nanosat Launch Vehicle (NLV), the research program has the potential of delivering aerospike flight data at a fraction of the cost of that of the former X-33 program, while still providing information useful for the development of future full-scale vehicles.

Nomenclature

LEO	=	Low Earth Orbit
LOX	=	Liquid oxygen
NLV	=	Nanosat Launch Vehicle (defined as 10 kg to LEO)
SLV	=	Small Launch Vehicle (defined as 100+ kg to LEO)

I. Introduction

OVER the last forty years, aerospike rocket engines have been the subject of much research. They were first investigated, primarily by Rocketdyne, in the 50's, 60's and 70's.[1,2,3,4] Because they offer advantages for missions with operations over a large range of atmospheric conditions, they were candidates for the propulsion system of the Space Shuttle but the less risky conventional bell-shape design was selected instead. Despite the decision to forgo the aerospike engine, the development work conducted for that program clearly indicated the potential for using the technology in follow-on vehicles.[5,6] It was not until the 1990's, however, that a large amount of research and development took place when NASA invested in aerospike technology for Single-Stage-To-Orbit (SSTO) Reusable Launch Vehicles (RLV) as part of the now-defunct X-33 program. This program led to the development of a prototype linear aerospike engine, the XRS-2200, which was ground-tested successfully on

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multiple occasions and demonstrated many key technologies. Unfortunately, due to the cancellation of the program, that engine never took flight and the interest in aerospike engine research declined significantly.

More recently, the California Launch Vehicle Education Initiative (CALVEIN), an academic/industry partnership between California State University, Long Beach (CSULB) and Garvey Spacecraft Corporation (GSC), developed a small liquid-propellant annular aerospike engine. This development led to the first and second known flights of such engines to date.[7] In this paper, we investigate the use of the technology for Nanosat (10 kg to LEO) and Small (100 kg to LEO) Launch Vehicles (NLV/SLV). Because the team is now focusing on the development of the former, analysis presented here will address the NLV only. Similar conclusions could be drawn for the SLV, however.

The next section presents a short background on aerospike engines. It is followed by a description of the baseline NLV concept which utilizes conventional technologies, including a classical bell nozzle. Improvements possible with the use of an aerospike engine for the first stage are discussed in the next section. Section V discusses the team's efforts towards the flight validation of the technology so that it may be transitioned to an operational vehicle. Finally, the paper ends with a summary of the more important conclusions.

II. Background on Aerospike Engines

A. Basic principles

Aerospike nozzles can be described as inverted bell nozzles where the flow expands on the outside of the nozzle instead of being completely constrained by the nozzle walls. Figure 1[‡] compares the flow at low (liftoff) and high (space) altitudes for both types. Unlike conventional bell-shaped nozzles, which operate optimally at one particular altitude, plug nozzles allow the flow expansion to self-adjust, thus improving thrust coefficients. This is particularly critical for Single-Stage-to-Orbit (SSTO) vehicles, which operate both in the atmosphere and in vacuum. This improvement over conventional bell-shaped nozzles occurs at altitudes lower than the design pressure ratio. At altitudes higher than the design altitude (or pressure ratio), plug nozzles essentially operate similarly to bell nozzles. Many references discuss these advantages[8,9] as well as typical flow characteristics on plug nozzles.[10,11] The main drawbacks associated with the aerospike nozzle are the often higher cooling requirements because the throat regions typically cover larger areas than for conventional bell shape nozzles[12] and the strong engine-vehicle interactions.

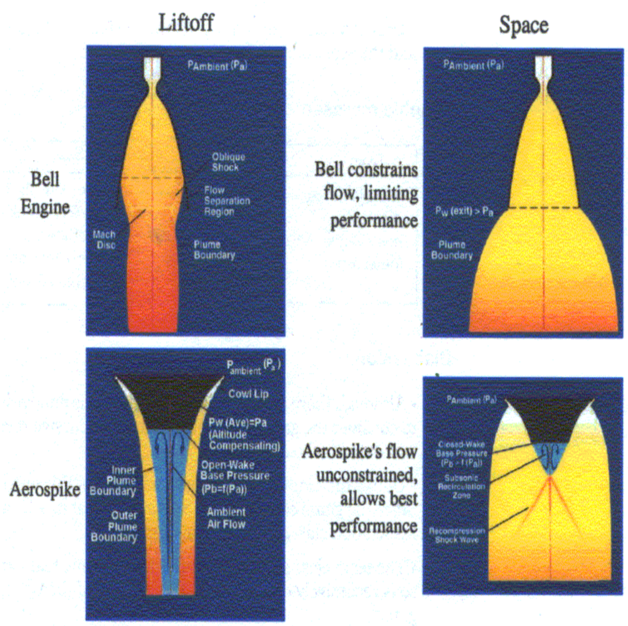

Figure 1. Bell versus Aerospike Nozzles

While the terms plug and spike nozzles are interchangeable, some authors associate aerospike nozzles with truncated spike nozzles with base bleed. In this paper, all three terms are used interchangeably.

In addition to these performance advantages over bell nozzles in atmospheric flight, plug nozzles may also offer improved packaging, reduced cost and increased reliability for space engines.[13]

For launch vehicles, both of conventional (cylindrical) type like typical expendable systems and of other shapes such as the formerly proposed Venture Star, aerospike engines typically cover the entire base of the vehicle, leading to a lighter structure for transferring the propulsion loads to the rest of the vehicle along with reduced base drag.

B. Prior developments and remaining issues

As outlined above, aerospike engines were heavily investigated particularly by Rocketdyne starting in the 50's. Several LOX/RP-1 and LOX/LH2 engines were tested, with thrusts ranging from a few thousand to a few hundred thousand pounds. Much research was also conducted using semi-analytical techniques in the early years (50's to

‡ Rocketdyne X-33 website, "http://www.boeing.com/space/rdyne/x33/aerospik/basics/advantag.htm", available 2001.

70's) (see e.g. Ref. 1) and complemented by extensive numerical simulations since the early nineteen nineties, in the United States, Europe, and Japan.[10,11,14]

Despite this extensive analytical research complemented with ground tests, there remain outstanding issues associated with the prediction of performance, both in terms of thrust and vehicle control. The primary concern deals with the prediction of the slipstream effects and their impact on vehicle performance and thrust vector control system authority and behavior, particularly in the transonic regime. As mentioned above, with the increase in computing power available, Computational Fluid Dynamics (CFD) has been used extensively to analyze these phenomena.[15,16,17] CFD, however, cannot be used with confidence without validating the models for the type of configurations being considered. This is particularly true in the engine base region where inaccuracies introduced in turbulence modeling can lead to 1-2% errors, whether base bleed is present or not. There is, therefore, a need to complement the data obtained from static fire tests and cold flow wind tunnel tests with flight test data which could be used for such CFD validation and/or calibration. The X-33 program included plans for powered flights onboard a SR-71[§] as well as several X-33 flight tests which would have addressed those issues, but the program was cancelled before any powered flight took place.

C. CALVEIN program objectives

An ongoing internally-sponsored program involving California State University, Long Beach (CSULB) and Garvey Spacecraft Corporation (GSC), intends to address these needs for obtaining flight test data. The motivation for conducting this research is directly linked to the team's on-going efforts towards the development of a Nanosat Launch Vehicle (NLV), which would be capable of delivering 10 kg to LEO.[18] Although the baseline vehicle does not employ an aerospike nozzle, aerospike technology could be used to improve the performance of the vehicle. The baseline NLV is described in the next section.

III. NLV Baseline concept

A. Sizing model description

An integrated spreadsheet sizing tool was initially developed by students as part of their Aerospace System Design project and subsequently improved (Figure 2). Each block in the diagram represents a spreadsheet that in turns contains analytical relations for that particular class of analyses. Starting with primary mission requirements, the thrust levels, engine performance and related characteristics such as specific impulse, chamber pressure and expansion ratio, structural mass fractions and stage 1/stage 2 lift-off mass ratios are defined.

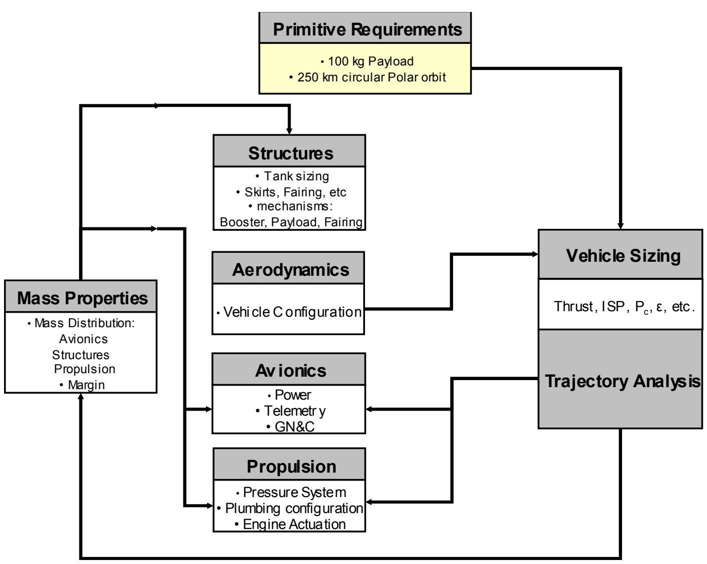

Figure 2. Integrated sizing tool

Calculated variables internal to the model include stage mass and performance data. These are in turn fed into a gravity-turn trajectory model. In order to obtain a reliable solution in terms of altitude and velocity at burnout, the trajectory simulation includes variable thrust, aerodynamic drag, gravity changes as a function of altitude and the effects of earth curvature. Because of the variation of these terms with altitude and time, it is not possible to determine a suitable set of vehicle characteristics and gravity turn initial conditions *a priori*. Instead, several runs must be made until satisfactory final conditions – in terms of burnout velocity, altitude and pitch angle – are obtained. Replacing this manual optimization process with an automated approach is one candidate area for improvement in future model versions.

[§] Linear Aerospike SR-71 Experiment (LASRE), available at http://www.dfrc.nasa.gov/Research/SR71/Lasre/index.html, 2004

Once suitable burnout conditions are achieved, the vehicle mass can be allocated to the various subsystems, the pressurization system is sized, and, given their size and operating pressures, tank masses are then estimated for the selected materials.

The tool was used to obtain a baseline vehicle capable of delivering a 10 kg payload to a 250 km polar LEO with direct orbit insertion.

B. Baseline vehicle

The baseline NLV concept is a 2-stage pressure-fed LOX-densified propylene vehicle (Figure 3).[18] The chamber pressure is nominally 2 MPa (300 psi) for the first stage and 1 MPa (150 psi) for the upper stage. The constraint that the vehicle be pressure-fed leads to structural design challenges. In order to limit the amount of gas needed to pressurize the vehicle, the cold gaseous helium is warmed, either via a heat exchanger at the engine or using small amounts of oxygen and hydrogen in the helium which reacts over a catalysts bed.[19]

This baseline vehicle was the result of trade studies conducted in order to obtain a system with as high a structural dry mass fraction as possible for each stage while keeping lift-off and payload masses constant. The result was a first stage structural mass fraction of 13.1% and a second stage structural mass fraction of 13.7%. While with these numbers an all-aluminum second stage is doable (tank pressures are around 1.25 MPa – 185 psi), the first stage (operating at higher pressures) ought to use composite materials. Other vehicle characteristics are listed in Table 1.

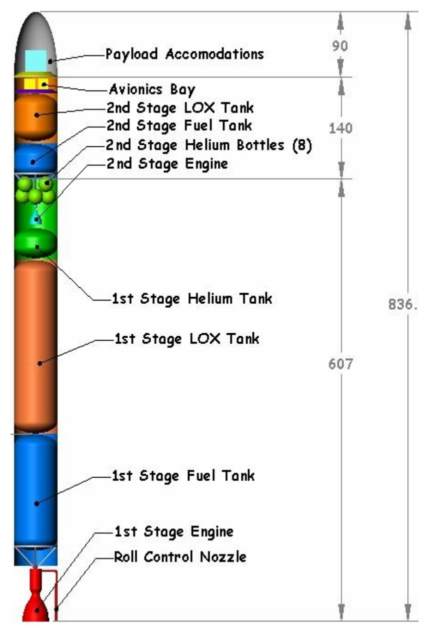

Figure 3. Baseline NLV

Table 1. Baseline NLV stage characteristics

	1st Stage	2nd Stage
Dry mass	171 kg (378 lb)	30 kg (67 lb)
Stage inert mass fraction	0.131	0.137
Chamber pressure	2 MPa (300 psi)	1 MPa (150 psi)
Sea-Level Thrust	20,000 N (4,500 lbf)	N/A
Sea-Level ISP	212 s	N/A
Vacuum Thrust	29,600 N (6,660 lbf)	1,900 N (430 lbf)
Vacuum ISP	314 s	347 s
Separation/ burnout time (from T.O.)	117 s	455 s
Separation/ burnout altitude	54 km	250 km, orbital

The key technological characteristics of the vehicle are:

- First stage composite tanks (note that in that case, it might be cost effective to make the upper stage tanks in composite as well since stage diameters are equal)
- Use of densified propylene as fuel (propylene offers slightly better performance than RP-1, is environmentally benign, is widely available, and, if densified to LOX temperature, has approximately the same density as RP-1)
- Use of hot gaseous helium
- Use of carbon/silicon carbide or silicon carbide/silicon carbide ceramic composite (C/SiC or SiC/SiC) for the upper stage engine (this option offers reduced mass when compared with refractory metals and a virtually inexistent erosion during the burn)

IV. Aerospike Engine for NLV

In order to assess the benefits of using an aerospike engine for the first stage instead of the conventional bell shape nozzle, the model was modified to simulate the near ideal thrust coefficient which can be obtained when operating at pressure ratios below the design point (after which the aerospike engine essentially behaves like a bell, being underexpanded). In order to account for the typical lower nozzle efficiency when the spike is truncated at approximately 20% of its ideal length, the nozzle efficiency was reduced from 98 to 96%. Also, the analysis was performed for an aerospike engine with an expansion ratio of approximately 40, which corresponds to an engine which can be fitted in the base of the 66 cm diameter vehicle.

A. Alternative to Use of Composite Tanks

The objective of the first analysis was to determine the possible increase in stage structural mass fraction while keeping the vehicle gross lift-off mass and size constant.

When this aerospike engine with an expansion ratio of 40 is substituted to the first stage conventional engine (which features an expansion ratio of 12), the available first stage structural mass fraction increases to 14.3%, which corresponds to having an additional 16 kg (34 lbm) available for the first stage structure (Table 2). This relaxed requirement on structural mass is due to the increase in specific impulse, ISP, during the first stage burn when compared to the baseline vehicle (Table 1): the specific impulse is larger at sea-level because the flow is not over-expanded like it is with an expansion ratio of 12 for a conventional bell; it is also larger at high altitudes because of the higher expansion ratio. With these numbers, preliminary structural analysis indicates that an aluminum vehicle may become feasible.

Table 2. NLV with aerospike first stage engine

	1st Stage	2nd Stage
Dry mass	187 kg (412 lb)	30 kg (67 lb)
Stage inert mass fraction	0.143	0.137
Chamber pressure	2 MPa (300 psi)	1 MPa (150 psi)
Sea-Level Thrust	20,000 N (4,500 lbf)	N/A
Sea-Level ISP	243 s	N/A
Vacuum Thrust	29,600 N (6,660 lbf)	1,900 N (430 lbf)
Vacuum ISP	347 s	347 s
Separation/ burnout time (from T.O.)	117 s	455 s
Separation/ burnout altitude	54 km	250 km, orbital

B. Increased Payload Capability

Alternatively, if an all-composite vehicle can be built (i.e. with composite cryogenic tanks for LOX and densified propylene), the use of an aerospike engine first stage translates into increased payload capability. To assess the improvement in payload capability, the stage dry mass fractions of the baseline NLV were kept constant and the payload was increased until the target orbit could no longer be reached. ***The payload capability grew by 23%, reaching 12.3 kg.*** This increase is due to the higher impulse provided by the aerospike first stage (2595 m/s) when compared with the conventional first stage (2486 m/s), thus reducing the requirement in second stage impulse. This prediction in payload increase is slightly higher than the 15% improvement obtained with the use of an aerospike engine for the Ariane 5 launcher[9], probably because of the different vehicle configurations.

V. Flight Test Development Program

A. Objectives

The results presented above clearly indicate that, while not required for the baseline vehicle, the aerospike engine concept can provide performance improvements to the proposed NLV. Similar conclusions could be reached for an SLV.

The CALVEIN team has, therefore, initiated on a program to further the research on aerospike nozzles, focusing in particular on flight tests for addressing issues discussed in Sect. II.B and which could benefit NLV/SLV as well as large scale launch vehicles. True to the team's philosophy based on an incremental improvement approach, the first objective was to develop a first version of a liquid-propellant aerospike engine and demonstrate flight capability before embarking on a more ambitious flight test program.

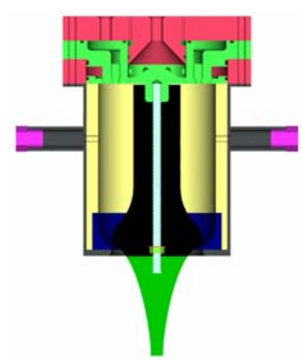

Figure 4. Modified aerospike engine

B. First flights of liquid propellant aerospike engines

Ref. 7 discusses in detail the development of the first family of engines leading to the first two known –and to date only– flights of liquid-propellant aerospike engines. Only a short summary of these developments is presented here.

The program was initiated in the Fall 2001 semester as part of the Aerospace System Design class at CSULB. Senior students developed a LOX/ethanol 1,000 lbf ablative annular aerospike engine. The engine was tested in April 2002 and, after operating for 200 ms, the central plug failed. With faculty and industry mentor participation, a new team of students modified the design. The primary modification was the addition of a titanium rod (shown in light blue on Figure 4) in the center plug to transfer the loads to the injector. A 4-second static fire test was successfully conducted in June 2003 (Figure 5) and the engine was integrated to the previously-flown Prospector-2 vehicle for a September 2003 flight (Figure 6).

Figure 5. 1000 lbf thrust engine static fire test (06/03)

Figure 6. First flight of a liquid-propellant aerospike engine (09/03). *Photo by Tony Richards.*

After a smooth countdown and nominal engine ignition, the thirteen-foot long P-2 quickly accelerated up a 60-ft launch rail and entered stable flight. Soon afterwards, the thrust decreased and, then an off-axis component arose which ultimately caused the vehicle to enter unstable flight. Post flight engine inspection showed that a significant amount of flow escaped around the outer surface of the graphite convergence ring and burnt the back of the chamber, leading to several secondary plumes. Benign at the beginning of the flight, these secondary plumes eventually led to asymmetric thrust and sent the vehicle out of control until it transitioned into a unpowered ballistic terminal descent. It is worth noting that the subsequent crash was caused by a separate and independent malfunction in the new non-pyrotechnic recovery that was flown for this time on this flight.

A majority of the parts of the damaged P-2 vehicle were still usable and these were subsequently integrated into a new vehicle, the Prospector-4 (P-4). In order to avoid the engine burn anomalies observed in the first flight, the tolerances on the outer ring interface with the aft engine flange were tightened so that the ring could not shift position during the burn. Less than three months after the maiden aerospike flight, the Prospector-4 vehicle was ready to fly with the refurbished engine. The 12-ft P-4 lifted off on Dec. 6, 2004, propelled by the 4450 N (1,000 lbf) LOX/ethanol aerospike engine, flying flawlessly (Figure 7) to an altitude of almost 1370 m (4,500 ft) above the launch site. The parachute deployed at apogee and the vehicle floated back to the ground for recovery (Figure 8). An onboard flight computer was used to collect chamber pressure, altitude (via barometric pressure) and three axis accelerations. Figure 9 shows vehicle longitudinal acceleration, chamber pressure and measured altitude. Due to uncertainties in propellant loading, accurate thrust measurements during the burn were not possible. The problem is further compounded by the fact that, in order to keep the costs to a minimum, no pressurization system was flown and the vehicle used ullage pressure to feed the propellants. This effect is visible in the chamber pressure data which reach maximum value after half a second and then start decreasing, leading to a low frequency combustion instability ("chugging") around 3 seconds after lift-off. The burn ends after slightly more than 8 seconds. At the end of the burn, a payload developed by students from Cerritos High School, CA, was ejected. Unfortunately, their home-made parachute tore upon deployment and their camera system fell from 2,000 ft. This deployment is visible in the accelerometer and altimeter data around 7.5 s. After this event, the higher pressure air in the vehicle is vented and the barometer located near the payload bay senses the correct altitude. Figure 10 shows the trajectory of the vehicle including a portion of the parachute descent.

Figure 7. Nominal Aerospike Engine Burn During the P-4 Flight, Dec. 03

Figure 8. P-4 After Recovery

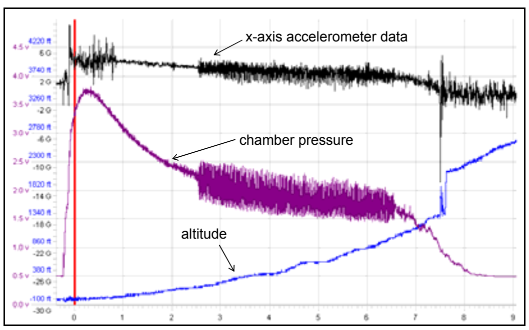

Figure 9. Data from P-4 Aerospike Flight

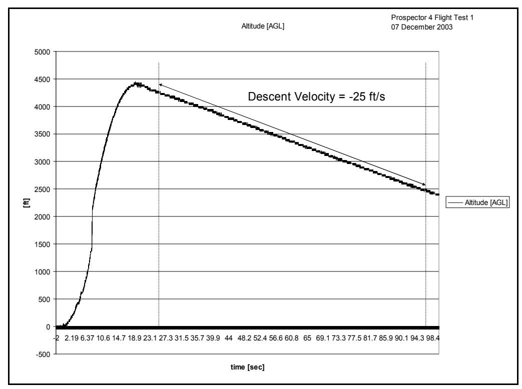

Figure 10. Flight Profile of the P-4 Vehicle

C. Multi-thruster concept development and plans

The aerospike engine research documented in Ref. 7 and summarized in the previous section corresponds to a significant yet only first step in the development of aerospike technologies for future launch vehicles. In order to address the issues raised in Sect. II.B, this demonstration program must be expanded to a larger area ratio engine flying over a larger range of flight conditions and with a higher fidelity data collection system. The team has now embarked on the next steps towards accomplishing this goal.

The first objective of the present CALVEIN aerospike research program is to increase the expansion ratio of the engine from 5 for the P-2 and P-4 engines to approximately 30 in order to validate the altitude compensation capabilities of the aerospike technology over various flight regimes. In the previous engine design, the expansion ratio was purposefully kept low in order to have a producible annular throat gap (about 4.5 mm). For larger expansion ratios, the single annular combustion chamber must be replaced by a series of small thrusters as is done on most large-scale aerospike engine designs.

The concept being considered here is similar to that of a General Dynamics design of the early nineties.[20] The engine is designed to power an upgraded Prospector-type vehicle and is, therefore, designed to generate 4450 N (1,000 lbf) with ten 445 N (100 lbf) thrusters arranged around an annular plug with an area ratio of approximately 30 (Figure 11). Although the nozzles depicted in the figure are conical two-dimensional nozzles, it is anticipated that the final design featured a more tailored contour that reflects the results of CFD-optimization. Each thruster is designed to operate at 2 MPa (300 psi). Thus, an expansion ratio of approximately 5 is achievable for each thruster. When combined, the resulting total expansion ratio is 30.

Several studies have shown that the nozzle of the thrusters must be designed carefully in order not to lose the advantages gained from using a plug nozzle.[17,21,22] In particular, adjacent circular thruster exits or gaps between thrusters give rise to complex three-dimensional flow fields with shock and viscous losses. In the present design, a “rectangular” thruster exit contoured on the plug nozzle is therefore baselined. This approach makes the thruster nozzle design more challenging than for a linear aerospike engine. For the latter, Booth *et al.*[23] investigated the flow in various throat-nozzle configurations. Their analysis showed that, while the flow behavior is important, other factors such as heat transfer and manufacturing costs are also elements to consider in the design.

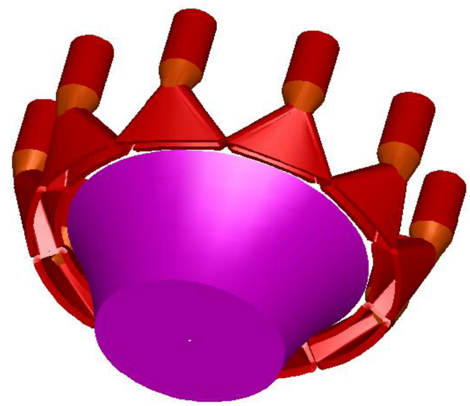

Figure 11. Multi-thruster 4450 N (1000 lbf) Aerospike Concept (area ratio of 30)

The CALVEIN team is currently evaluating high temperature ceramic composites, such as SiC/SiC, which are leading candidate materials for the aerospike thrusters. In this case, the manufacturing processes allow basically for any arbitrary thruster/nozzle shape to be made. In parallel, a project employing CFD-optimization techniques for thruster/nozzle shape design is being planned. Another project, which has already started, addresses the development of the injector for the thruster. Finally, this engine concept may also be used for investigating thruster throttling for thrust vector control (TVC) as was done on the XRS-2200. It is expected, however, that, due to the axisymmetric arrangement, the TVC authority be less than that of linear aerospike engines.

Overall, the proposed concept addresses multi-thruster flowfield, heat transfer and manufacturing process issues. The engine could then be integrated into an upgraded Prospector-class vehicle (e.g. with the addition of larger tanks and pressurization system) for instrumented flight tests. The pace at which the team is able to achieve this goal is dependent on the level of external funding that becomes available.

VI. Conclusion

The paper shows that the use of an aerospike engine may lead to a reduction in risks associated with cryogenic composite tank development by allowing an all-aluminum pressure-fed NLV. Alternatively, it can translate into significant improvements in performance, increasing the payload to LEO by 23%.

The first steps towards operational use of the aerospike concept have been taken by developing and demonstrating flight capability of a first family of 1,000 lbf thrust annular aerospike engines. The next steps involve the development and flight tests of a family of higher expansion ratio engines on an upgraded Prospector-class vehicle in order to expand the data collection over a larger range of flow regimes and operational conditions. Such data will be critical to the validation of the aerospike engine technology and its transition from research to operations on NLV/SLV, as well as larger launch vehicles.

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