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Preliminary Performance Data of an 18kN Ablatively Cooled, Blowdown Liquid Rocket Engine

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The South African First Rocket Engine (SAFFIRE) program by the University of KwaZulu-Natal's (UKZN) Aerospace Systems Research Institute (ASRI) aims to develop a flight-weight liquid rocket engine (LRE) to be used on an indigenous South African launch vehicle. The first step towards a flight-weight engine design was developing and testing the Ablative Blowdown Liquid Engine (ABLE), a prototype LOX/Jet A-1 engine, designed to operate at 12 bar and producing 17.6 kN of thrust. This paper provides an overview of the SAFFIRE ABLE ground tests conducted at South Africa's Denel Overberg Test Range (OTR) and includes preliminary data from ignition, engine start-up and steady hot-fire tests.

I. Engine and Facility Overview

To our knowledge, the SAFFIRE ABLE engine is the first successfully tested liquid rocket engine developed in Africa and, at the time of testing, became the second most powerful LRE developed by a university, producing a mean thrust of 18.2 kN. The propellants were liquid oxygen (LOX) and Jet A-1 aviation kerosene at mass flow rates of 5.66 kg/s and 2.27 kg/s, respectively. A 12 bar design chamber pressure was selected to enable blowdown testing of the engine by reducing the required pressure rating of the propellant supply tanks.

A liquid rocket propulsion test facility was developed to test and validate the engine's performance and is described in a partner paper by Cooper et al [1]. The containerized mobile liquid rocket engine test facility was designed to be modular and transportable.

The Jet A-1 feed system used a cavitating venturi downstream of the supply tank which enabled control of the flow rate of the blowdown system and mitigated propellant overflow on engine start-up. The LOX feed system did not have sufficient pressure head to allow for a cavitating venturi on the main line and therefore a bypass line with a reduced flow rate was used incorporating a cavitating venturi to facilitate engine start-up.

II. Ignition and Start-Up Testing

The combustion chamber was designed to fail safely in the event of an overpressure by using frangible bolts on the nozzle section. These were designed such that the nozzle would separate from the combustion chamber at 18 bar, or 50% overpressure, directing any blast downrange and away from the test facility. An overpressure event could be caused by above-nominal propellant mass flow, combustion instability, or a hard start. A hard start is caused by a delayed ignition, where propellants accumulate in the combustion chamber and then detonate [2].

The bolted nozzle section allows for ignition testing with the nozzle removed from the combustion chamber. This testing configuration, as shown in Fig. 1, referred to as 'nozzle-off', eliminates the risk of an overpressure event as there is no converging section to sonically choke gas flow. The purpose of the nozzle-off tests was to determine if the custom-developed pyrotechnic ignition system was sufficiently energetic to vaporize and ignite the propellants without risk of quenching or delayed ignition. These tests also validated the ignition, valve timings and start-up procedures for nozzle-on hot-fire tests.

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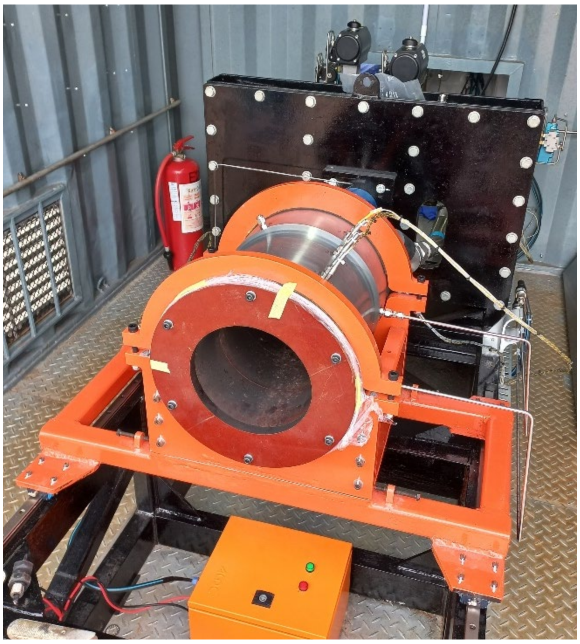

Fig. 1 ABLE in the nozzle-off testing configuration.

The pressurant supply was set to 29.5 bar and 19.7 bar for the fuel and oxidizer run tanks, respectively. Dome-loaded regulators were used to manage the pressurant supply, however, no means were available to adjust the supply pressure during the test and the supply pressures were therefore set before testing. On engine start-up, the injector would be subject to a pressure differential (ΔP) of approximately 15.5 bar in the LOX manifold, as no significant back pressure would arise with the engine operating without a nozzle. The nominal ΔP across the faceplate is 3.5 bar during engine operation. The cavitating venturi in the Jet A-1 line decoupled the flow rate from the downstream pressure such that the flow rate was independent of chamber pressure. This resulted in a constant flow rate and constant pressure drop of 2.5 bar through the injector fuel manifold during cold-flow testing, engine start-up, and engine operation.

The LOX bypass line served multiple purposes, one of which was to restrict the flow with a cavitating venturi so that the injector LOX manifold did not overpressure on start-up. The second function of the bypass line was to supply a reduced oxidizer flow rate and therefore reduced chamber pressure on start-up. This approach is referred to as levelled combustion and is the same approach used in the development and start-up of the Fastrac engine [3], for example. The operating specifications of Level 1 and Level 2 combustion are detailed in Table 1. The levelled combustion approach was taken to further reduce the risk of overpressure on start-up, as a nozzle separation due either to an ignition delay or defect in the frangible bolts would have prematurely halted the test campaign.

Tab. 1 SAFFIRE ABLE levelled combustion specifications.

Parameter	Units	Level 1	Level 2
LOX flow rate	kg/s	3.0	5.66
Fuel flow rate	kg/s	2.27	2.27
O/F ratio	-	1.32	2.49
Chamber pressure	bar	5.7	12
LOX bypass line	-	Yes	No

A pyrotechnic ignition system was used to ignite the propellants entering the engine. The igniters were inserted into the combustion chamber, through the nozzle, prior to testing. This approach allowed for multiple hot-fire tests without having to disassemble the engine to reinstall and rearm the igniters. The ignition ring and igniter cartridges were made of soft, combustible materials such as polylactic acid (PLA) so that they did not damage the nozzle when evacuating the chamber at high speed. Annularly-positioned potassium nitrate-based grains formed the pyrotechnic component of the ignition system. These were configured such that the flame projected radially inwards near the injector face and were initiated with e-matches.

III. Hot-Fire Testing

The first hot-fire tests conducted during the 2021 ABLE test campaign were nozzle-off ignition tests, such as the one shown in Fig. 2. The outcome was to verify that the ignition system, valve timings and start-up and shutdown procedures all performed nominally, with a reduced risk of catastrophic failure due to no pressure build-up in the chamber. The ignition system successfully ignited the propellants with sufficient energy to sustain combustion, paving the way for the start-up testing of the engine in the nozzle-on configuration.

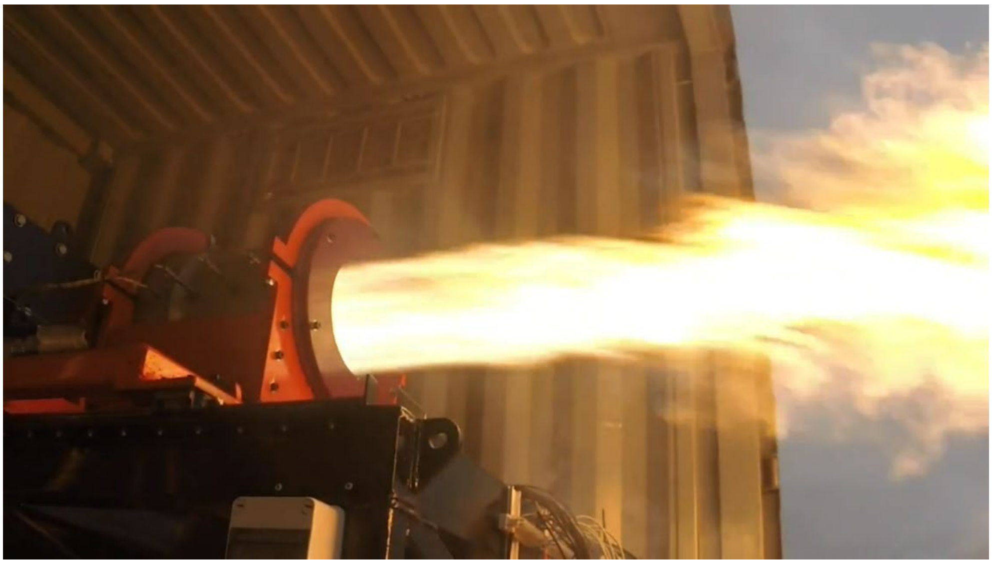

Fig. 2 Nozzle-off ignition test.

Following successful ignition tests, nozzle-on, Level 1 combustion tests were performed. These hot-fire tests utilized the LOX bypass line exclusively, resulting in a reduced LOX flow rate, chamber pressure and O/F ratio. The outcome was to validate the ignition and start-up process with the presence of backpressure in the chamber and to determine the timing for the ramp-up to Level 2 combustion. Fig. 3 shows the associated chamber and injector pressure histories for a 10 s Level 1 combustion test. The decrease in injector pressure with time is attributed to the LOX feed system cooling down, resulting in less multiphase flow and a higher propellant density. This resulted in an increase in chamber pressure with time, as both the LOX density and mass flow rate increased. Fig. 4 shows the nozzle-on Level 1 combustion test operating with a dark, fuel-rich plume.

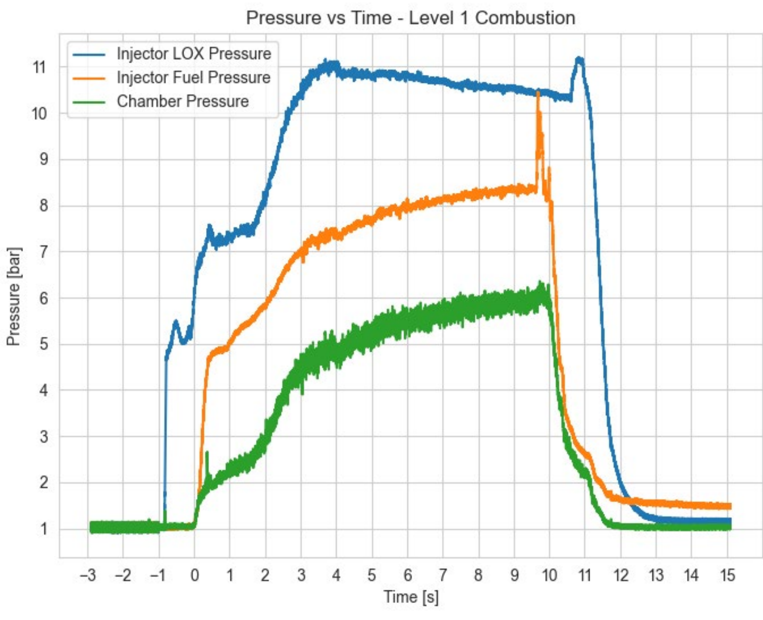

Fig. 3 Injector and chamber pressures histories for the 10 s level 1 combustion test.

At approximately 4 s into Level 1 combustion, the engine had reached a state where the start-up transients had settled, and changes in pressure were as a result of the cooling and temperature changes of the LOX and LOX feed system. It was determined that 6 s into level 1 combustion, the flow was sufficiently stable to ramp-up to Level 2 combustion by permitting the LOX flow through the primary feed line instead of the bypass line.

Fig. 4 Nozzle-on Level 1 combustion.

Nozzle-on Level 2 combustion tests were conducted to determine engine performance characteristics at nominal operating conditions. Fig. 5 shows the injector and chamber pressures histories for the 16 s Level 2 combustion test. The levelled combustion approach ensured no occurrence of overpressure in the chamber, as no pressure spike above nominal pressures was observed. The increase in chamber pressure with time is again attributed to the chilling down of the LOX and LOX feed system throughout the test. The maximum operating chamber pressure was 12.4 bar, sampled over 1 s periods.

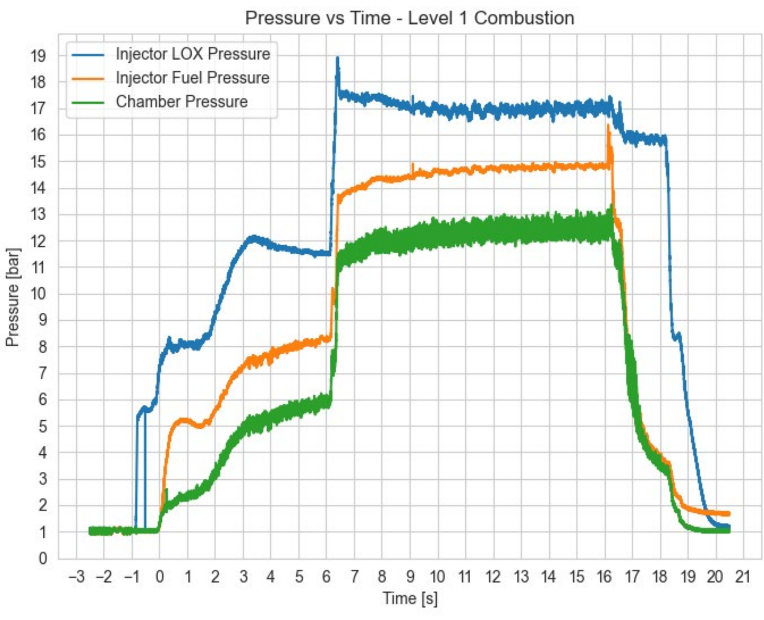

Fig. 5 Injector and chamber pressures for the 16 s Level 2 combustion test.

The thrust history for the 16 s Level 2 combustion test is shown in Fig. 6. A peak thrust of 19.4 kN was measured at the end of the test with an average thrust of 18.2 kN sampled over the last second before engine shutdown. The chamber pressure and thrust were higher than the nominal design points of 12 bar and 17.7 kN, respectively. This is due to the increased flow rate of LOX at 6.28 kg/s instead of 5.66 kg/s, with fuel operating close to nominal at 2.26 kg/s. An image of the ABLE engine operating under nominal conditions is shown in Fig. 7.

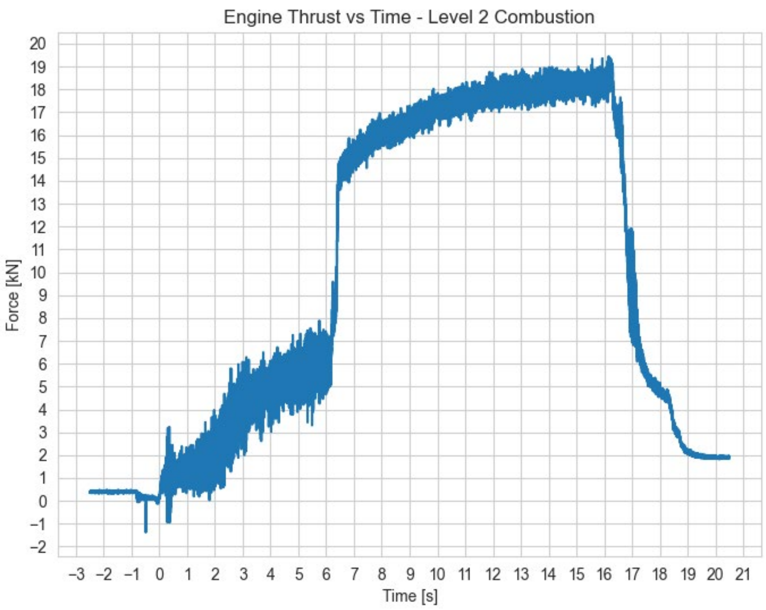

Fig. 6 Thrust curve for 16 s Level 2 combustion test.

Fig. 7 Nozzle-on level 2 combustion.

IV. Conclusion

Multiple hot-fire tests of the SAFFIRE ABLE engine were conducted. The ignition tests validated the ignition system and start-up procedures. The levelled combustion approach prevented overpressures in the combustion chamber on engine start-up and no nozzle separation or failure occurred. Data from Level 1 combustion tests were used to inform the ramp-up timing to Level 2 combustion at 6 s. Due to the increased LOX flow rate, the chamber pressure and thrust from the 16 s Level 2 combustion test were above nominal at 12.4 bar and 18.2 kN, respectively, compared to the design point values of 12 bar and 17.6 kN. The lack of a cavitating venturi on the primary LOX feedline resulted in less control over the propellant flow rate compared to the LOX bypass and fuel feed lines. At the time of testing, the ABLE engine became one of the most powerful liquid rocket engines developed by a university. The design of the ablatively cooled ABLE engine chamber is described in a separate paper by Wunderlin et al [4].

Acknowledgments

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