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Design and development of an 8KN liquid rocket engine system

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Abstract

This paper details the indigenous design, fabrication, and static hot-fire testing of an 8kN class liquid rocket engine using kerosene and gaseous oxygen at the Bola Ahmeed Tinubu Centre for Space Transport and Propulsion, Epe, Lagos. The project aimed to build local capacity and advance self-reliance in launch vehicle technology by validating critical systems, including injector design, propellant feed, cooling, and combustion stability.

Across two static hot-fire tests, the engine demonstrated structural integrity and confirmed system workability. Although neither test achieved the 8kN thrust target, Test 2 showed improved performance over Test 1, reaching a maximum thrust of 6.281 kN and a specific impulse of 137.81 seconds due to enhanced combustion efficiency and propellant mixing. Performance shortfalls were attributed to limitations in chamber pressure, injector atomization, and nozzle design. The collected data provides an experimental baseline and key insights for optimizing future high-thrust engine designs in Nigeria.

Keywords: Liquid rocket engine; Kerosene; Gaseous oxygen; Static hot-fire test; Thrust; Specific impulse; Indigenous propulsion.

1. Introduction

The capacity for space launch serves as a benchmark for technological sovereignty and acts as a crucial enabler for state development across communication, navigation, earth observation, and scientific sectors [1]. At the heart of any launch system lies the rocket engine, with liquid rocket engines LREs being the preferred propulsion choice for orbital launch vehicles due to their high specific impulse, throttability, and restart capability [2]. A liquid rocket engine consists of a thrust chamber and a feed system (either pressure-fed or turbopump-fed) that delivers propellants at high pressure. Turbopump systems rely on configurations like gas generator, staged combustion, or expander cycles. These cycles are classified as "open" if the turbine exhaust is dumped or routed separately from the main thrust chamber, or "closed" if it is entirely recycled [3]. [4] also reported that liquid propellants is one of the most widely used rocket propulsion systems. [5] also worked on the development and testing of a liquid rocket engine which was part of a multi-stage mixed vehicle of solid and liquid propulsion, manufactured and assembled with a very limited budget and using local methods, resources and industries available in Colombia. He reported a a breakthrough in the area with regard to South American countries and emerging space powers, which up to date have done work on solid fuel rockets mainly. [6] also reported that the engine system requirements for different engine cycles significantly influence the design of the mixing head

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and also outlined the influence of the engine cycle and systematic design on the fuel injector design and the physics. [7] designed and tested a 3 kN pressure-fed liquid rocket engine utilizing liquid methane and liquid oxygen to gain foundational knowledge for large-scale engine development. This paper details the engine's design, its coaxial gas-liquid injection system, its pyrotechnic ignition, and outlines four upcoming propulsion research tracks at Delft University of Technology. [8] successfully demonstrated the feasibility of throttling LRE using HTP as an oxidizer operating at SL conditions. A throttling range of 10% – 100 % relative to the nominal operating point (design point) was demonstrated.

Globally, access to liquid propulsion technology is tightly controlled. The design, materials, manufacturing processes, and test infrastructure involved are both capital and knowledge intensive [9]. As a result, only a few nations possess the indigenous capability to develop and qualify LREs. For emerging space nations, developing this capacity domestically is therefore a strategic necessity to reduce dependence on foreign suppliers, build local expertise, and create a sustainable foundation for future launch vehicle programs. In recognition of this need, an indigenous effort towards building capacity in liquid rocket engine technology was initiated [10]. The objective of the program was to develop end-to-end competence in LRE design, component manufacturing, integration, ground testing, and performance evaluation using local resources and expertise to the maximum extent possible [11]. An 8kN thrust class Liquid Rocket Engine was designed, fabricated, and assembled. The 8kN thrust level was selected as an appropriate starting point because it is large enough to validate key LRE technologies such as propellant feed systems, combustion chamber cooling, ignition, and thrust chamber performance, while remaining manageable in terms of cost, safety, and test facility requirements. The engine was developed around a bipropellant architecture and incorporated critical subsystems including the thrust chamber assembly, injector, propellant valves, feed lines, and ignition system [12].

Following the manufacturing and integration, the engine underwent a series of acceptance and hot-fire tests to verify structural integrity, combustion stability, and overall performance against design targets. The successful design, construction, and testing of the 8kN LRE represent a significant step in establishing indigenous liquid propulsion capability. It demonstrates the feasibility of local design and manufacturing, provides data for model validation, and builds the human and infrastructural base required for scaling to higher thrust engines in future launch vehicles. This paper presents an overview of the indigenous development efforts. It details the design philosophy, manufacturing approach, test set-up, and results obtained from the 8kN LRE static hot-fire test campaign. The lessons learned and the way forward for advancing national liquid propulsion capability are also discussed.

2. Materials and methods

2.1. Mission Setting

The design and development of an 8kN liquid rocket engine started with the mission goal and objectives. Part of the mission set was to develop an LRE system that can attain ten kilometres(10km) above sea level (Table 1 and Figure 1).

Table 1 Trajectory Analysis for 10 kN Rocket Launch Capacity

Propulsion Requirements	Values
Burn Time (s)	7
Specific Impulse (s)	251
Rocket Mass (kg)	103.69
Propellant Mass Ratio	0.23
Mass of Propellant (kg)	23.68
Propellant Mass Flow Rate (kg/s)	0.82
Mass of Final Rocket (kg)	80
Rocket Diameter (m)	0.3
Altitude at Burn Time (m)	5300
Maximum Altitude(m)	10000
Time at Maximum Height(s)	44.2

Total Impulse (N-s)	58320.67
Average Thrusts (N)	8331.52
Thrust to Weight Ratio	8.19

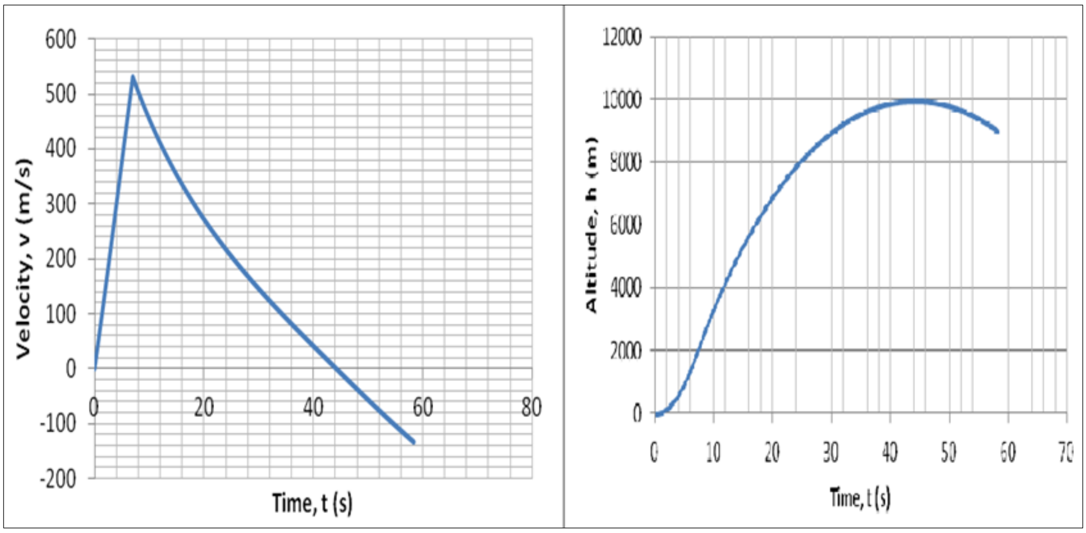

Figure 1 Trajectory analysis of rocket altitude and velocity

2.2. Engine Designs and Specifications

The propellant combination selected for the design are Kerosene (DODECANE)($C_{12}H_{26}$) as the fuel while the oxidizer is gaseous oxygen(O_2), with the combustion equation as shown in Equation 1. The propellant characterization is shown in Table 2. The propellant budget is also display in Table 3 and the engine specification in Table 4.

$$C_{12}H_{26}+12.5O_2 \rightarrow 12CO+13H_2O \quad (1)$$

Table 2 Propellant Characterization

Characteristics	**Values**
Propellant Density (kg/m^3)	0.4
Propellant Temperature (K)	3503
Chamber pressure (MPa)	0.65
Heat Ratio	1.22
Molecular Weight (gmol)	22.1
Propellant Gas Constant (J/kgK)	376.03
Specific Impulse (s)	251
Characteristic Velocity (m/s)	1759

Table 3 Propellants Budget

Propellant Parameters	**Values**
Mass of Propellant (kg)	23.69
Amount of Gaseous O_2 (kg)	16.62
Volume of Gaseous O_2 (m^3)	0.02
Mass of Gaseous O_2 Tank (kg)	5.67
Amount of Kerosene (kg)	7.07
Volume of Kerosene (m^3)	0.01
Mass of Kerosene Tank (kg)	0.21

Table 4 Thrust Chamber Design Specifications

Componets	**Parameters**	**Values**
Nozzle	Throat diameter (m)	0.11
	Nozzle exit diameter (m)	0.2
	Nozzle divergent length(m)	0.18
Combustion Chamber	Chamber diameter(m)	0.21
	Chamber length (m)	0.29
	Chamber wall thickness(m)	0.01

2.3. Fabrication of Components

Figure 2. shows the computer aided design of the designed 8kN liquid rocket engine, from the parts drawing, and assembly drawing. Also, Figure 3 shows the computer aided drawing of the appropriate test stand meant for the engine. Figure 4 shows the drawings for the tanks. The fabrication process of the engines, which was carried out at the engineering workshop, Bola Ahmeed Tinubu Centre for Space Transport and Propulsion, Epe, Lagos is shown in the Figure 5-6. This was done using locally sourced materials. A blow down pressure-fed system was selected to reduce system complexity and cost for this technology demonstration. The kerosine-gasoline propellant combinations were selected based on easy of storability, density, and compatibility with indigenous materials. Water was used regeneratively for cooling of the engine to manage wall heat flux. Tank pressure of 3.6 MPa was selected to provide 0.6 MPa margin above Pc. Solenoid/pyro valves were used for propellant isolation. Check valves and burst discs were incorporated for safety.

The thrust chamber and nozzle were fabricated from the laboratory at the Bola Ahmeed Tinubu Centre for Space Transport and Propulsion, Epe, Lagos. Structural analysis was performed to verify chamber wall thickness against internal pressure and thermal loads with a safety factor of 1.5. All welds were designed per ASME standards and subjected to NDT. The thrust produced was measured from the installed 30kN (material: Stainless steel, capacity: 30kN, Protection: Usually IP65 or IP67, Mounting: Bolted through holes, force applied in the center and Output: 2.0 mV/V standard) on which the chamber is resting.

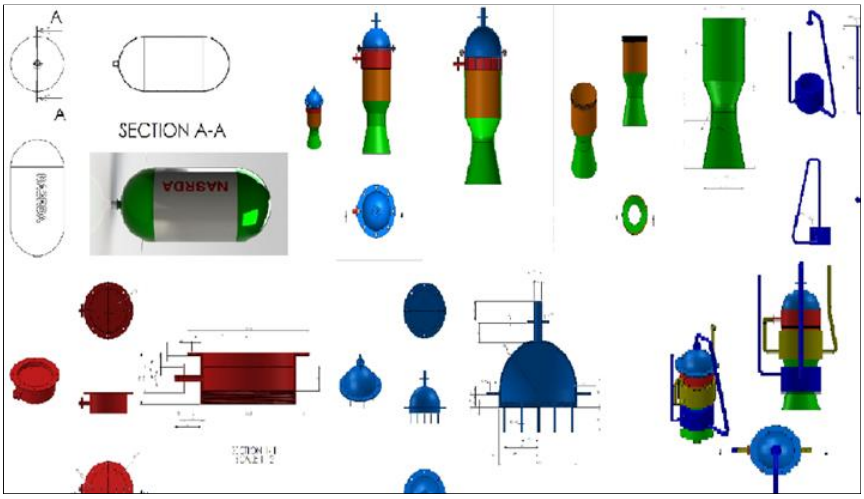

Figure 2 CAD of the liquid rocket engine component parts

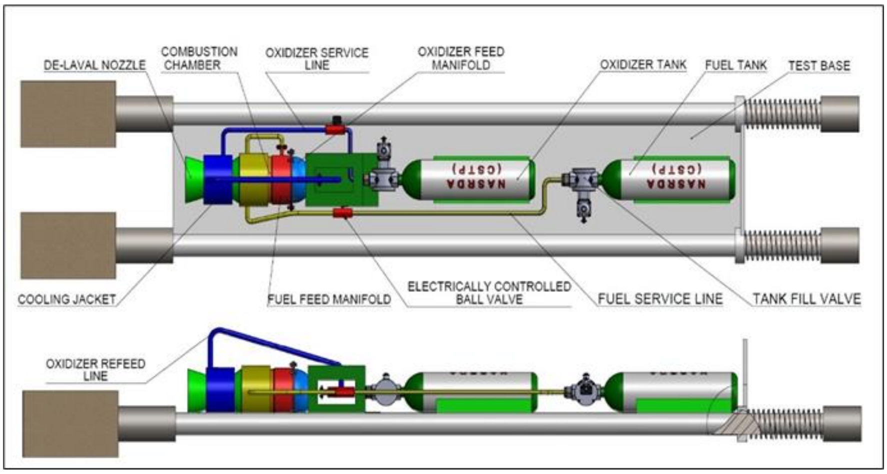

Figure 3 CAD of the Liquid rocket engine assembled on test stand

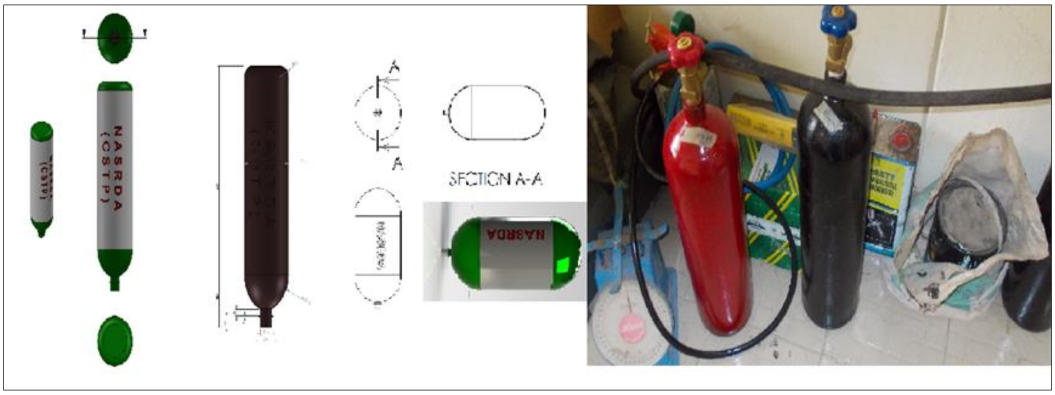

Figure 4 Liquid rocket engine tanks

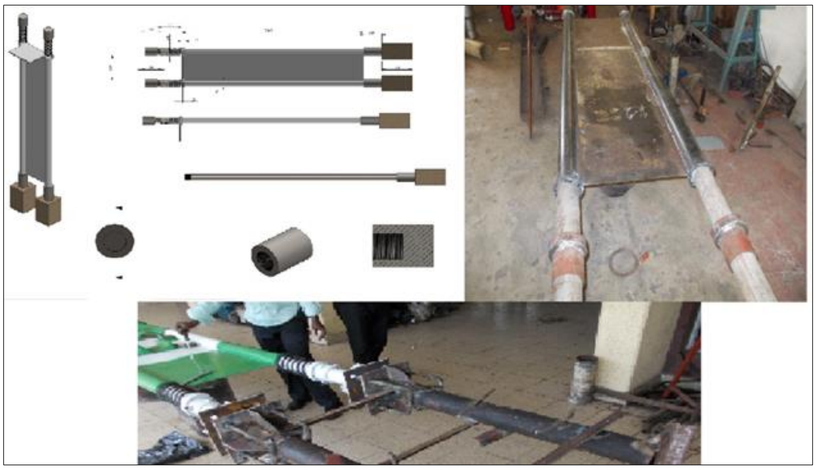

Figure 5 Liquid rocket engine static test stand fabrication process

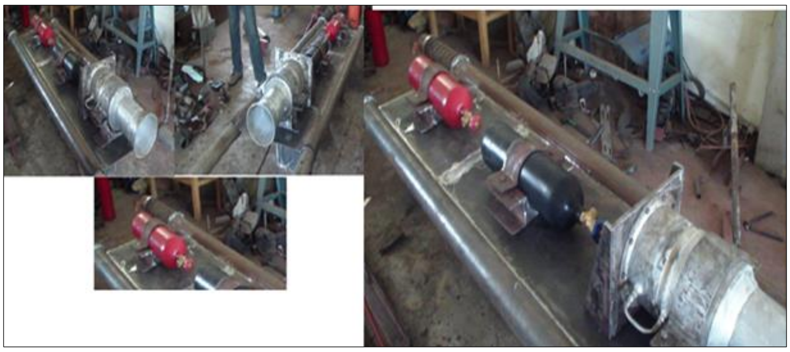

Figure 6 Liquid rocket engine fabrication processes

2.4. Experimentations

Figure 7 The designed liquid rocket engine static test

The engineering capability of the design and development of 8kN liquid rocket engine was tested for operationabilities, feed system testing, valves calibrations, injectors releases and combustion dynamics testing. Two static tests were carried out for validation of the designed engine workabilities and performances. The tests were carried out using a test facility designed at the centre (Figure 7). The test started with the release of gaseous oxygen and its calibration. The release of the kerosine from the tank followed. The flows were allowed to stabilise before ignition. These methods helped the atomization and mixing both at the injector and the chamber steadily respectively before ignition. The measure thrust profiles were measured using data acquisition system [13][14]. Other major objectives of the experiment were the compatibilities of the propellant combination and feasibilities of the designed engineering of the valve calibrations and assembly.

A blow-down feed system was adopted in the experimentation. The blow-down mode in a liquid rocket engine works by pre-pressurizing the propellant tanks before ignition of the propellant. As the engine fires and propellant is consumed, the ullage volume increases and tank pressure naturally drops. This droppage is regulated through a control valve. The method of pressured fed system is suitable and very mass-efficient for this design because it has the lowest dry mass of any feed system [15][16]. However, it is propulsion-inefficient. Since chamber pressure drops during the burn, thrust and specific impulse also decrease. However, certain amount of residual propellant is left to maintain minimum feed pressure. The only implication is that the tanks are heavier. Blow-down is adequate and commonly used for short burns, low ΔV missions, and applications where simplicity and reliability matter more than peak performance. This is what we priotized at this stage of our design.

3. Results and discussion

Results from the two static hot-fire tests were conducted to validate an 8 kN liquid rocket engine using kerosene and gaseous oxygen as propellants and are presented. The measured performance parameters are presented in Table 5 and their thrust profiles in Figure 8. Neither of the two tests achieved the 8 kN thrust target. Test 2 exhibited improved performance across all primary metrics. Compared to test 1, test 2 showed a 7.4% increase in maximum thrust, a 7.2% increase in average thrust, and an 11.5% increase in specific impulse. Burn time also increased from 7.43 s to 7.80 s. The hot-fire test results indicate that while the 8 kN design objective was not met, test 2 demonstrates a clear performance improvement and provides a validated baseline for further development of an indigenous kerosene/GOX engine. Test 2 achieved 78.5% of the target maximum thrust and 73.9% of the target average thrust. The increase in specific impulse from 123.56s to 137.81s suggests improved combustion efficiency and propellant mixing in test 2. This is likely attributed to better flow calibration in the injector system, which enhanced atomization of kerosene and mixing with gaseous oxygen, leading to more complete combustion.

Table 5 Measured Delivered Ballistic Parameters

S/N	Parameters	Unit	Test 1	Test 2
1	Max Thrust	KN	5.850	6.281
2	Average Thrust	KN	5.509	5.908
3	Total Impulse	KNs	41.322	46.086
4	Specific Impulse	s	123.56	137.81
5	Burn Time	s	7.43	7.8
6	Burn Rate	m/s	0.0037	0.0035

The longer burn time and 11.5% higher total impulse for test 2, despite a marginally lower effective burn rate, indicate that more of the propellant energy was converted to useful thrust over the duration of the firing. The thrust deficit relative to the 8 kN target can be attributed to several factors inherent in liquid engine development. Also, this could be attributed to the implication of blow down mode adopted in the design. Its inadequacies in propulsion efficiency (i.e. 85-92% of the total ΔV that a regulated system could for the same mass) [16]. This could be seen in the results obtained. That is, the inability of the both engine tests to deliver 8kN as expected. Also, the measured thrust values imply that the chamber pressure was below the design prediction. This may be due to non-optimal mixture ratio, injector pressure drops, or heat losses to the chamber walls. The improvement in specific impulse (Isp) in test 2 confirms that injector design is a dominant factor. Incomplete atomization or poor oxidizer-fuel distribution in test 1 likely resulted in combustion inefficiency. The nozzle expansion ratio may not have been fully optimized for sea-level test conditions, limiting the conversion of chamber energy to kinetic energy and thus thrust. The repeatability of test duration and stable thrust traces validate the test set-up, feed system, and data acquisition. The results confirmed that achieving the 8 kN target for a kerosene/GOX engine will require further indigenous optimization efforts focused on the follows. Increasing the number of elements and optimizing orifice sizing to improve propellant atomization and mixture ratio control. Increasing chamber pressure through higher propellant flow rates while managing cooling to prevent wall erosion. Re-designing the throat area and expansion ratio to maximize thrust coefficient for the target operating condition.

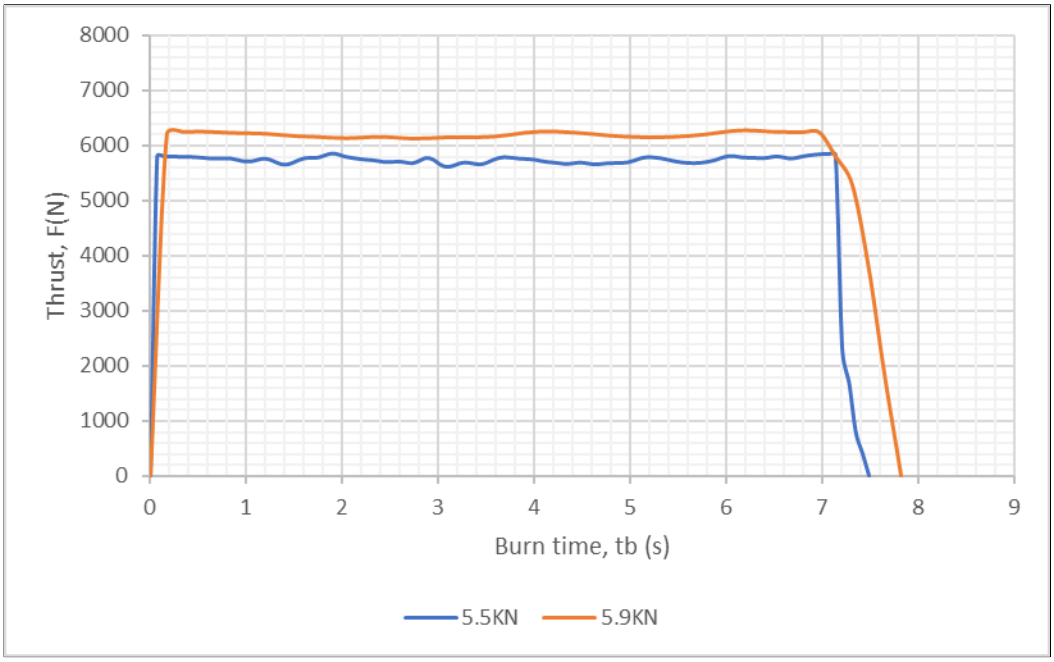

Figure 8 Thrust-time curves for both test 1 and test 2

4. Conclusion

The successful design, fabrication, and static hot-fire testing of 8 kN liquid rocket engine represent a significant milestone in the establishment of an indigenous liquid propulsion capability. In addition to validating local design and manufacturing processes, the test campaign generated essential experimental data for computational model calibration and contributed to the development of test infrastructure required for future higher-thrust engines. The results provide a reliable experimental baseline to guide subsequent design iterations. These outcomes directly support ongoing indigenous research and development efforts aimed at developing sustainable and dependable liquid propulsion systems using domestic resources. This paper presents the complete development cycle, including design philosophy, manufacturing approach, test facility set-up, and static firing results. Major technical insights and the strategic roadmap for scaling to launch vehicle class engines are also discussed. This work further advances capacity building in aerospace propulsion and demonstrates the growing contribution of Nigerian engineers to the nation's space launch capability.

Compliance with ethical standards

Acknowledgments

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Disclosure of conflict of interest

There are no conflicts of interest.

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