Available online at www.elixirpublishers.com (Elixir International Journal)

Fluid Dynamics

Elixir Fluid Dynamics 108 (2017) 47716-47720

ISSN: 2229-712X

The Effect of Variation of Rocket Nozzle Chamber Pressure on Exit Pressure for Optimal Gas Expansion Using Computational Fluid Dynamics (CFD)

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ARTICLE INFO

Article history:

Received: 26 April 2017;
Received in revised form: 10 July 2017;
Accepted: 20 July 2017;

Keywords

Nozzle;
Altitude;
Optimization;
Rocket;
Energy consumption.

ABSTRACT

The optimal expansion of rocket nozzle has been occasionally attained due to the variation of altitude as the ambient pressure of these altitudes decreases with increase in altitude. Optimal gas expansion occurs only at a particular altitude when the nozzle's exit pressure (Pe) is equal to the ambient pressure (Pa) of that altitude. There are special altitude adoptable nozzles such as aerospike nozzles, expansion-deflection nozzles and others designed for optimal gas expansion as the altitude increases. However, these nozzles adds weight to the rocket and are expensive for production. This study has been conducted with an experimental Rocket nozzle designed and developed by NSS (Nair Service Society) College of Engineering Palakkad-Kerala, India using computational fluid dynamics to determine the effect of varying the combustion chamber pressure to balance the nozzle exit pressure to the ambient pressure as the rocket ascends in altitudes. From the results obtained, it was observed that there is a conformance of up to twenty three (23) kilometers using the nozzle understudy for which further variation in combustion chamber pressure developed subsonic flows and shock waves in the divergent section of the nozzle. Further analysis showed that the total mass of propellant consumed was 36% per stage engine less than most conventional method of launching rockets engines to orbits. This method of variation of chamber pressure to balance the nozzle exit pressure with the ambient pressure provides potentials for reduction in energy consumption of a rocket.

1. Introduction

Since the inception of the modern rocket science by Tsiolkovsky *et al*, (1903) and satellite launch systems into the outer space orbit, the pressure thrust component (the product of the nozzle exit area Ae, and the pressure difference between the exit pressure P_e and the ambient pressure P_a among the momentum component $\dot{m}v_e$ has been a source of concern for the rocket system regarding the optimum thrust generated as a result of the ambient pressure decrease at corresponding altitudes as represented by the basic thrust equation as shown in equation (1).

$$F = \dot{m}v_e + (P_e - P_a)A_e \quad (1)$$

For the best rocket thrust, the pressure thrust component must be zero ($P_e = P_a$) to maintain only the momentum thrust which is dependent on the mass flow rate and the exhaust velocity only. It can also be deduced from the simple equation of thrust that since the ambient pressure decreases, the overall rocket thrust also increases assuming a steady exit pressure. However, the increase in the thrust of the rocket is counterproductive due to under and over expansion flow of the exhaust gases at the exit area of the nozzle. Often times in launch services the pressure thrust condition $p_e > p_a$ occurs during flight. The nozzle flow at the exit area is no longer mono directional but extend to other direction in x-y plane resulting to uncoordinated flow. As a result of this condition about 30% of the thrust generated is not useful because of bidirectional flow at the nozzle exit section which gives rise to a fraction of propellant (fuel) wastage (Burt, 2006).

The pressure difference condition $p_e < p_a$ of the thrust scarcely occurs except with the designers considerations where the condition $p_e = p_a$ is specified somewhere at a particular altitude, this results to over expansion nozzle ratio prior to the designated altitude. Studies have shown that a little bit of over expansion of about 5% is preferable to under expansion but greater values reduce the thrust to a reasonable amount (Taylor, 2009). This effect of pressure difference comes as a result of variations in altitude pressure. The pressure difference conditions consumes propellant of the rocket more than the required amount regarding the thrust output delivered, though the speed of the rocket increases but certainly with unnecessary cost that could be minimized. This is analogous to condition especially when moving from point A to point B with an average speed. The energy conserved with an average speed at the end of the trip is more than covering the same trip at higher speed.

Altitude compensating propulsion systems are not a new idea, with the vast majority of nozzle concepts developed half a century ago. After an initial surge of interest, the dominance of multiple stage launch systems inevitably caused a halt to

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the research and development of these concepts (Schomberg *et al*., 2012). A resurgence of interest in reducing the cost per kilogram to orbit corresponding with the emergence of single-stage-to-orbit spacecraft has resulted in a reconsideration of altitude compensating nozzles for modern propulsion systems (Schomberg *et al*., 2012). Unfortunately, to date, there has been little testing of full-scale nozzles. However, majority of testing conducted has been in the interests of private business and the military, information on these concepts is scarce within the public domain.

There are a number of problems that this pressure thrust differences developed to; these include;

- Limitation of rocket payload (Satellites, Probes, etc) mass
- Increase in the number of rocket engine stages the rocket system takes on board
- Consumption of undeserved propellant

2. Altitude Compensation Nozzles

A number of efforts have been employed to provide solution to the thrust pressure difference of the rocket system. The National Aeronautics and Space Administration (NASA) Scientist and Engineers in 1970 developed aerospike engine that maintains its aerodynamic efficiency across a wide range of altitudes. It belongs to the class of altitude compensating nozzle engines. A vehicle with an aerospike engine uses 25–30% less fuel at low altitudes, where most missions have the greatest need for thrust (Gajula *et al.*, 2016). Aerospike engines have been studied for a number of years and are the baseline engines for many single-stage-to-orbit (SSTO) designs and were also a strong contender for the Space Shuttle Main Engine (SSME). However, no such engine is in commercial production, basically because of it affordability due to cost of production and weight addition on the rocket system although some large-scale aerospike are in testing phases (Edwin, 2011).

In a similar vein, Plug nozzles also fits to a class of altitude compensating nozzles much like the aerospike which sustains its effectiveness at a wide range of altitudes (Lutz, 2009). Plug nozzles use a shaped rocket nozzle with a poppet-shaped plug to allow the shape of the rocket exhaust to be altered. This is used to fine-tune for changes in altitude; at low altitudes the plug is dragged back to cause the exhaust to spread out, while at high altitudes the lower air pressure will cause this phenomena to occur logically. A substitute construction for the same basic concept is to use two nozzles, one inside the other, and adjust the distance between them. This pattern has the advantage of better control over the exhaust and simpler cooling arrangements (O'Leary and Beck, 1992). However, these nozzles also contribute to energy demand and mass addition to the system that is critical to the mission requirement.

Expanding Nozzle was also developed and used in 1998. The nozzle employs a stationary or primary exit section that is enveloped by a movable one that slides up and down in way that balances the optimum condition of thrust performance. However, this design also consumes extra materials and mass that requires moving parts for operation. This is also a limiting solution regarding energy and cost of production. The concept can be visualized in the figure 3 below

Dressler et al (2001) obtained a US Patent US6591603 B2 for developing an expansion-deflection nozzle that comprises of a plug at the exit cone section which aerodynamically is compensated with the varying ambient temperature across altitudes.

Just like the plug nozzles, the only difference with the Gordon *et al* design is the attachment of the plug with the body of the combustion chamber or the throat section of the nozzle to reduce the thermal loads on the mechanism. However, this also has the same production issues as discussed for the solutions above.

It is therefore noteworthy to mention that the above efforts and solutions for ambient pressure compensation nozzles on rocket engine proved successful. However, mechanisms, mass and thermal loading showed complexity and excess energy consumption in the above solutions. Consequently, this method proposed herewith will provide less complexity and cost effective in ambient pressure compensation design by simulation of the corresponding chamber pressures that will deliver equivalent exit pressures in view of optimizing thrust as the ambient pressure varies along the altitudes. The chamber pressure values will be used to obtain determinant mixture ratios considering a range of propellant for controlled and automated fuel intake system in view of appropriate optimization of thrust. This would further reduce the amount of fuel consumed and also increase the payload mass. Furthermore, the chamber pressure and the exit pressure of a rocket engine determine the nozzle geometry in terms of the expansion area ratio and therefore affect the exit angle. This study will also look at the limits of chamber pressure variations with respect to the exit pressure by considering a range of chamber pressure using the computational fluid dynamic software.

3. Method and Procedures

The experimental rocket nozzle's dimensions as shown in table 1 are geometrically modelled in two dimension for ease of analysis with computer aided design software called ANSYS designed modeler using Workbench Platform. The nozzle model is then imported into an ANSYS mesh software to be meshed for a total of 5527 faces and an orthogonal quality of 0.75 and skewness of 0.24. The boundary zones are specified as Inlet, Outlet, Axis and Walls with the interior indicated as a fluid zone. Thereafter, the meshed surface is imported to ANSYS FLUENT for set up and subsequent computations. The material for the fluid zone is specified as the product of combustion of LOX/RP-1 with it accompanied thermo- physical properties. The energy equation is being enabled with a k-ω turbulent model for viscous effects. The boundary conditions for the inlet, were being specified at 14.2 MPa and inlet temperature of 3400 Kelvin. The operating pressure is indicated at the ambient temperature of 63115.74 Pa at the designed altitude of 3.31km. Other Parameters for solution control remained at default values. Subsequently, the number of iterations were being set to 2000 for computation. The solution eventually converged at about 779 iterations. The same process was repeated for Chamber Pressures 13MPa, 12MPa, 11MPa, 10MPa and 9MPa.

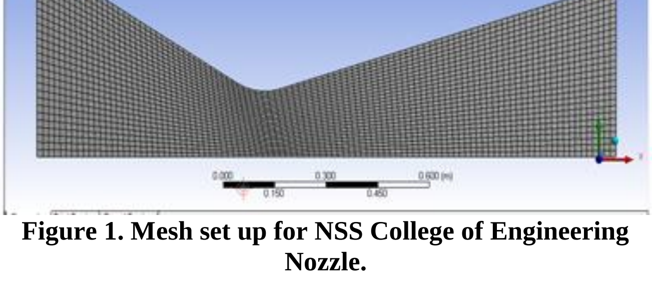

Figure 1. Mesh set up for NSS College of Engineering Nozzle.

The corresponding nozzle exit pressures at equal altitude's ambient pressures were obtained as presented in figures 2-13 in the next Chapter.

4. Results

4.1 Simulated Result for Variable Chamber Pressure using Experimental Nozzle at NSS College of Engineering Kerala, India

Figures 2-13 is a CFD simulation of 1.8 meter length nozzle designed and developed by NSS College of Engineering Palakkard-Kerala, India. Results showed a decrease in exit pressure as the chamber pressure is being lowered. The nozzle is designed at an optimal altitude of 3.31 kilometers and thereafter adopts a continual optimization of up to 21 kilometers with an exit pressure of 4.7 KPa. Any further decrease in chamber pressure creates shock waves that results to subsonic flow and therefore defeat the essence of supersonic flows in rocket nozzles. Table 2 shows the CFD parametric variables of the experimental nozzle.

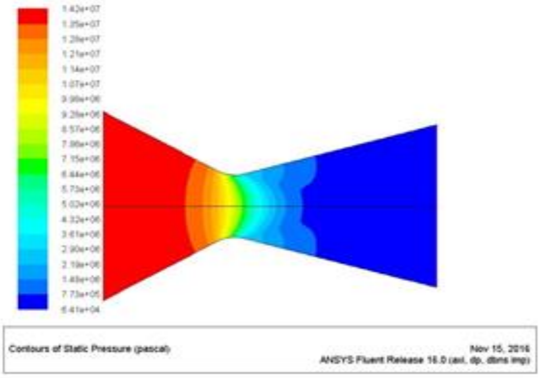

Figure 2. Contour of Experimental Nozzle Static Pressure variations at 14MPa.

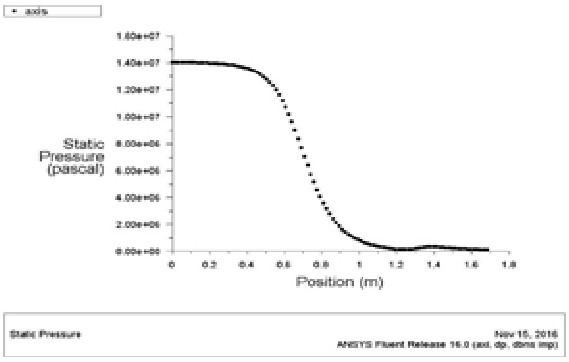

Figure 3. Graph of Experimental Nozzle Static Pressure variations at 14MPa.

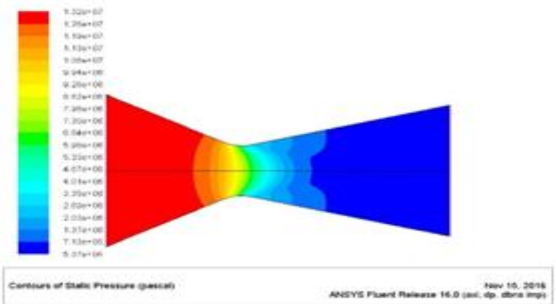

Figure 4. Contour of Experimental Nozzle Static Pressure variations at 13MPa.

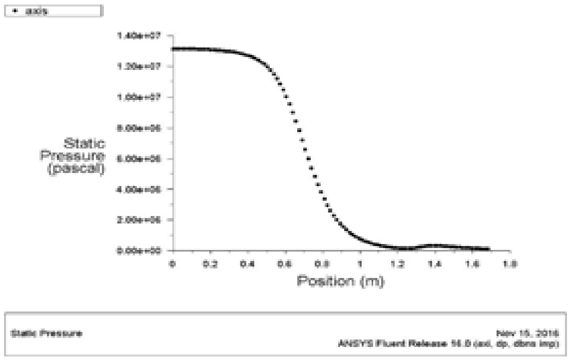

Figure 5. Graph of Experimental Nozzle Static Pressure variations at 13MPa.

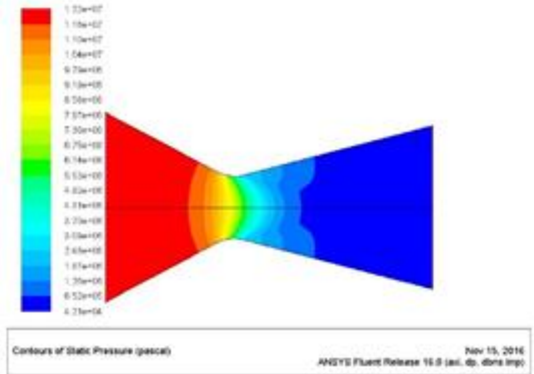

Figure 6. Contour of Experimental Nozzle Static Pressure variations at 12MPa.

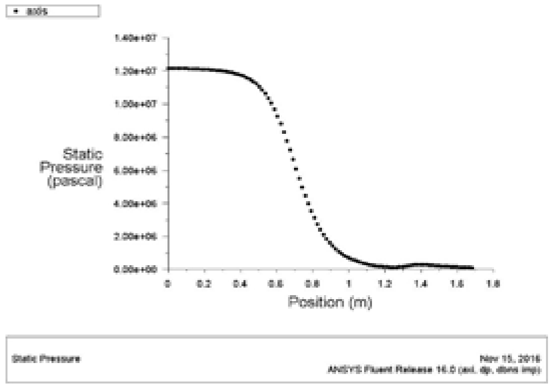

Figure 7. Graph of Experimental Nozzle Static pressure variations at 12MPa.

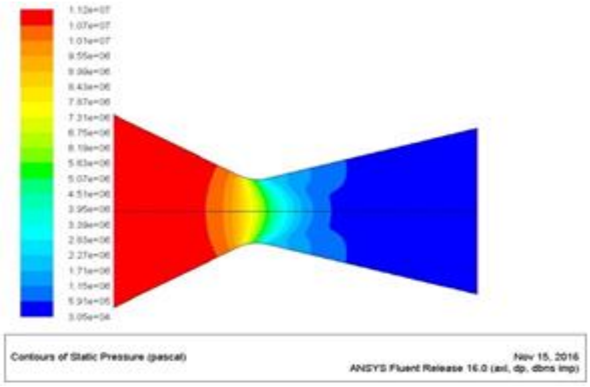

Figure 8. Contour of Experimental Nozzle Static Pressure variations at 11MPa.

Table 1.Table of results showing parametric variations of NSS Experimental Nozzle.

Chamber Pressure (Pa)	Exit Pressure (Pa)	Altitude (Km)	Mass Flow rate (Kg/s)	Exhaust Velocity, Ve (m/s)	Mach No.
14,235,960.00	63,115.74	3.31	826	2,220	3.57
13,232,530.00	53,666.67	4.45	770	2,220	3.57
12,232,660.00	42,061.86	6.15	713	2,220	3.57
11,233,060.00	30,472.17	8.41	643	2,220	3.57
10,233,330.00	18,854.93	11.77	597	2,220	3.57
9,021,266.00	4,750	21.42	527	2,220	3.57

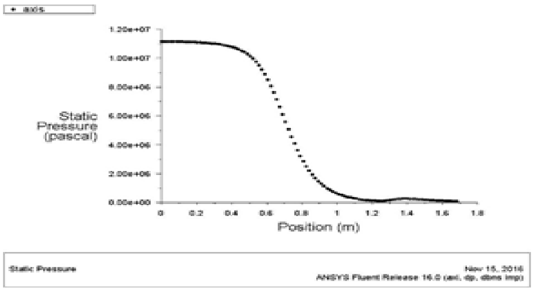

Figure 9. Graph of Experimental Nozzle Static Pressure variations at 11MPa.

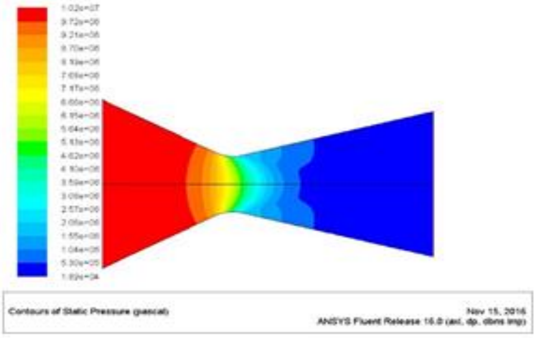

Figure 10. Contour of Experimental Nozzle Static Pressure variations at 10MPa.

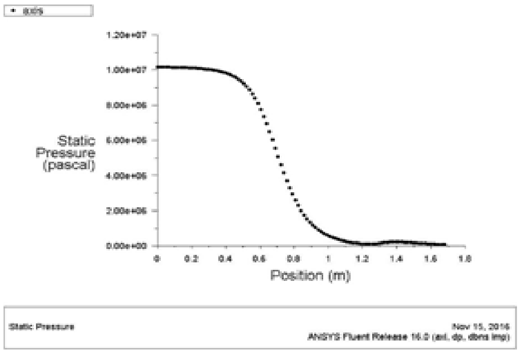

Figure 11.Graph of Experimental Nozzle Static Pressure variations at 10MPa.

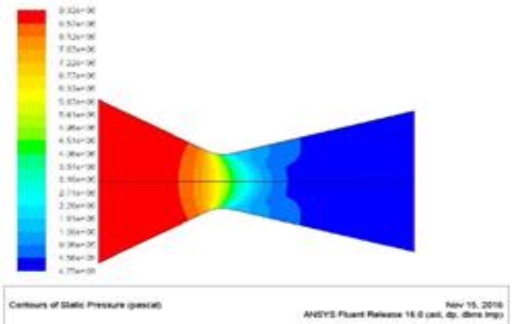

Figure 12. Contour of Experimental Nozzle Static Pressure variations at 9MPa.

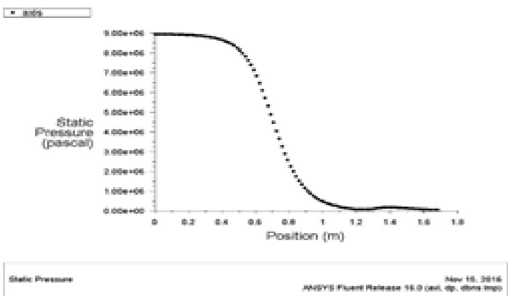

Figure 13. Graph of Experimental Nozzle Static Pressure variations at 9MPa.

4.2 Result of energy conservation for the experimental nozzle at NSS University using the developed model.

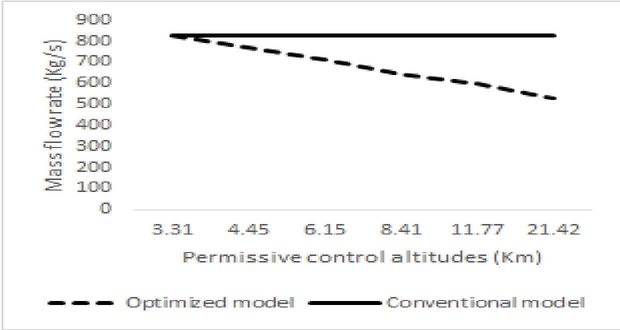

Figure 14. Graph of Mass flow rate against permissive altitude comparing optimized and conventional model.

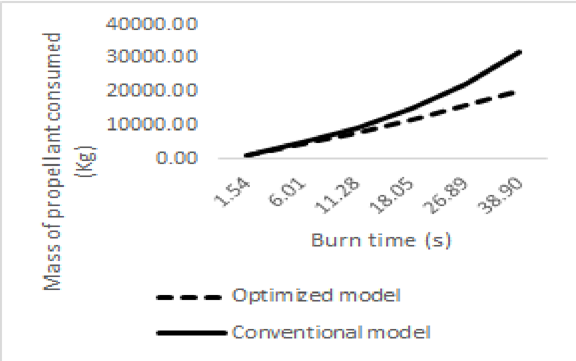

Figure 15.Graph of Mass of propellant against burn time comparing optimized and conventional model.

The energy or propellant (fuel) consumption on the experimental nozzle manufactured by NSS College of Engineering has been analyzed using the thrust optimization model for altitude compensation compared to the conventional model. Figure 14 shows a graph obtained from the data generated by the CFD analysis which the mass flow rate of the thruster is determined across the operational altitudes. The conventional model gave a constant rate of about 800 kg of propellant mass throughout the operational altitude. The optimized model has a decreasing rate of mass of the propellant across the operational altitudes. This signifies that for a total burn time of about 40 seconds, the optimized thrust model for altitude compensation in this study would consumed a total of 20,500.92 kilograms as against the 32,132.36 kilogram for the conventional model. This saves the mass of propellant or fuel for the rocket of up to 36.2 %. This is an improvement in the analysis and experimentation conducted by NASA Report on design of liquid rockets (1969) where the report stated that about thirty percent (30%) of thrust is lost due to differences in ambient and exit pressure of the rocket as a result of variations in ambient pressure as the altitude changes (Schomberg *et al*., 2012). Furthermore, another method or technique for altitude compensation reviewed is the tripropellant rocket which uses three propellants, as opposed to the common bipropellant rocket or monopropellant designs, which use two or one fuels, respectively. Tripropellant rockets appear to offer fairly impressive gains altitude compensation designs, although to date no tripropellant rocket design has been developed to the point of testing that would prove the concept (Verma, 2002).

5. Conclusion and Recommendation

Rocket Engines take about ninety percent (90%) of its contents as fuel or propellant used as energy source to take the rocket to its desired orbit (NASA, 2012).

This means that the cost of launch services is 90% cost of fuel, the remaining components take 10%. It is important to minimize the amount of propellant used in the rocket. Among other ways of optimization of propellant (energy) consumption, optimization analysis is therefore carried out in this study by controlling the upstream parameter (chamber pressure) to maintain a balance between the exit nozzle pressure and the ambient pressure as the rocket transcends along altitudes. In order to effect a variable chamber pressure to determine this balance, a CFD analysis have been carried out by varying the chamber pressure to observe the effect on the exit pressure on the assumptions of isentropic, adiabatic, inviscid flow and one dimensional flows.

It is established in this research that the effect of chamber pressure variation with respect to altitude for optimization of thrust resulted to an exponential decrease in mass flow rate but to a certain minimum threshold, and minimal energy consumption as compared to conventional models of rocket engines being used.

A further consideration for the effect of the model was carried out using computational fluid dynamic (CFD) using a theoretical designed nozzle in ideal situation and an experimental rocket engine nozzle designed and developed in NSS College of Engineering Kerala Palakkad, India. An analysis of the model on these nozzles established that the mass flow rate decreases exponentially, the exit velocity and the Mach number remained constant as the chamber pressure is varied. Most importantly, the CFD analysis showed that the effect of chamber pressure variation for optimal thrust and nozzle expansion is limited to some extend of about 19 kilometers with NSS College of engineering experimental nozzle, thereafter, the nozzle might experience shockwaves or subsonic flow in the divergent section of the nozzle. In terms of energy conservation, CFD analysis also showed that a total mass of propellant saved by adopting the optimal thrust and expansion model give rise to 36% energy conserved as compared to the conventional model. Consequence to these results, the following prepositions are possible;

That for a rocket of three staged engine, at least two of the engines can adopt optimal thrust and expansion model in this research, thereby saving 72% of the energy utilized.

That an optimal thrust and expansion control range of maximum 23 kilometers is possible on rocket engine if this model is adopted.

6. Recommendations

Further research and development should be continued in this study regarding the determination of the flow rates of fuel and oxidizer for these chamber pressures at the respective altitudes.

This development requires designers and control Engineers to involve position sensors and actuators to give correct flows of the fuel and the oxidizers to the combustion chamber after the sensors could have determined the altitude or the ambient pressures at those altitudes.

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