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EFFECT OF THRUST ON MATHEMATICAL MODEL OF PRESSURE-FED LIQUID ROCKET

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ABSTRACT: This research aims to show the importance of mathematical software's by pointing out various instances where and mathematical modeling has been used. The second aim is to develop a simple (non-transient) mathematical model for determining nozzle parameters, propellant feed system and pressuring system and engine specifications i.e. main components of most rockets and how they vary with thrust. The model is then verified and validated by three common methods.

Keywords: Mathematical modeling, LOX, LH2, Pressure-fed

INTRODUCTION

As space exploration gains widespread interest, more countries are looking to develop their space programs in order to join the space race. There is no end to the future prospects to space nor there is to the dangers faced by the astronauts. The very first dangers faced on Earth itself are related to getting into space. The unusually high escape velocity of the Earth makes making rockets very difficult. From fuselage heat shield to cryogenic fuel tank, from main engines to orbital engines, from simple indicator LED's in the command module to wiring within the engines, each aspect is faced with a challenge.

The research carried out during the course is based on pressure-fed cycle in which the whole system is normally heavier than counterpart i.e. turbo pumps. Pressure is used to introduce propellants into the thrust chamber which makes the system simpler with less moving parts. Less moving parts lead to a simpler design and less chances of failure

Author in (Kluger, 1958) developed digital computer program for start and shutdown transient analysis to provide realistic simulation of propellant lines and combustion deadlines. However, a hydraulic oscillations act as pressure surges appears but the instantaneous propellant flow and transient pressure determined accurately. (Huzel and Huang, 1992) gave insight to components, subsystem design, engine systems design, engine development, flight vehicle application, actual rocket-engine design basic physical and design principles for advanced students and engineers in all phases. (Allahverdiza and Dadashzadeh, 2014) investigated thrust level control of liquid propellant engines by formulating PID and FUZZY controllers on non-linear systems of liquid propellant engine in which FUZZY controller was found to be a good choice as per comparison results on intrinsic and informational imprecision. (Frey *et al.,* 2010) presented Astrium hot-gas simulation tool code named as Rocflam-II to validate experimental subscale data and compare wall heat flux and combustion efficiency. And verified the suitability and versatility of Rocflam-II code. (Santana *et al.*, 2000) presented robust modelling, stability and dynamic analysis of liquid propellant rocket engines (LPRE) in which linear models of components were obtained afterwards laplace transforms was applied to represent block diagram. Robust performance was obtained in terms of step response while analyzing simple pressure fed (LPRE) system. (Habiballah *et al.*, 1998) investigated the main design issue in liquid propellant combustion devices and processes involved in it optimization of design methodology. He also suggested high pressure liquid oxygen/hydrogen injection and combustion.

Propellants on which this research is based are liquid cryogenic bipropellants; named Liquid Hydrogen and Liquid Oxygen due to their excellent combustion properties, highest specific impulse of any Liquid-based space propellants, and low molecular mass of Hydrogen fuel all of which contribute towards decreasing the mass of propulsion system. With the rocket industry expanding rapidly, mathematical modeling has found itself at the heart of all missions. However, even widely gaining attention is the accuracy of these models to represent small detail with precision including; transient, development testing and disaster management. This paper investigates the role of different angle on injection fraction rate of a doublet injector. The optimum angle is 50 degrees.

With the emergence of faster modern computers, transient modeling is now expected to increase. While most mathematical models were build assuming steady state conditions, only the transient models can give the true picture of events occurring as the system reaches its design conditions bringing more flexibility. Hypothetical dangers to the rocket models cannot be foreseen by steady state models while they can be foreseen with transient models. Fuel injectors cause atomization i.e. fine spray particles that are easy to burn, making fuel injector making it crucial to the efficiency of rocket engine. Injector performance may be enhanced by altering many parameters relating to the injector and propellant.

The experiment reported success and emergence of new lightweight Engines with less elaborate cooling systems. The external pressurization was stable and behaved as expected. Self-pressurization however proved to be elusive but promising, not yielding the predicted results but. Its promising nature however makes it a desirable for the future (Chakroborty and Bauer, 2004).

MATERIALS AND METHDOS

Nozzle: A converging diverging nozzle is the most crucial part of rockets with the purpose of accelerating subsonic flow to supersonic speeds to aid rockets in achieving escape velocity. A conical nozzle was chosen to be at the heart of nozzle design in this research. An isentropic nozzle is the most efficient nozzle design. To ensure highest possible efficiency of the nozzle, its design is therefore centered on isentropic flow equations. Mass flow rate, Mach, temperature and pressure are modelled on isentropic flow whereas its nozzle diameter at throat and exit is based on geometry of circle.

Thrust Chamber: Modelling this part is based on the concept of characteristic length or L^*

$$L^* = \frac{V_{\text{comustion chamber}}}{A_{\text{throat}}}$$

L^* measures thrust chamber efficiency independent of nozzle efficiency. It is the product of mass flow rate and average combustion products density, essentially becoming a function of time during which propellants stay in the combustion chamber. Since this time is very small and difficult to measure, L^* excludes any need to measure this time making thrust chamber calculations much simpler.

Cryogenic Pressure Tanks: These store liquid propellants under pressure until needed in the combustion chamber. There are two essential aspects needed in modeling cryogenic pressure tanks,

1. Material Strength
2. Material Thermal Properties

This research focuses only on material strength of double hemispherical ended cryogenic pressure tank. The figure below shows how a double spherical ended tank looks like.

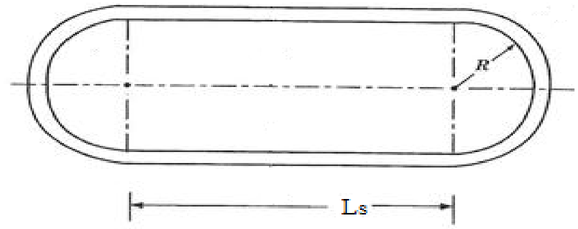

Figure 1: Double Spherical Ended tank

Pressurant Tank: The hydrogen pressurant used for this research is housed in a spherical pressure vessel. Pressure-fed cycles operate by feeding propellants into combustion chamber by using a pressurant gas stored at low temperatures (200 – 270 K) and high pressures (20 – 30 bar). The pressurant gas normally used is helium because of its inertness and low density, which contribute to better safety and lower overall mass of the system.

Algorithm: Mathematical coding always begins with an algorithm. It simplifies working by defining the scope, inputs required and necessary assumptions. The Algorithm developed for this model is shown below:

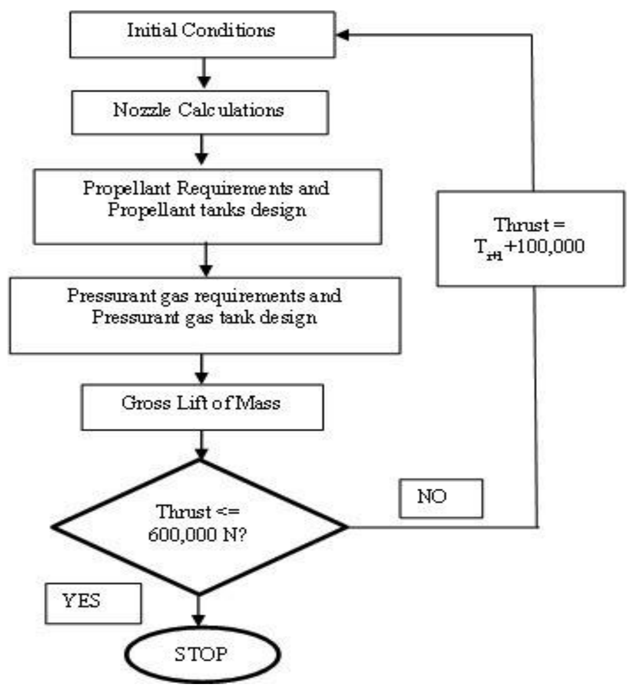

Figure 2. Approach to Mathematical Modeling to achieve the stated objective

Initial Inputs: The following values serve as inputs of MATLAB code:
Burn Time = 100 s
Mass of Payload = 200 kg
Earth's gravity at sea level g = 9.81 m/s2
Atmospheric Pressure at sea level= 101,325 Pa
Ratio of specific heats γ = 1.26
Mr = 18 kg/kmol

Oxidizer to Fuel Ratio O/F = 3.5
Chamber (Stagnation) Temperature = K
Chamber (Stagnation) Pressure pc = 34.4 bar
Nozzle Efficiency= ε = 0.99
Expansion Ratio = $^{A_{exit}}/_{A_{throat}}$ = 8
Characteristic Velocity = C* = 1600 m/s
ISP = 347.208 s

Tsiolkovski's Equation: Ideal Rocket Equation or more commonly known as Tsiolkoski's Equation, is the core equation that determines the mass ratio of space crafts for any mission.

$\Delta V = ISP * g_0 * \log(\frac{\text{Total mass}}{\text{Mass after burn}})$

From Earth's surface ΔV is a measure of velocity needed by the rocket for orbital insertion.

In space, it is the measure of velocity needed by the space craft for orbital maneuvers of such as corrections and movement into different orbits.

In cases where engine specifications and exhaust velocity are known, the mass fraction of the spacecraft can be calculated.

RESULTS AND DISCUSSION

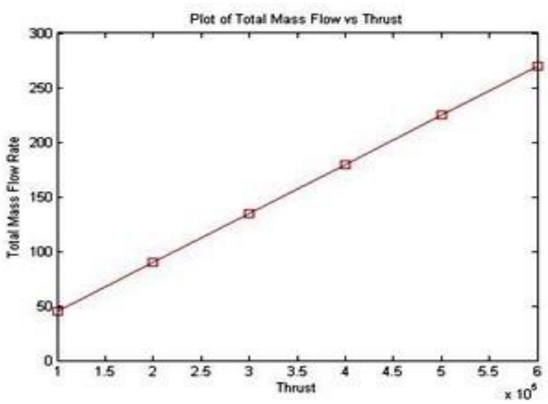

Figure 3: Mass Flow Rate of propellants with increase in thrust.

Figure 3 (above) shows an increase in mass flow rate of propellants. The increase is to be expected since mass flow rate is related to thrust by the equation below

$$\dot{m}_{total} = \frac{F_{Thrust}}{I_{SP} * gravity}$$

This equation used in the mathematical model specifically implies that thrust is directly proportional to mass flow rate. It is however essential to note that thrust changes with altitude so the value selected is fixed at sea level.

Figure 4 plots increase in propellant mass required with an increase in thrust. Propellant mass requirements are calculated by the equations written below,

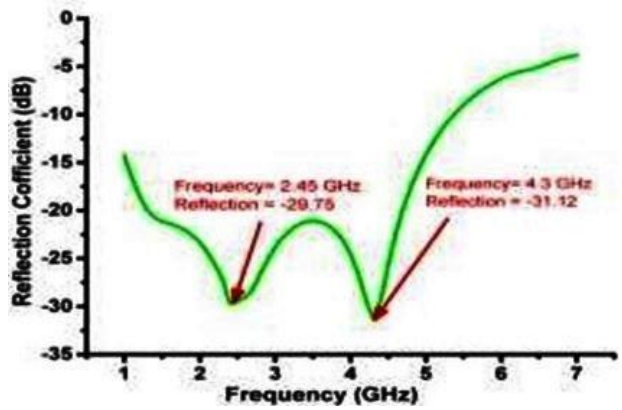

Figure 4 Required Propellant Mass with respect to thrust.

$$\text{Req. Propellant} = \dot{m}_{prop} * \text{Flight Time}$$

Since the total mass flow rate, dependent of thrust varies, the total propellants required for flight also vary.

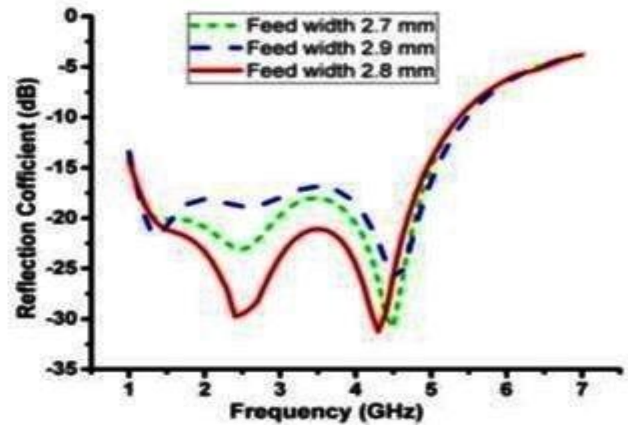

Figure 5 Increasing propellant cylinder dimensions with increasing propellant requirements.

Similarly, as propellant requirements increase, bigger tanks are required to house the propellants. So propellant tanks increase in size. This is shown in figure 5.

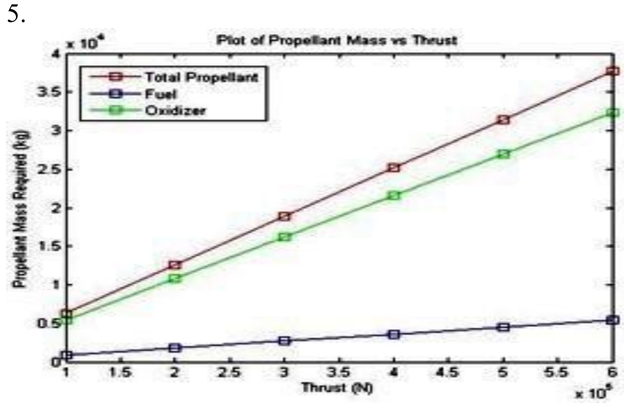

Figure 6 Effect of thrust on Nozzle Dimensions

Nozzle throat and exit diameters determine the most important dimensions affected by thrust. Increase in mass flow rate would definitely contribute to greater throat area of the nozzle as shown by its equation below.

$$A_{throat} = \frac{\dot{m}_{total} * C^*}{p_c}$$

Where, C*= Characteristic Velocity

Since expansion ratio is fixed, nozzle exit diameter is expected to depict exactly the same behavior as its throat diameter with respect to thrust.

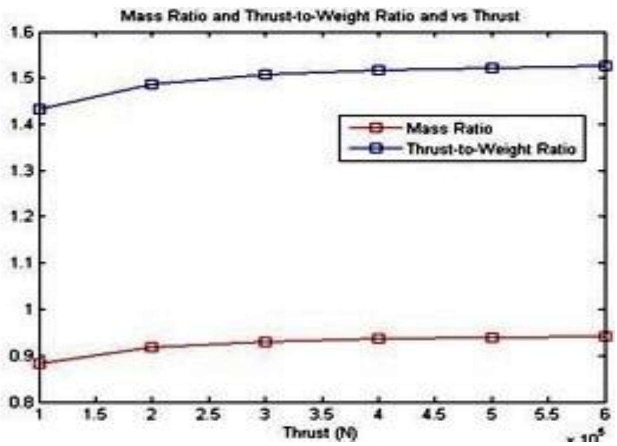

Figure 7. Change in mass ratio and thrust ratio with increase in thrust

Mass ratio and Thrust to Weight ratio increase gradually with increase in thrust.

Theoretical Calculations: All calculations carried out on MATLAB were also handwritten to identify errors in equations, logical errors, input errors. Several mistakes were identified in this way and corrected. Extraordinary values were also checked multiple times.

Program Testing: The program initially written on MATLAB Simulink was tested by running the same code in software to confirm if the software was performing the very function it was designed to perform correctly.

Comparison against Existing Project: Existing project based on pressure-fed cycles were found and the necessary inputs were fed into the program and run to test if the program yielded results or with relative standard percentage error below 30% (RSPE < 30%). Aquarius rocket is a pressure-fed concept rocket designed by Space Launch Systems/Loral in partnership with Microcosm Inc. and Wilson Composite Technologies.

Conclusions: Change in thrust affects: Nozzle Design, Cryogenic pressure tank design, Pressurant tank design and Thrust Chamber remains unaffected by change in thrust which implies mathematical modeling is inefficient method to calculate thrust chamber length.

Future recommendations: Heat transfer and thermal equations can be involved in determining thickness of Cryogenic pressure tanks. Transient modelling could be used as an improvement of the current mathematical model.

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