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Research Article

Study of Paper on Rocket Engine Thrust Chamber and Nozzle Cooling

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ABSTRACT

Modern rockets used in space missions have thrust generated of thousands of newtons. Thrust generated due to burning of propellants and oxidiser results in huge temperature and heat generation. Most heat flux occurs near proximity of nozzle throat which needs cooling as combustion temperature and heat is a threat to structural metals. This paper reviews different cooling methods and efficacy on rocket chambers and nozzles. In this paper experimental and numerical modelling investigation done by researchers is also highlighted with results and discussions. This paper also brings in light of critical research gap to be addressed by future generations. A rocket nozzle must absorb a wide range of loads caused by thermal expansion and contraction, as well as shocks from startup pressurisation and flight accelerations. A rocket engine nozzle is provided that is capable of dampening nozzle vibrations generated during operation. The nozzle has an annular closed chamber surrounding it near the gas exhaust end. Within the chamber, there is a dense but unrestricted particulate mass capable of frictional movement.

Keywords: Rocket thrust; Nozzle; Propulsion engine; Heat transfer; Cooling

INTRODUCTION

Most commonly used rocket engines in space missions are turbo pump fed liquid propellant engines with solid propellants boosters due to higher thrust. Liquid propellant engines consists of liquid oxidizer i.e oxygen and a liquid fuel such as kerosene/methane/hydrazine.in combustion chamber propellants react producing hot gases which is finally to be accelerated through supersonic nozzles and imparting thrust to vehicle whereas in solid propellant engines propellant to be burnt is stored in combustion chamber in form of grains. Once ignited there's no stopping as internal cavity grows larger [1], It is important to safeguard nozzle and thrust chamber walls from high temperature and heat generation as combustion temperature is more than melting point of metals. If not cooled properly the chamber no longer withstands thermal stress and it will eventually fail [1]. This paper evaluates all cooling techniques available which is regenerative cooling, radiative cooling, ablative cooling, film cooling, transpiration cooling, inert heat sink, endothermic heat sink, open tube convective cooling (dump cooling) (Table 1) [2].

Table 1: Showing ISRO rocket engines with propulsion system used in space missions [2].

Isro Rocket Engine	Propulsion System	Height (m)	Lift-Off Weight (tonne)	Payload Mass (kg)	Orbit
SLV - 3	All Solid	22.7	17	40	Low Earth Orbit
ASLV	All Solid	23.5	39	150	Low Earth Orbit
PSLV - XL	Solid and Liquid	44	320	1860	Sun Synchronous Polar Orbit
GSLV MK II	Solid, Liquid and Cryogenic	49	414	2200	Geosynchronous Transfer Orbit
GSLV MK III	Solid, Liquid and Cryogenic	43.43	640	4000	Geosynchronous Transfer Orbit

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Design considerations of thrust chamber

For complete combustion, sprayings, mixing of propellants and oxidizer, volume should be larger, for different propellants volume may varies. If chamber pressure is high and propellants being used for reactive with proper mixing then small volume chamber can be used. If cross sectional area and volume of a thrust chamber is small then mass and surface area will be small then gas velocity and rate of heat transfer increases but if cross sectional area and volume is larger than rate of heat transfer decreases. The idea is to keep total heat absorbed by wall minimal so we chose optimal diameter and volume for chamber. Chamber mass is a function of diameter, length, chamber pressure, nozzle area ratio. This can affect cooling requirements. The chamber pressure must not exceed a certain value as increase in chamber pressure will increase heat transfer [1]. Selection of thrust chamber depends on various parameters which are typical requirement of propulsion system. Which depends on type of mission, engine and thrust requirement, burn time. Since combustion gas temperature and gas composition choice of propellant is important considerations while designing thrust chamber. Propulsion requirement also depends upon parameters like specific impulse, run time, number of engines, thrust chamber pressure. Operational requirement like exterior temperature requirement, engine specification like contraction or expansion ratio, nozzle type, fuel/oxidizer ratio [3].

Objective

Takeshi Kanda and Masaki Sato of JAXA studies effect of radiative cooling in thrust chamber through increasing length of cylindrical section and its diameter respectively. They found that radiative heat flux to total heat flux increasing upto 30% [4]. Mustafa Emre Boysan Modelled A LOX/Kerosene liquid propellant engine having chamber pressure of 60 bar and thrust of 300 km. He studied effect of number of rectangular cooling channels and geometry on efficiency of cooling. He found out that the best achieved efficiency of cooling is obtained at 4×2 mm^2 cross section area of channel and 100 cooling channels with relatively high pressure drop. If the increasing number of cooling channels is upto 50%, pressure drop is approximately 51%. He noticed that increasing aspect ratio will increase cooling efficiency up to an optimum level and it also increases pressure drop [5]. Alexander W. Miranda And Mohammad H Naraghi studies film cooling and presented a cfd model of varying flow rate and composition at film layer, they figured out that when pure hydrogen is used in film layer maximum film cooling is accomplished and when mixture of rich hydrogen and oxygen is injected it is minimal. Despite of all, when mass flow rate was increased then heat flux decreased [6]. Kevin Ciasullo Modelled A converging regenerative nozzle for an aluminum- water combustor, nozzle has single pass circular channel that spirals around the converging wall. The model was developed to determine potential energy transfer into nozzle, chemical equilibrium analysis in order to calculate the convection and radiation heat transfer into the nozzle from the combustion reaction and also pressure drop [7,8]. A Adami, M Mortazavi And M Nosratollah presented an algorithm which minimises the wall temperature so that high heat flux costly materials to be replaced by composite or steel materials. [9] Mo Bai studied the heat transfer and pressure drop in open cell foam and modelled alongwith numerical evaluation. Open foam cell is a porous medium having large surface area to volume ratio, intensified fluid and high thermal conductivity which enhances heat transfer rate [10]. J. Stoll And J. Straub passed highly compressed air at inlet of converging diverging nozzle. There is a turbulence mixing of gaseous and heat transfer was examined. Theoretically calculated heat flux was compared with experimentally calculated heat flux [11]. C.J. Marek, R.R. Tacina conducted an experiment in free stream turbulence in total of four levels and found out that when turbulence intensity was increased from 7% to 35% then there was decrement in film cooling effectiveness up to 50%. [12] K.A. Juhany, M.L. Hunt, J.M. Sivo have done an experimental investigation to predict dependence of film effectiveness on mass flux, Mach number and velocity. They varied velocity and mass flux by varying total pressure. When injection rate is increased than film effectiveness also increases. Larger injected coolant Mach number increases effective coolant length [13]. T. Kanda, F. Ono, M. Takahashi, T. Saito, Y. Wakamatsu investigated effect of shock wave on supersonic film cooling in wind tunnel of mach number 2.5. When coolant was injected with sonic speed weak shock formed with pressure ratio of 1.21 has no effect on film cooling effectiveness and in restricted region strong shock with pressure ratio of 1.44 decreases film effectiveness [14]. M. Hombsch, H. Olivier conducted an experimental investigation on supersonic flow over flat plate to obtain a laminar turbulent boundary layer region. Coolant is injected at different injection angles. Different injection geometries, coolant mass flow rate and free stream conditions are investigated to obtain correlation [15]. RANS (Reynolds Averaged Navier-Stokes), DNS (Direct Numerical Simulation), LES (Large Eddy Simulation) is being used by researchers for computational modelling and analysis of cooling flows. Selection criteria differ on basis of complexity of problems to be solved. Below in Table 2 showing different authors work on simulations/modelling.

Table 2: Numerical work and focus of study.

S. No	Authors	Simulation/Modelling Tool Used
1	L.S. Jansson, L. Davidson, E. Olsson	k-ε Model and Algebraic Stress Model to Simulate Steady and Unsteady Simulations
2	J. Stoll, J. Straub	Finite Difference Boundary Layer Code To Predict Wall Heat Flux In Converging Diverging Nozzle.
3	C. Cruz, A. Marshall K. Dellimore, B. Betti, E. Martelli	Spalart-Allmaras (SA) Turbulence Model Along With RANS Based Solver
4	B. Betti, E. Martelli, F. Nasuti, M. Onofri	3D RANS Solver
5	M. Tyagi, S. Acharya, F Muldoon	LES and DNS
6	A. Matesanz, A. Velazquez, M. Rodgriguez	LES and DNS To Study Slot Film Cooling In Converging-Diverging Nozzle

MATERIALS AND METHODS

In steady state method temperature of thrust chamber and heat transfer rate reaches thermal equilibrium. It has two ways radiation and other is regenerative cooling whereas in transient heat transfer or heat sink method temperature instead of reaching equilibrium keeps on increasing with duration of thrust. In regenerative cooling jacket is built around thrust chamber and coolant which is either one of propellants fuel/oxidiser or both is circulated before it is fed into injector generally used in high thrust engine with high heat transfer rate and high chamber pressure. Preheating of fuel also increases energy level. Radiation cooling is used in medium thrust engine and medium chamber pressure and heat transfer rate, it is used for nozzle area ratio of 6-10. In this cooling method thrust chamber made up of single wall and when it gets hot glows white hot/ red and finally radiate heat to surroundings [1].

Other cooling methods are injection cooling and ablative cooling. They are also used in combination of injection, ablative and regenerative. Injection cooling thrust chamber is coated with film through flowing coolant inside it. Coolant will protect chamber wall and especially nozzle from high temperature high velocity exhaust flow of gaseous, it absorbs thermal energy of propellant fuel. Generally used as coolant because combustion near wall is lower than normal due to fuel rich conditions. There are three methods of injection cooling–(1) Transmission cooling (2) Film cooling (3) Curtain cooling. [4] In transpiration cooling fluid seeps into thrust chamber structure acting as a protective level reducing both convective and radiative heat flux and absorbs heat from chamber wall. [4] In film cooling thin film fluid is layered over thrust chamber wall to protect from heat flux, small quantities of coolant are injected through orifices of injector. Coolant must have high boiling point and high heat of vaporization [1]. Ablative cooling a mass of solid ablative shield is coated inside of thrust chamber, the shield burns and takes away heat while rest materials insulate thrust chamber [1]. regenerative cooling is widely used as in liquid propellant engine fuel is cryogenic so having large temperature difference between combustion gas and coolants which increases heat transfer rate along with preheating of fluid increases energy level and also cooling jacket reduces thrust chamber weight [17]. It is clear from below diagram showing variations of strength of materials at different temperature that at very high temperature, strength of materials degrades which is why cooling is so important in engines. Metals when operated at elevated temperature then their properties changes, it becomes more ductile, the strength increases a little first then decreases, generally ferrous metals strength is maximum at 200 °C (Table 3a and 3b) [10].

Table 3a: Strength of metals (% of their strength at 20°C) WRT To Increase In operating temperature [10].

Materials	100°C	200°C	300°C	400°C	500 °C
Copper	95°C	83°C	73°C	59°C	42°C
Bronze	101°C	94°C	57°C	26°C	18°C
Cast Iron	-	100°C	99°C	92°C	76°C
Wrought Iron	104°C	112°C	116°C	96°C	76°C
Steel Casting	109°C	125°C	121°C	97°C	57°C
Structural Steel	103°C	132°C	122°C	86°C	49°C

Table 3b: Representation of effect of cooling methods, choice of propellants, limit and losses and geometry [3].

Cooling Method	Effect On Propellant Choice	Heat Transfer To vehicles	Advance Nozzle Geometry	Exterior Engine Temperature	Propulsion System Losses	Chamber Pressure Limits
Regenerative	H2, N2H4, N2H4+ EDA 0.5 N2H4+ 0.5 UDMH	Minimum	Limited By Passage Sizes	Approaches Coolant Temperature 500 °F to 1000 °F	Pressure Losses	Affected By Coolant Passage Design
Radiation	Compatibility With Walls And Coating Critical	Maximum	Limited By Pressure Configuration	Maximum 3300 °F	Large Chamber Size Due To Low Pc	50 psi Or Less Than 90 psi For Low Thrust
Ablation	Limits Run Time	Long Soak Transient	Throat Erosion Critical	Maximum 500 °F to 800 °F	Chamber Weightand Run Time Limit	Limits Run Time For Throat Applications
Film	Cooling Properties Important	Minimum	Applicable	Can Be Controlled	Isp Loss	No Limit
Transpiration	Cooling Properties Important	Minimum	Applicable	Can Be Controlled	Isp Loss	No Limit
Open Tube	H2 Best	Coolant Temperature May Be<1500 °F	Applicable Limited By Passage Size	1500 °F	Affects Optimum O/F	No Limit
Inert Heat Sink	Compatibility With Walls And Coating Critical	Function Of Time and Soak Transient	Time Limited	Limited<1000 °F	Time Limit Vs Chamber Weight	Limits Run Time
Endothermic Heat Sink	Flame Temperature Important	Can Be Limited	Time Limited	Can Be Limited	Time Limited	Limits Run Time

RESULTS AND DISCUSSION

A rocket engine creates a high-speed propulsive jet of fluid, often high-temperature gas, using stored rocket propellants as the reaction mass. According to Newton's third law, reaction engines like rocket engines generate thrust by expelling mass rearward. Rocket engines can be used to propel spaceships and ballistic missiles in a vacuum as, unlike most combustion engines, rocket vehicles carry their own oxidizer. A rocket engine is a reaction engine that can be used for spacecraft propulsion as well as terrestrial uses, such as missiles. Heated exhaust being expelled that functions similarly to the basketball. Although the gas molecules that make up the exhaust don't weigh much individually, they have a lot of momentum since they leave the rocket's nozzle quickly. As a result, the rocket travels in opposition to the exhaust with the same total amount of force.

CONCLUSION

The focus of present study was to investigate different thrust chambers and nozzle cooling methods available and generally used in rockets in various space missions. The author thoroughly studied various research papers and discussed their mentioned works in a very crisp and crystal way to be easily understood. The author also studied parametrically all methods available and believes that even with huge advancement in composite materials which has higher temperature and higher thermal conductivity, thrust is limited due to cooling capacity. Today the need is of faster and larger duration thrust rockets so cooling capacity limitations has to be extended and more focused should be on low cost cooling methods so author in future would like to do numerical evaluation while modeling heat exchange rate and heat transfer analysis in gambit/fluent and later validating using experimental setup especially in combination of regenerative cooling, film cooling, injector biasing, and transpiration cooling. First model would be using coax element injector c 103 as thrust chamber material and regenerative cooling while second would be impinging doublet injector, haynes 230 as thrust chamber material and ablative cooling.

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