ROCKET ENGINE NOZZLE COOLING

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PROJECT GUIDE

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ABSTRACT

Rockets are such fascinating machines that when we look up we feel how far our universe has expanded over the years. From the beginning of civilization humans always looked up in the sky and got inspired. Always wanted to fly and get into space to explore what lies ahead. Counter argues by philosophers that humans first need to focus on earth but few audacious men and women break the jinx and build vehicles after years of failures and hard work which takes humanity to another level. It is because of that fearless and moonshot vision that we achieved a lot in terms of communication and connectivity. Humans landed on moon & mars, sent satellites in orbit and sent rovers deep into space and all of this was and is possible because of thrust produced in tons due to propulsion. Modern rockets used in space missions have thrust generated of thousands of newtons. On an average countries and private rocket industries launch more than 100 rockets carrying satellites every year. Thrust generated due to burning of propellants and oxidiser results in huge temperature and heat generation. Most heat flux occurs near the proximity of the nozzle throat which needs cooling as combustion temperature and heat is a threat to structural metals. We have progressed so far and further deep space exploration needs better machines structurally to handle thermal temperature and also fuel efficiency and there comes cooling techniques. Different cooling techniques have been implemented and experiment is still in continuation to combine different such mechanisms to achieve better results. Scientists have inspired the future generation to continue R & D on cooling techniques with different nozzle designs, materials and cooling methods to achieve superior cooling experiences. In this dissertation rocket nozzle cooling methods have been discussed and efficacy on rocket chambers and nozzles. In this dissertation experimental and numerical modelling investigation has been performed and is also highlighted with results and discussions. A spacex falcon f9 rocket is made in cad also analysed with volume enclosure to find out thermal stress and maily a merlin 1d engine is designed and regenerative cooling is performed and results has been compared with combined regenerative + ablative & film cooling and the cooling results are shown along with thermal simulation analysis. This dissertation also brings in light of a critical research gap to be addressed in future.

CONTENTS

Candidate Declaration	ii
Certificate	iii
Acknowledgement	iv
Abstract	v
Contents	vi
List of Tables	vii
List of Figures	viii
Symbols	x
Abbreviations	xi
Chapter 1: Introduction	**13-21**
1.1.1: Motivation	10
1.1.2: Rocketry	11
1.1.3: Rocket Engine cooling	13
Chapter 2: Literature reviews	**22-32**
2.1: Literature reviews	25
2.2: Objective of the present study	32
Chapter 3: Methodology	**33-34**
3.1.1: Theoretical hand calculation	33
Chapter 4: Results & discussion	**45-59**
4.1.1: Simulation modelling	
Chapter 5: Conclusions	**60**
References	61-62
List of Publications	63

LIST OF TABLES

Table 1.1 Rockets cooled with techniques of Regenerative cooling 19

Table 1.2 Rocket engines which had ablative chambers for cooling techniques 20

Table 1.3 Rocket engines which used film cooling or combination of film cooling 21

Table 2.1 Data observed in experiments conducted 31

Table 2.2 Data observed in cooling condition of stream and blowing ratio 39

Table 4.1 Heat transfer analysis formula used shown in table 40

LIST OF FIGURES

Fig 1.1 Schematic diagram of thrust chamber in axial direction 19

Figure 1.2 Rocket engines diagram of chamber for film cooling 22

Figure 1.3 Schematics of chamber temperature profile 23

Figure 1.4 Schematics of engine chamber showing film protection layer 24

Figure 1.5 Schematics of engine chamber showing radiation cooling 25

Figure 2.1 Diagram showing maximum wall heat flux for various cases as functions of wall temperature 27

Figure 2.2 Diagram showing two phase flow & flow boiling regime 28

Figure 2.3 Diagram showing variation with respect to Lf in total mass 29

Figure 2.4 Diagram showing variation with respect to percentage NH3 in total mass 30

Figure 2.5 Diagram showing design algorithm for optimised hydrazine propulsion system 30

Figure 2.6 Diagram showing wall temperature in y axis and downstream distance and temperature of gas is 500 k 32

Figure 2.7 Diagram showing film cooling effectiveness at different mass flux, injection angle and injection temperature 33

Figure 2.8 Diagram showing film cooling effectiveness at distance from combustion entrance 34

Figure 2.9 Diagram showing setup of transpirational cooling 35

Figure 2.10 Diagram showing Non uniform and uniform flow of injection 36

Figure 2.11 Wall heat flux and temperature boundary layer during circumferential injection 36

Figure 2.12 Diagram showing injection in nozzle flow in slots and circumferential 37

Figure 2.13 Comparison of mean cooling effectiveness η and for variation of the coolant mass flow rate 38

Figure 2.14 Diagram showing turbulence stress in contours 38

Figure 2.15 Diagram showing efficiency and correlating cooling data efficiency 39

Figure 3.1 Hand calculation of rocket equation 40

Figure 3.2 Hand calculation of payload, structure & propellant 41

Figure 3.3 Hand calculation of propulsion efficiency 41

Figure 3.4 Hand calculation of heat transfer analysis in 4 stages of rocket 42

Figure 3.5 Hand calculation and variation curve of kinetic energy 42

Figure 3.6 Surface regression of solid propellant rockets 43
Figure 3.7 Hand calculation of propulsive speed of 4 stage booster rockets 43
Figure 3.8 Characteristics curve plotted for composite propellant burning rate vs pressure 44
Figure 3.9 Hand calculation of gas energy according to st roberts law 44

Figure 4.1 Diagram showing full cad model of converging diverging nozzle 46
Figure 4.2 Diagram showing cutout of converging diverging nozzles 46
Figure 4.3 Proper visualisation of converging diverging nozzles with radius 47
Figure 4.4 Proper visualisation of specific dissipation rate 48
Figure 4.5 Proper visualisation of medium selection for analysis 49
Figure 4.6 Proper visualisation of pressure boundary condition 50
Figure 4.7 Proper visualisation of numerics 51
Figure 4.8 Proper visualisation of simulation control 51
Figure 4.9 Proper visualisation of turbulent kinetic energy 52
Figure 4.10 Proper visualisation of pressure in entire nozzle 52
Figure 4.11 Proper visualisation of velocity magnitude 53
Figure 4.12 Proper visualisation of velocity magnitude in Y direction 53
Figure 4.13 Proper visualisation of velocity in Z direction 54
Figure 4.14 Proper visualisation of velocity in X direction 54
Figure 4.15 Proper visualisation of residual velocity, pressure, omega vs time in X, Y, Z directions 55
Figure 4.16 CAD model of spacex rocket for purpose of full scale temperature profile\ further 55
Figure 4.17 Temperature profile of outer surface of entire falcon 9 rockets in enclosed volume where heat transfer is taking place 56
Figure 4.18 Figure showing fully meshed nozzles used for simulations of heat transfer analysis 57
Figure 4.19 Temperature profile of outer surface of entire falcon 9 rockets in enclosed volume where heat transfer is taking place 57
Figure 4.20 Figure showing mesh log event for volume ratio, aspect ratio, non orthogonality and skewness 58
Figure 4.21 Figure showing convective heat transfer simulations and temperature profile of falcon 9 merlin 1 d vacuum engine rocket 59

SYMBOLS

A* = Area of Nozzle Throat (mm2)

Bi = Biot Number

C* = Characteristic Velocity (m/s)

cp = Specific Heat Constant Pressure (J/kg*K)

D = Jet Diameter (mm)

D* = Nozzle throat Diameter (mm)

Gj = Jet Mass Flux (kg/m2)

g = Gravitational Acceleration (m/s2)

h = Convective Heat Transfer Coefficient (W/m2*K)

k = Thermal Conductivity (W/m*K)

L= Characteristic Length (m)

M = Mach Number

M* = Mass Flow Rate (kg/s)

Nu = Nusselt Number

Pr = Prandtl Number

P = Pressure (Pa)

q* = Heat Flux (W/m2)

R = Resistance (m2*K/W)

Re = Reynolds Number

rf = Local Recovery Factor

r/D = Radial Distance [Jet Diameters]

Sc = Schmidt Number

Sh = Sherwood Number

T = Temperature (K or °C)

t = Time (s)

th = Thickness (m)

v = Velocity (m/s)

V = Volume (m3)

X = Distance {streamwise}(mm)

X/D = Streamwise Distance [Jet Diameters]

Y/D = Spanwise Distance, [Jet Diameters]

Z/D = Jet Height, [Jet Diameters]

Symbols [In Greek]

α = Coefficient of Linear Thermal Expansion (1/K)

γ = Specific Heat ratio

ε = Turbulent Dissipation Rate

μ = turb Eddy Viscosity (Pa*s)

ν = Kinematic Viscosity (Pa*s)

τij = Reynolds Stress Tensor

ω = Exponent of the viscosity-temperature relation

ηeff = Film Cooling Effectiveness

η = Cycle Efficiency

ϱ = Density (kg/m3)

ABBREVIATIONS

3-D [Three-Dimensional]

CD [Convergent-Divergent]

CFD [Computational Fluid Dynamics]

DNS [Direct Numerical Solution]

EBkε [Elliptic-Blending Kinetic Energy]

LES [Large Eddy Simulation]

LOX [Liquid Oxygen]

LPRE [Liquid Propellant Rocket Engine]

RANS [Reynolds-Averaged Navier-Stokes]

SSN Single Slot Nozzle

TKE Turbulent Kinetic Energy

Simscale

Chapter 1
INTRODUCTION

1.1.1 Motivation

As rockets are subjected to extreme conditions in space and in such conditions when temperature goes beyond 3000 K more than safe conditions of operation then materials and structural design gets compromised and chances of failure is just a matter of time. That's why we need effective cooling of the engine and nozzle to decrease combustion temperature, faster rate of heat transfer to increase efficiency and performance and ultimately rockets carry heavier payloads as fuels are saved. With increased focus on deep space research rockets need to be designed keeping in mind distance they need to travel and also cost reduction on the mission so advanced cooling techniques need to be developed as current methods cannot be effective in the long run. Since the heat load is highest at the throat region of nozzles, cooling necessities arise there (Turner, 2010). The author is simulating a model of Merlin 1d engine of spacex rocket in computational fluid dynamics method for heat transfer and other in simscale software. Regenerative cooling produces best cooling in nozzles and the author tested experimental data and also compared it with other methods originally used by spacex scientists and engineers. Rockets chamber can be made with thickest and titanium like material which can handle heat but again it can’t be possible to use it as rockets has to carry payloads and if materials are heavier it will defeat the entire purpose of saving fuels carrying extra load and being agile say faster to cover larger distance in shorter period of time so the best possible solution is to find cooling solution like carbon composites on nozzle and chamber surface which due to ignition higher temperature will burn off and take away heat which simply states that engine is for one time use only so another best methods like film cooling, regenerative cooling or radiative cooling like material got red hot and transmit heat even to vacuum through radiation mode and also combination of heat transfer methods like film cooling plus in addition to regenerative cooling techniques.

1.2. Rocketry

1.2.1 Ancient Rocketry

We indians mostly has our first experience of rockets in diwali and in that there is tube in top and stick and no bottom the tube is mostly filled with gunpowder acted as one of mini solid propellant rockets which can be ignited putting in a bottle, I used to wonder what's the role of stick but actually its guiding the direction otherwise it can go in random direction and blasts. The inspiration for this goes back through many hundred years ago in 1232 used in war between Mongols and chinese called fire arrows. A lot of experiments after that had been performed to increase the range of fire arrows [1].

But rocketry became rocket science after Issac Newton gave three governing laws and defined how and why of rockets even in outer space, after which scientists started working on it and Indian king Tipu Sultan and his son Haider Ali also used rockets in first anglo mysore war against britishers after which they started reverse engineering and developed more lethal rocket artillery and won many wars and then it opened doors for more advancement in rocket science [1].

1.2.2 Modern Rocketry & India’s ISRO

Russian school teacher Tsiolkovsky is considered as father of modern rocketry, he proposed that with better propellants and careful design rockets can be taken to outer space in vacuum as range depends on escape velocity of exhaust gas and also he had given equations for rockets still valid to this date. In between 1882-1945 Goddard an american scientists developed first sounding rocket on solid propellant and he published numerous papers on his contributions and later he thought that liquid propellant rocket can better perform than solid counterpart and he made one which flew but fell apart it was quite not upto mark but onset a beginning of liquid propellant rockets used in modern rockets. He also developed many advanced rockets later and thus with such significant contributions he is considered to be the father of liquid propellant rocketry.

Germans also pioneered rockets and started a society for space travel and many scientists emerged due to interests, visualization and availability of resources. During the war the german scientists started making missiles and made V2 a lethal rocket first used it on London city in world war 2nd. It used a mixture of oxygen and alcohol and thrust generated was enough to destroy cities and military establishments. When germany falls western part captured by united states and eastern germany goes to soviet by default and many unused rockets gone to usa russia and allied countries. Many scientists either settled in these countries and then started a period of cold war and these countries also realised the potential of rockets not just in terms of missiles but also outer space [1].

India as a country was never far behind in the space race. Country established INCOSPAR in 1962 renamed in 1969 as ISRO and the then prime minister believed in the potential of Indian scientists vikram sarabhai and homi jehangir bhabha and despite lot of challenges they were able to convince that they wanted peaceful uses of space science for development of india. Indian space interests were not driven by landing on the moon or mars but communication, remote sensing, connecting Indians through television. The geographical location of India also gave an advantage as the magnetic equator passes through Thumba in Thiruvananthapuram, the location in itself has a story of how the church campus became the so-called mecca of rocket science in india. When indian government ordered village to be relocated for thumba equatorial launching station in 1963 they protested but it was sarabhai who convinced father and then father collected people and told them about vision and now a painted picture published in india today that parts was carried on bullock carts and bicycle and from there to space india's story of first sounding rocket. After the mysterious death of sarabhai batton of ISRO was passed to satish dhawan by indira gandhi. ISRO in partnership with ROSCOSMOS launched its first satellite aryabhatta and after the first failed launch in 1979 again in 1980 became the 6th country in the world to launch SLV rockets in space [2].

India launched many launch vehicles which includes SLV, PSLV & GSLV. The space programme of India is focused on peaceful purposes and that success can be seen in India being the largest collection of remote sensing satellites which helped the Indian government to help its farmers and increase agriculture capacity and also avoid any natural disasters and relief programmes. India didn't stop there but developed its own GPS named IRNSS. Many satellites help the military in wars against pakistan and help keep expansionism of china in check. ISRO launched chandrayaan missions and mangalyan was launched even before china and we have been successful in our very first attempt. ISRO is the most respected organisation of India when it comes to space exploration because of its cost efficiency and success rate of launches. The organisation helped India in its growth and always reminds the people of the nation of their potential. The long term vision includes, man missions to the moon, reusable rockets, sending missions to planets like Jupiter and its own international space station. DOS opened a path for private players in space and it will boost India's capacity in the space race [2].

The future of space exploration is very promising as NASA, ROSCOSMOS, JAXA, ISRO, CNSA, CSA ASC & ESA the major space agencies are all heading in the right direction. From deep space exploration to understanding what lies there to building bases on mars to start human missions for researching as space has a lot of minerals which can be extracted and used on earth. To meet all these future perspectives we need rockets that are faster, hold more payload, more efficient in terms of fuel consumption so that it can go to Jupiter or beyond and carry payloads. We need better cooling techniques to safeguard rocket materials, better heat transfer and fuel efficiency. A lot of research is going on materials, cooling techniques, and combinations of best propulsive fuels. In the next headings we will talk about cooling techniques in rocket engines.

1.3. Rocket Engine Cooling

Temperature normally at the sun surface is 5800 k and normally at the throat of nozzle it is more than half of what sun surface is and for that we need a better cooling technique as the temperature is more than enough to melt any material and change its properties which could sabotage the missions of the rocket. The several techniques are as follows : Regenerative cooling, Ablative cooling, Film cooling, Radiative cooling, Mixture of cooling techniques, Hint sink cooling.

1.3.1 Regenerative Cooling

It is used generally in cryogenic liquid propellant engines which produce higher thrust. In this technique to cool the engine propellants passed through combustion chamber understand it like heat exchanger the outer surface is colder due to cryogenic propellants and inside is hot due to combustion, propellants passed through tubes and takes heat and when it is preheated passed directly to main combustion chamber. As we all want to increase heat transfer from combustion walls to propellants, we need to increase surface area for effective heat transfer so many experiments were done using materials of high heat transfer coefficients and also corrugated. Velocity of flow and thermal conductivity of engine materials also increases the effectiveness of regenerative cooling. The mechanical constraints is that to maintain the flow, the pressure in tube side must be higher than the combustion chamber pressure as natural flow happens only from higher to lower side. There is a possibility that leak chances are low but can't be ignored due to heavy pressure but even if there is leak it turns out to be advantageous for cooling as if leak starts acting like film cooling then combination of regenerative and film cooling will take place [3].

Rocket	**Fuel**	**Oxidiser**	**Thrust [N]**	**Chamber Pressure [bar]**	**Country**
Vikas	N2O4	MON3	260,000	70	India
PSLV - C52	UH25	MMH	220,000	60	India
PSLV - C50	Kerosene	LOX	240,000	50	India
GSLV MK 111	N2O4	MON3	196,500	75	India
PSLV - C51	UH25	MMH	230,000	55	India

Table 1.1 Rockets cooled with techniques of Regenerative cooling

Showing below the Cross sectional view of regenerative cooling engine, coolant flow is highest at the throat of the nozzle to increase heat transfer to maximum as that region is very critical.

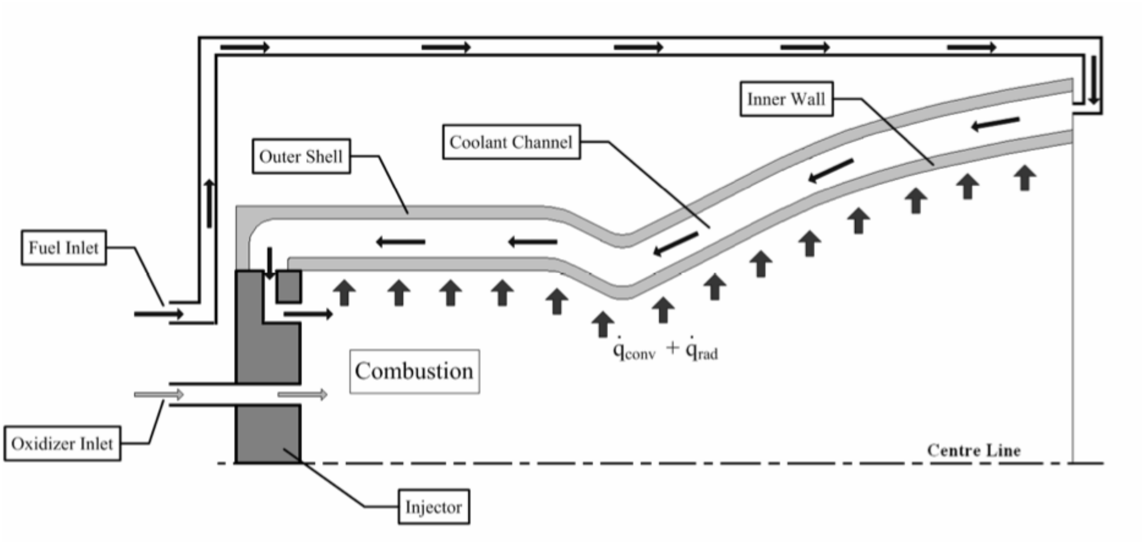


Fig 1.1 Schematic diagram of thrust chamber in axial direction [3]

1.3.2 Ablative Cooling

Ablative cooling is a technique which utilises latent heat of evaporation and predominant chemical reaction. It is the simplest and cheapest method for cooling rocket engines. Ablative materials in the form of liners are attached to the combustion chamber and they act as insulators as the thermal conductivity of ablative materials is very low. When combustion gases flow they are having high temperature and they start vaporising the liners of ablative materials the material will carry heat with it. For example in the Apollo lunar rocket engine carbon composites were used as ablative liners. This is very effective as it is light weight, cheap with no moving parts which means no wear and tear and when temperature in the combustion chamber is very high it starts vaporising layer by layer and takes away heat with it.

However the biggest drawback is that it sometimes damages and vaporises engine materials also which makes it unsuitable for reusing so it is not highly preferred over regenerative and film cooling techniques. So while designing, more focus is done on thickness to be suitable as some materials of boundary of liners and engine inner surface could vaporise so that it can be compensated.

Rockets	**Fuel**	**Engine**
Spacex Falcon 1	UH25/MMH	Merlin 1A
United alliance	LOX/Kerosene	Delta 1v
United alliance	Hydrolox	Rs 68 A
United alliance	N204	Rs 68

Table 1.2 Rocket engines which had ablative chambers for cooling techniques.

1.3.3 Film Cooling

In film cooling the cooling techniques are simple where fuel is passed into the combustion chamber through porous holes or slots cut into. The fuel impregnates the chamber into vapor form then it forms a protective layer over the surface so that walls of engines can be protected and cooled. Thermal Stress is significantly reduced making the rockets reusable for future missions. To increase effectiveness it is always better to use both film cooling and regenerative cooling or combination of film cooling with some other cooling techniques. The film cooling is also divided into gaseous film cooling, liquid film cooling and combination of both liquid and gaseous film cooling techniques [4].

Rocket Engine	Cooling techniques	Country
F1	Liquid Film Cooling	USA
J2	Liquid Film Cooling	USA
LE5	Liquid Film Cooling	Japan
Vulcain 2	Gas-Liquid Film Cooling	EADS Astrium
RS 27	Gaseous Film Cooling	ESA
RD 171	Gaseous Film Cooling	Russia
RD 180	Gaseous Film Cooling	Russia

Table 1.3 Rocket engines which used film cooling or combination of film cooling..

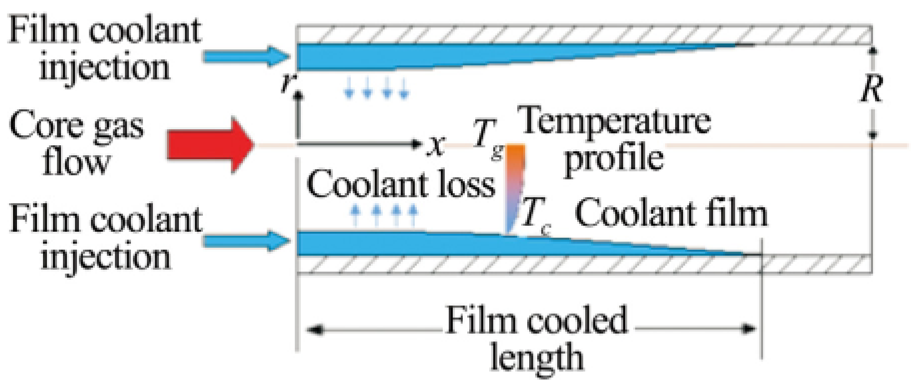


Figure 1.2 Rocket engines diagram of chamber for film cooling [4].

Factors which affects film cooling are as follows :-

- Radiation effects on rocket surfaces.
- Pressure, temperature and heat flux.
- Highly accelerated propellants flow.
- Two phase liquid flow conditions.

- Unsteady and turbulent flows.
- Density gradient & compressibility effects.

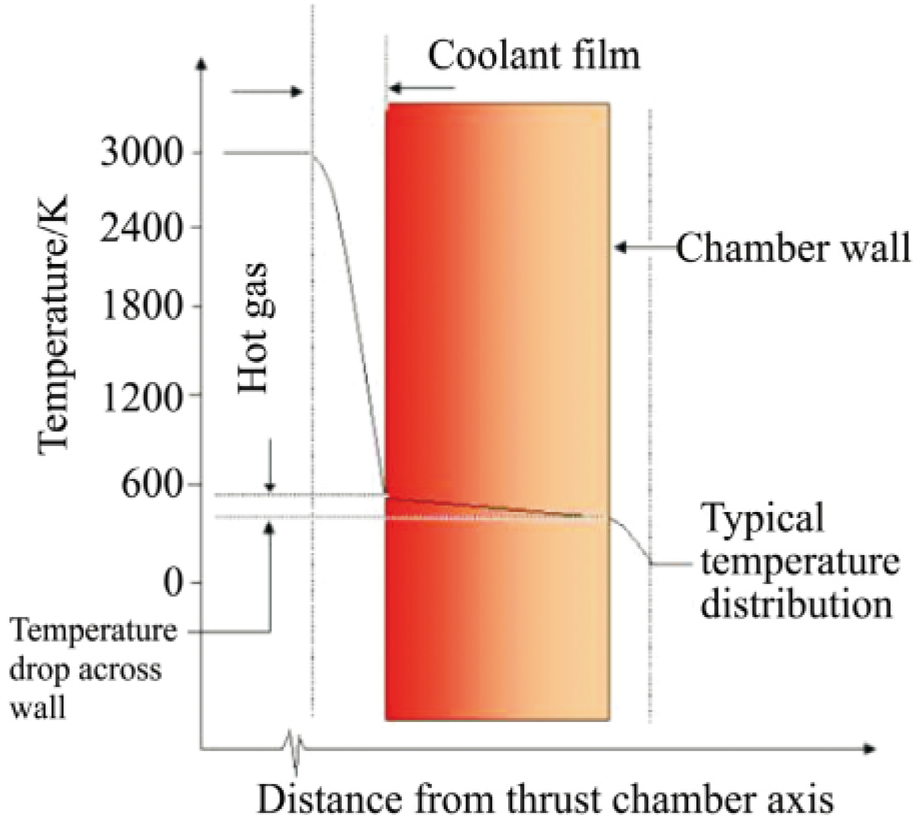


Figure 1.3 Schematics of chamber temperature profile [4].

In most missiles and nuclear warheads the cooling techniques used is gaseous film cooling instead of film cooling as in many experimental research carried out it was found that multi slots was possible in gaseous propellants mainly nitrogen and propane and high energy with minimal erosion was possible. From the nozzle wall it has lower heat load. External shock wave, coolant mach number, turbulence in mainstream, injector geometry & blowing ratio effects gas cooling in rockets. These parameters are mostly taken care of while designing cooling techniques.

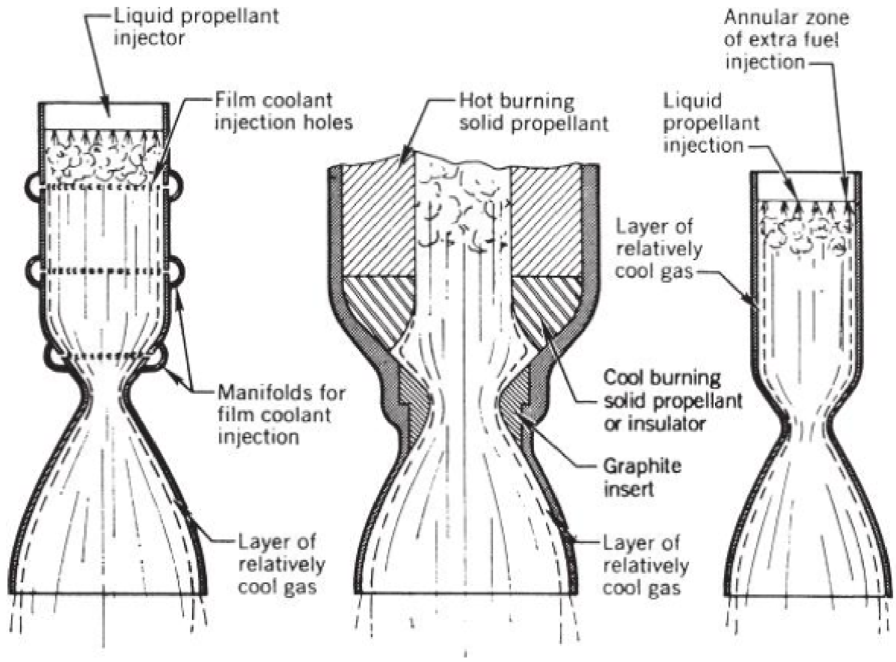


Figure 1.4 Schematics of engine chamber showing film protection layer [3].

It is shown in figure how fuel is injected through multiple injector holes and it forms a layer with low velocity and in large amounts it quickly covers the surface . Solid propellants near throat regions get cooled extra and cannot burn properly leading to reduction of impulse in rocket engines. That's the reason it is more suitable to use film cooling with a combination of other techniques [3].

1.3.4 Radiation Cooling

In radiation cooling techniques the converging side is much smaller than diverging side means the nozzle area ratio which is diverging area upon converging area is between 250 to

375 and converging side surface is coated with rhenium with iridium till throat section and upper nozzle exit section surface is coated with niobium and disilicate and lower nozzle cross section surface with titanium. In the diverging section radiation has almost negligible effects as compared to the converging side where temperature at the throat is similar to peak temperature of converging side. In general while heat transfer through radiation it increases the temperature of the surface up to 20-30 %.

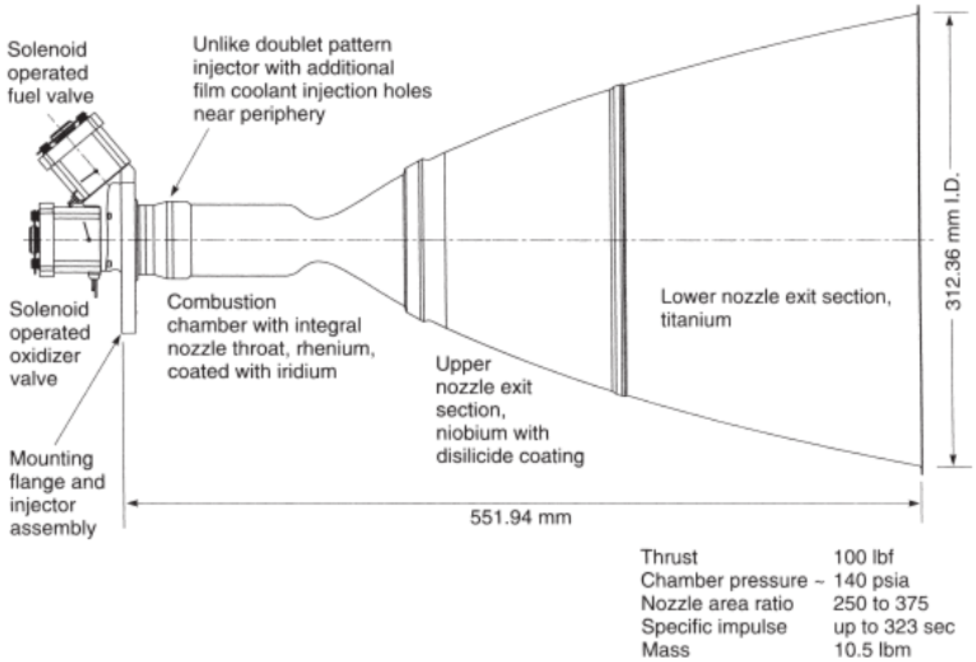


Figure 1.5 Schematics of engine chamber showing radiation cooling [3].

Chapter 2
LITERATURE REVIEW

2.1: Literature Reviews

Two prominent scientists of Japan society for aeronautical and space sciences in their paper of radiative heating in the combustion chamber of liquid propellant rocket engines found the relationship between length of cylindrical section of thrust chamber and its diameter. The key finding was that radiative heat flux upon total heat flux varies between 25-35 %. Fuel used was O2 as oxidiser and methane along with ethanol and hydrogen [5].

In the msc thesis submitted by Mustafa emre boysal performed simulation modelling on a liquid propellant engine running on fuel of LOX/Kerosene with combustion chamber pressure of 60 bar and thrust produced of 300 kN. Established relationship with the effect of number of rectangular cooling channels and geometry on efficiency of cooling. Results produced that cooling efficiency at 4x2 mm2 cross section area of channel with 110 cooling channels is best and also pressure drop is higher. If The Increasing Number Of Cooling Channels Is Up To 40-45 %, Pressure Drop Is Approximately 47-53 %. One of the major achievements is increasing aspect Ratio which in turn increases cooling efficiency and drop in pressure also increases. [6]

In the aiaa conference paper Alexander W. Miranda and Mohammad H. Naraghi conducted a simulation Cfd Modeling at film layer with variation in composition and Flow Rate, it was also published that in film layer purest hydrogen produced maximum film cooling but it is minimal with injection of mixture of rich hydrogen and oxygen also when mass flow rate increased simultaneously heat flux decreased. [7]

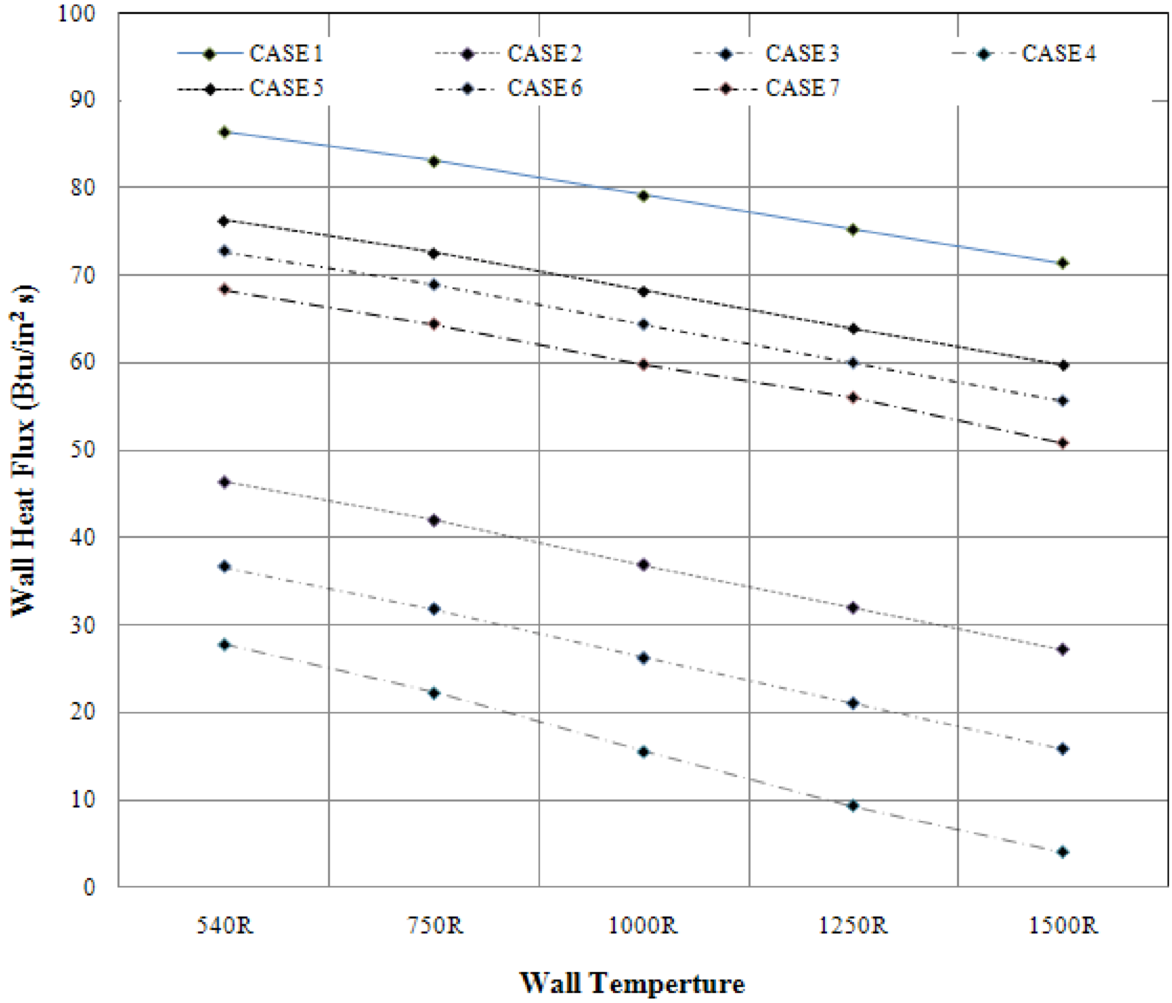
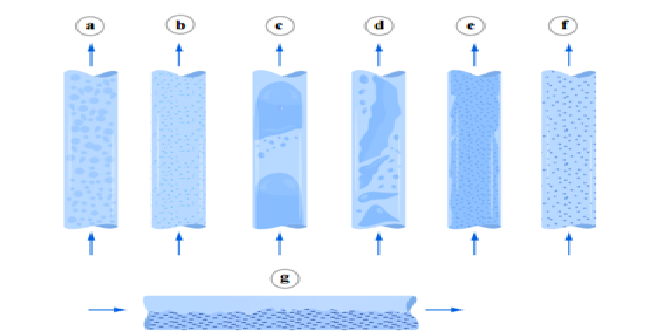


Figure 2.1 Diagram showing maximum wall heat flux for various cases as functions of wall temperature [7].

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 [8].

Flow Quality	Flow Rate	Flow Regime
Low	Low and Intermediate	Bubbly
	High	Dispersed Bubbly
Intermediate	Low and Intermediate	Plug/Slug
	High	Churn
High	High	Annular
	High (post dry-out)	Mist



Typical configuration of (a) bubbly flow, (b) dispersed bubbly (i.e. fine bubbles dispersed in the continuous liquid phase), (c) plug/slug flow, (d) churn flow, (e) annular flow, (f) mist flow (i.e. fine droplets dispersed in the continuous vapor phase) and (g) stratified flow. Note: mist flow is possible only in a heated channel; stratified flow is possible only in a horizontal channel.

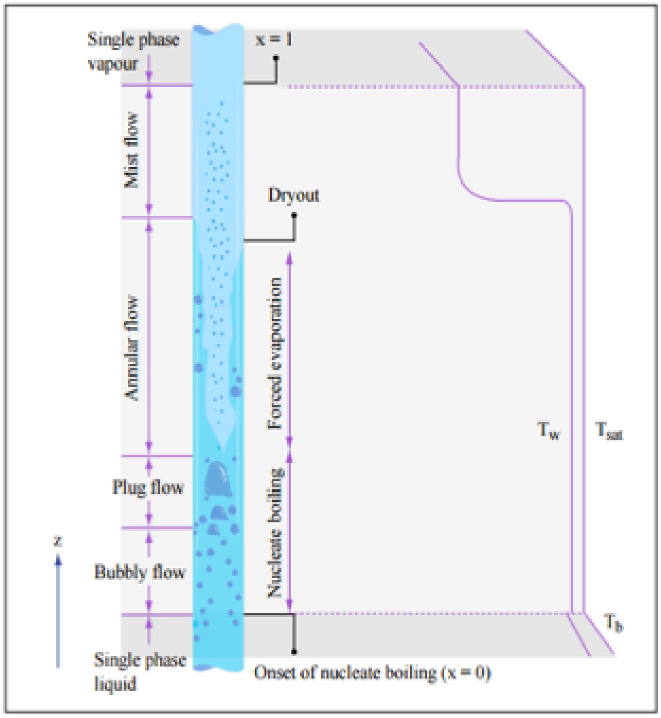


Image by MIT OpenCourseWare.

Figure 2.2 Diagram showing two phase flow & flow boiling regime [9].

A Adami, et al. in a paper published in an international journal of aerospace engineering that they replaced high heat flux costly materials by composites or steel, an algorithm they wrote actually minimises the wall temperature, the minimum total mass comes out through an equation between propellant minimum mass and the dry minimum mass. lower Isp Increases of having a total mass minimalistic and total mass heavier. Increase in leads to having a higher total mass for their algorithm as per their perspective could be brought in and can be better used for total impulse and thrust level required at all levels. [10]

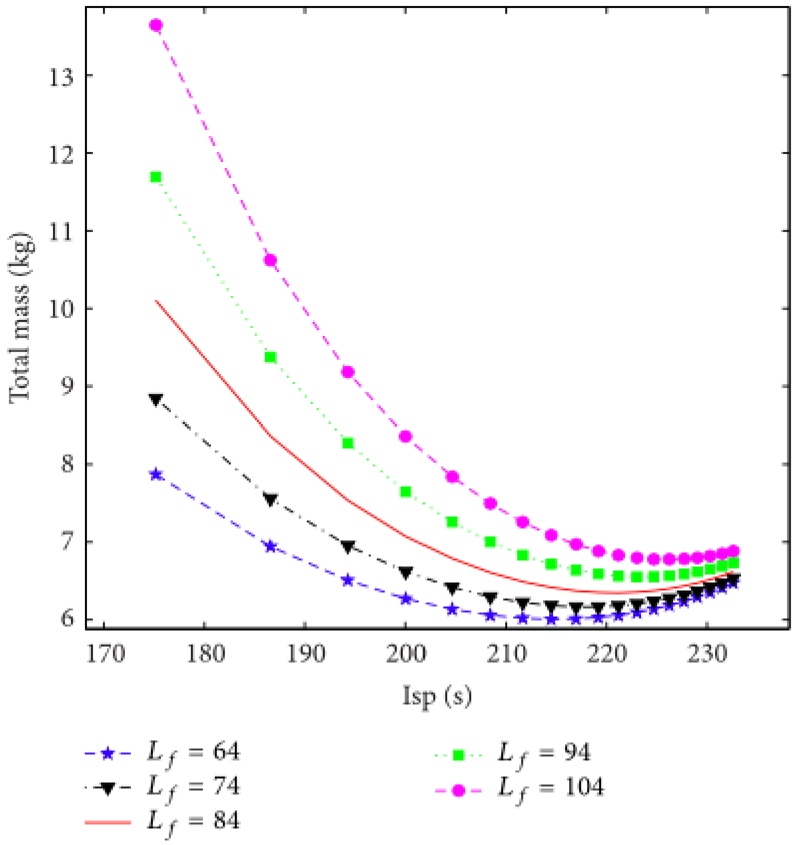


Figure 2.3 Diagram showing variation with respect to Lf in total mass [10].

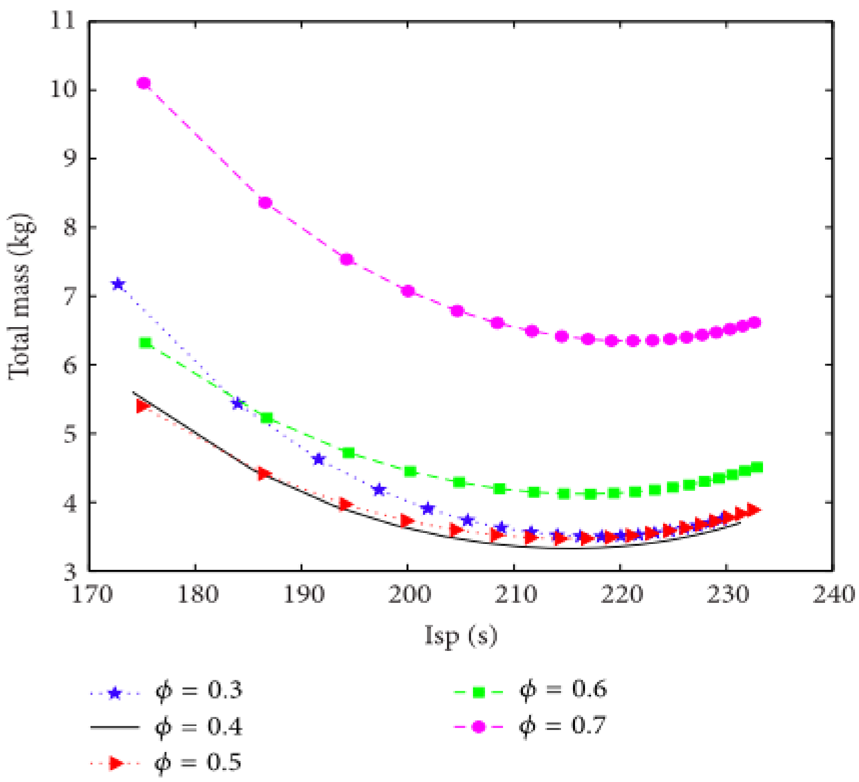


Figure 2.4 Diagram showing variation with respect to percentage NH3 in total mass [10].

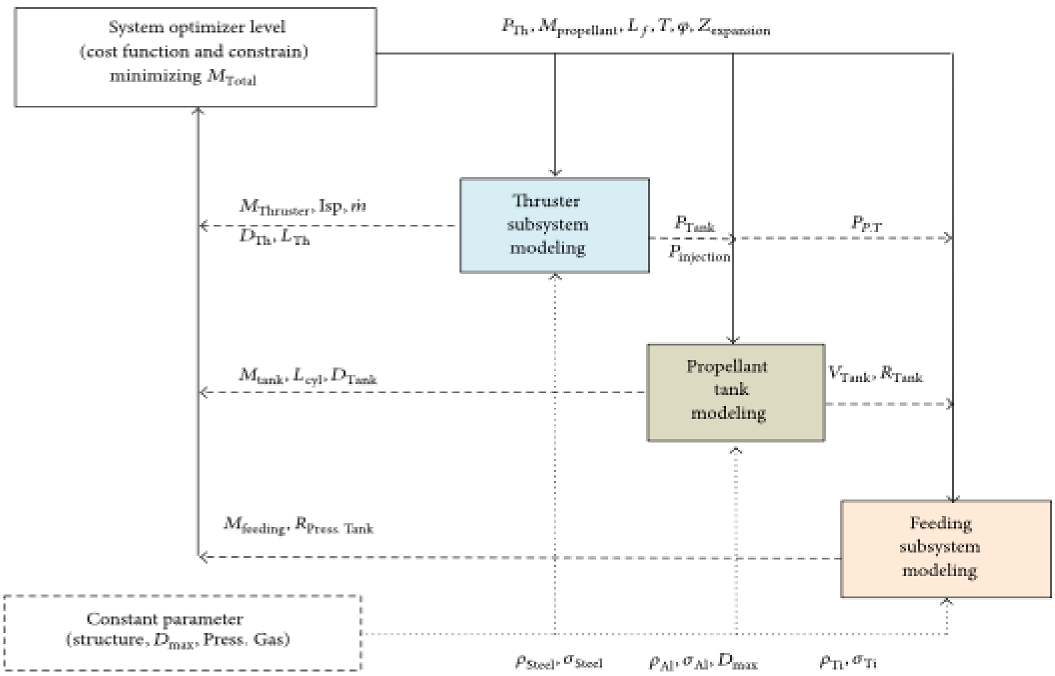


Figure 2.5 Diagram showing design algorithm for optimised hydrazine propulsion system [10].

In the paper published in the international journal of thermal sciences Mo Bai evaluated numerically and simulated open cell foam heat transfer and pressure drop which is a critical condition sometimes it is too high and is not suitable for optimum heat transfer and performance. Diamond unit cell computational fluid dynamics in fluent was presented, the call really has unit cell microstructure of aluminium tetrakaidecahedron. As we know in engineering materials if porosity is high then surface area increases which ultimately increases heat transfer rate as it has high surface area to volume ratio [11]

Two NASA scientists J. Stoll And J. Strau performed many experiments for heat transfer analysis and film cooling analysis found out that there was turbulence mixing of gases when air passed is highly compressed also with their experimentally calculated value was then compared with theoretical data to find out how much there was deviation then continued on changing pressure and velocity of compressed air [12].

In lewis research centre of NASA two young and dynamic scientists C.J. Marek and R.R. Tacina performed an experiment to check how effective film cooling is inversely related to free stream turbulence intensity and also found that when increase in turbulence Intensity was 7 to 35 % simultaneously it was found 50 % decrement in film Cooling [13].

Blockage area, percent	Hot-gas flow rate, kg/sec	Hot-gas temperature, K	Cooling flow, kg/sec	Cooling flow, percent	Cooling air temperature, K	Downstream distance, cm								
						0.95	3.50	6.04	8.58	11.12	13.66	16.20	18.74	21.28
						Test-plate temperature, K								
0	1.45	595	0	0	529	557	573	577	578	579	579	579	577	571
	1.44	595	.039	2.62	338	355	385	426	460	482	496	506	511	509
	1.46	594	.079	5.16	315	324	335	353	379	404	424	440	451	453
	1.45	592	.119	7.62	306	314	326	341	360	379	396	411	422	423
	1.45	594	.161	9.97	298	306	318	332	346	358	370	381	389	382
52	1.45	594	0.039	2.60	345	376	436	479	504	519	527	533	535	531
	1.46	588	.077	5.04	318	328	347	379	413	438	455	468	476	475
	1.46	587	.120	7.61	309	317	330	350	378	402	422	436	446	445
	1.42	593	.156	9.93	299	306	317	332	349	365	380	392	402	403
72	1.45	596	0	0	575	584	590	590	590	590	589	---	587	580
	1.46	589	.036	2.42	358	400	467	503	523	534	541	---	545	540
	1.47	581	.076	4.95	329	339	365	406	441	464	479	---	495	494
	1.45	589	.118	7.54	319	325	340	365	398	424	443	---	465	465
	1.46	590	.163	10.00	311	315	324	341	364	387	405	---	429	432
90	1.48	584	0	0	592	592	593	592	592	591	591	590	587	579
	1.48	586	.044	2.87	354	401	466	507	531	545	553	557	558	554
	1.49	585	.088	5.57	329	352	391	434	470	493	508	518	523	519
	1.48	586	.124	7.77	315	324	345	379	417	445	466	480	489	488
	1.48	590	.163	9.89	308	315	331	354	385	412	433	450	460	458

Table 2.1 Data observed in experiment conducted [13].

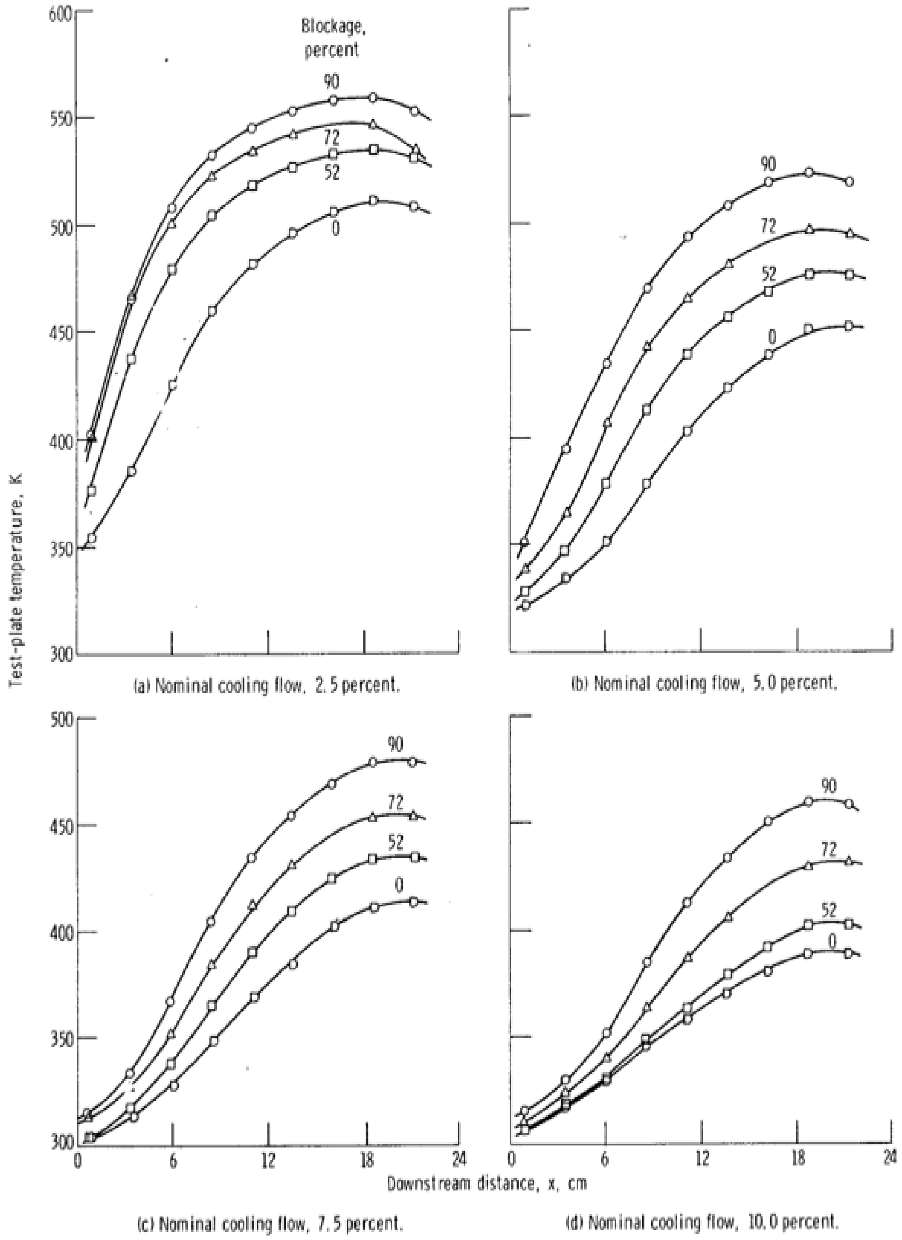


Figure 2.6 Diagram showing wall temperature in y axis and downstream distance and temperature of gas is 500 k [13].

Chuan Fan, et al. conducted experimentation on supersonic combustors to check dependence of film cooling effectiveness on injection temperature, injection angle and mass flux. They found out that film cooling is more effective with an increase in coolant mass flux [14].

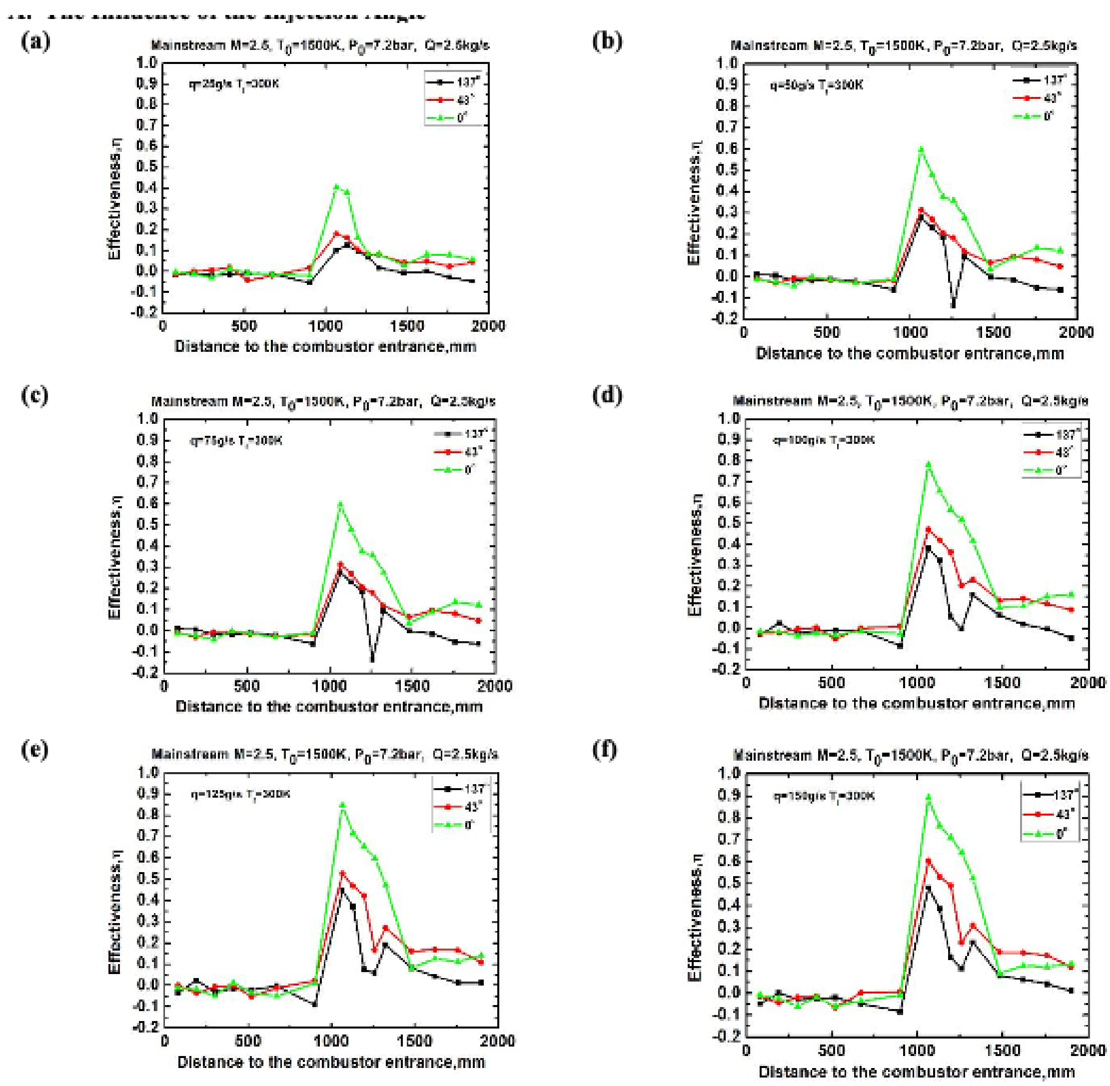


Figure 2.7 Diagram showing film cooling effectiveness at different mass flux, injection angle and injection temperature [14].

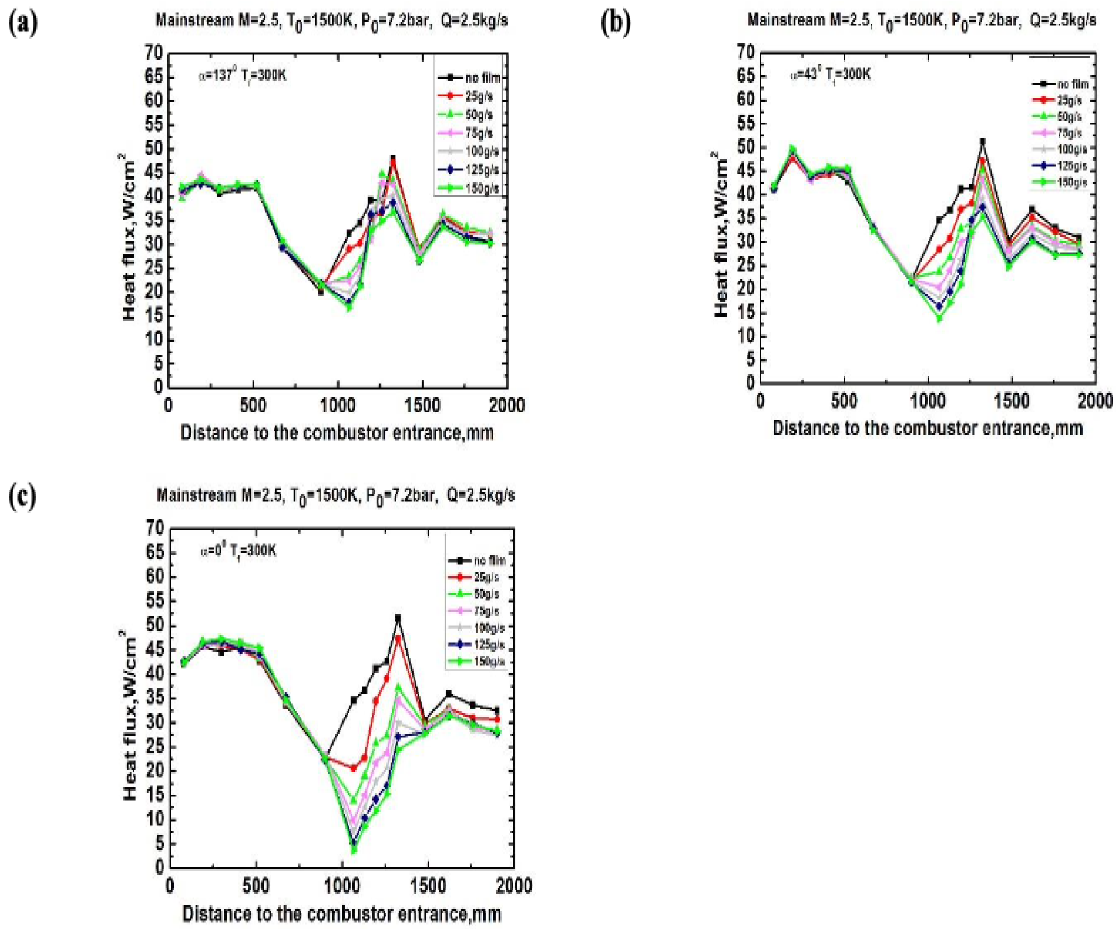


Figure 2.8 Diagram showing film cooling effectiveness at distance from combustion entrance [14].

K.A. Juhany, et al. published a research paper in the journal of thermophysics and studied the effect of injection of mach number and temperature on supersonic film cooling temperature range was 210-240 k and mach number 2.2 to 2.6 so the result when mach number was increased then film cooling effectiveness rate also increased heat transfer. The film cooling technique is more effective at 2.4 mach number and temperature was 220 kelvin [15].

Maximillian. Hombsch and Herbert Olivier published a research paper journal of spacecraft and rockets and conducted Experiments in shock wave laboratory in RWTH aachen film cooling of supersonic flat flows for boundary region for laminar and turbulent region by flowing coolant through different plate surface geometries, Coolant mass flow rate and injection angle was varied by them and film cooling effectiveness and efficiency curve was plotted and compared with theoretical say empirical formulae [16].

Transpirational cooling has been modelled by Valentina Koonig, Michael Rom, and Siegfried Muller for hot gas and poros medium coupling at interfaces in subsonic and supersonic two types of combustors and results has been compared and for two domains solver RANS has been solved and also porous medium has been solved one by one [17].

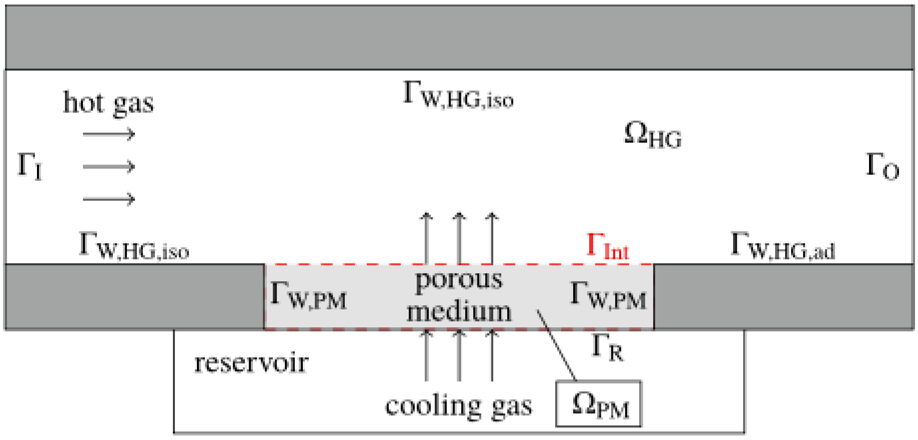


Figure 2.9 Diagram showing setup of transpirational cooling [17].

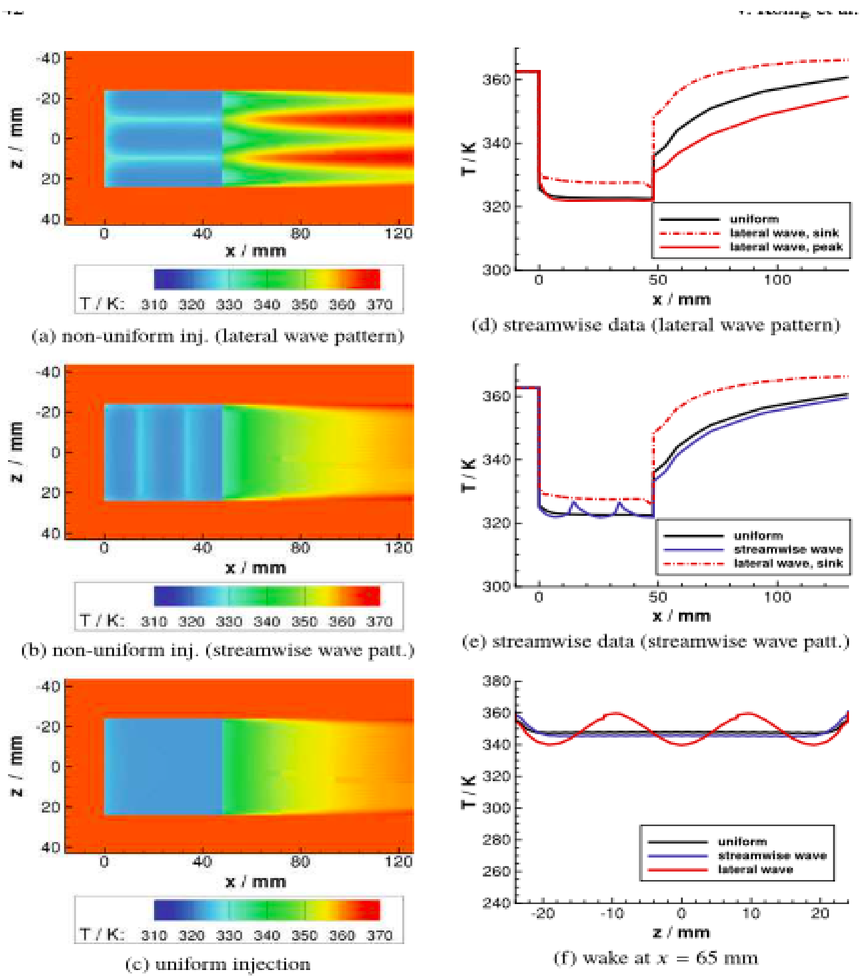


Figure 2.10 Diagram showing Non uniform and uniform flow of injection [17].

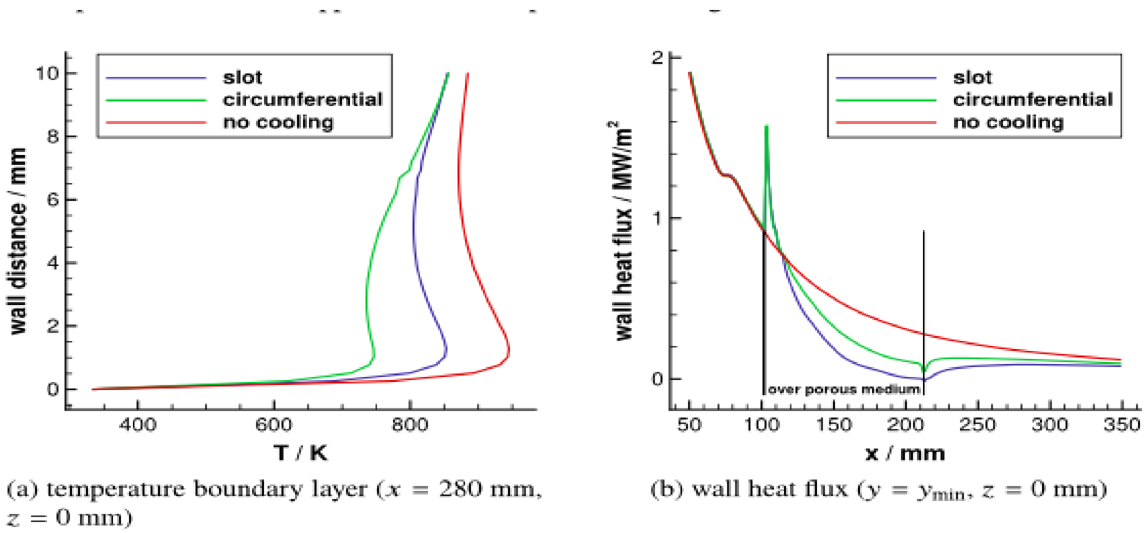
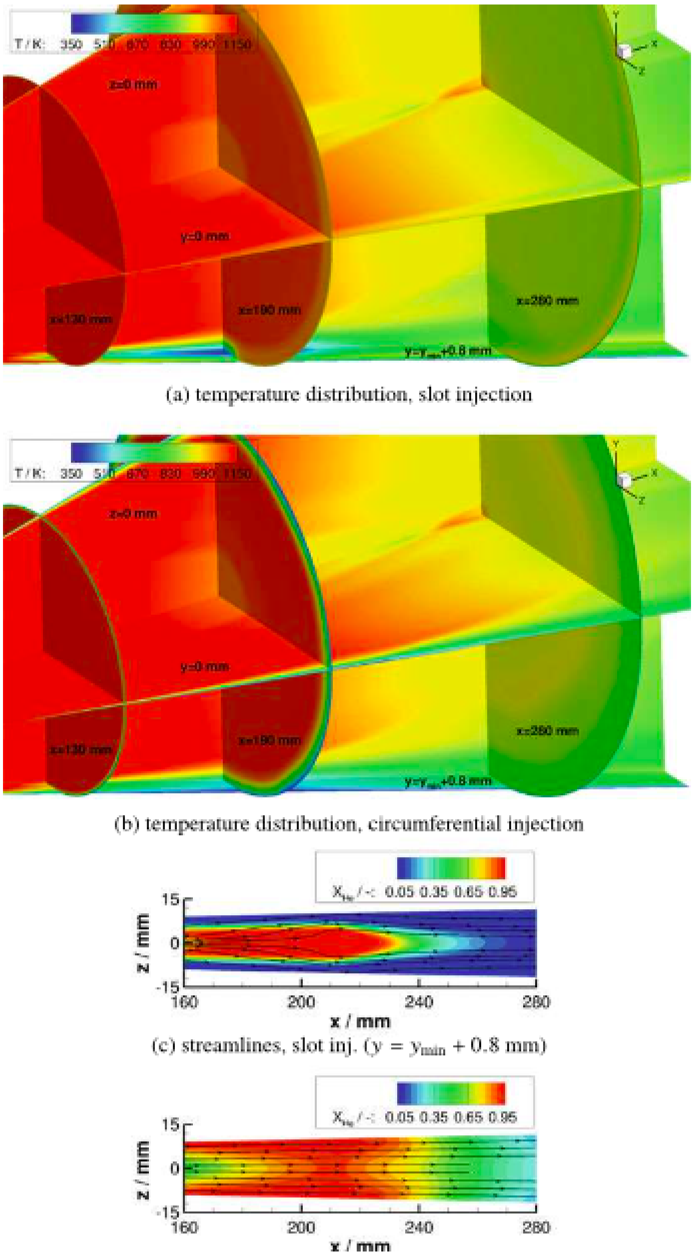


Figure 2.11 Wall heat flux and temperature boundary layer during circumferential injection [17].



(a) temperature distribution, slot injection

(b) temperature distribution, circumferential injection

(c) streamlines, slot inj. ($y = y_{min} + 0.8$ mm)

(d) streamlines, circ. inj. ($y = y_{min} + 0.8$ mm)

Figure 2.12 Diagram showing injection in nozzle flow in slots and circumferential [17].

Johannes M. F. Peter and Markus J. Kloker published a research paper in which they passed compressed gas in supersonic combustors at mach number of 3.2 at turbulent boundary layer also the gas used is helium and wall condition is adiabatic and turbulence modeling used is DNS and flow assumed is periodic in Z direction [18].

Table 2.2 Data observed in cooling condition of stream and blowing ratio [18].

Case	F	Geometry	$s^{\star}$ (mm)	$p_c^{\star}$ (Pa)	p_c/p_∞	Coolant exit condition
C-I	$0.3\overline{3}$	G01	0.6	15915	0.584	Overexpanded
C-II	0.59	G01	0.6	28000	1.000	Matched
C-III	$0.6\overline{6}$	G01	0.6	31830	1.168	(weakly) underexpanded
C-IIIa	$0.6\overline{6}$	G02	0.9	31830	1.168	(weakly) underexpanded
C-IV	1.00	G01	0.6	47745	1.752	Underexpanded
C-IVa	1.00	G02	0.9	47745	1.752	Underexpanded

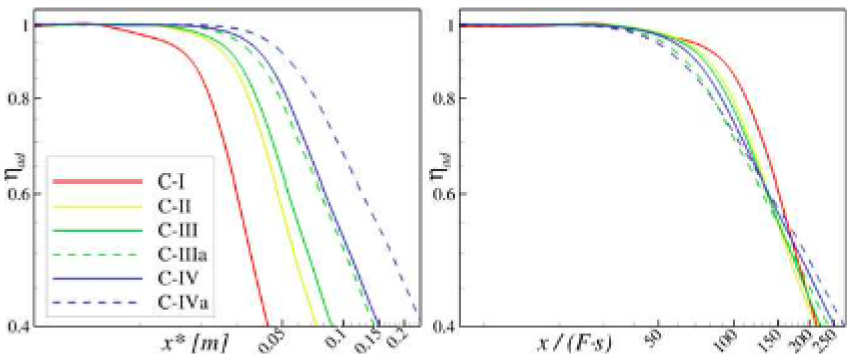


Figure 2.13 Comparison of mean cooling effectiveness η and for variation of the coolant mass flow rate [18].

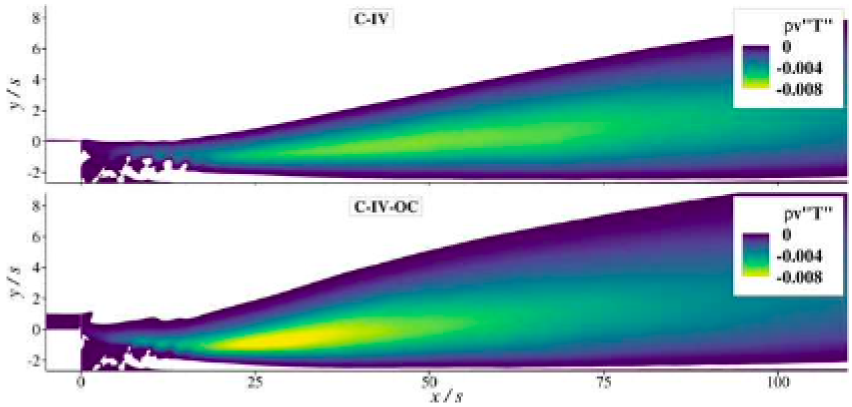


Figure 2.14 Diagram showing turbulence stress in contours [18].

Table 2.3 Investigated coolant Mach numbers with momentum ratio M [18].

Case	F	Geometry	s^* (mm)	p_c^* (Pa)	Ma_c	M
C-IIIa	$0.6\overline{6}$	G02	0.9	31830	1.80	0.221
C-IIIa-Ma	$0.6\overline{6}$	G02	0.9	22200	2.42	0.254
C-IV	1.00	G01	0.6	47745	1.80	0.331
C-IV-Ma	1.00	G01	0.6	32037	2.50	0.387

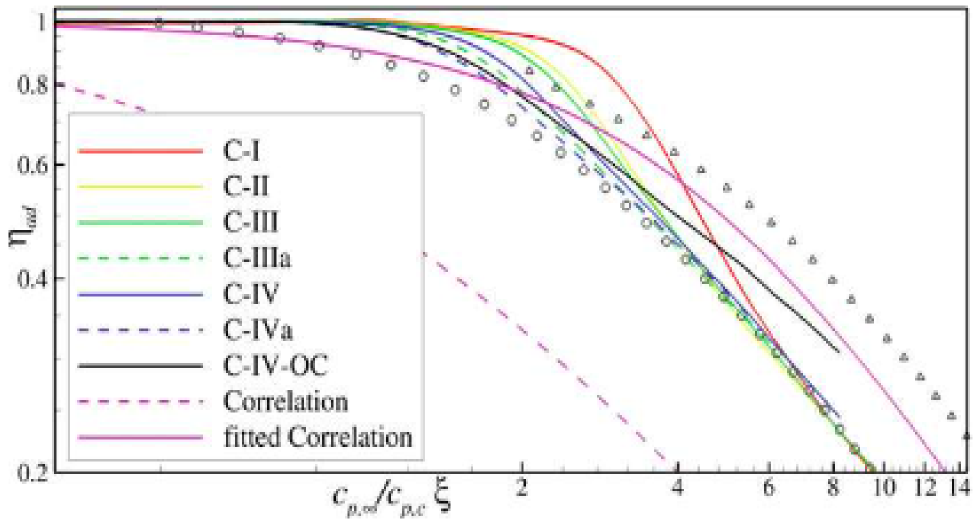


Figure 2.15 Diagram showing efficiency and correlating cooling data efficiency [18].

2.2: Objective of the present study

The study presented focussed on collective relevant information from the topic of current research. Literature review provides concrete information for further studies and helps improve cooling efficiency in author's research. Different combination cooling techniques provided authors a step to conduct research on spacex rocket engines and improve cooling of the merlin 1d vacuum engine used mostly in falcon 9. Cooling efficiency has been compared with coolant mass flow, different geometries, injection ratios and their correlation with cooling effectiveness. In this current research the author has simulated and tried to establish relation between pressure, velocity, turbulent kinetic energy, specific dissipation ratio and cooling with change in injection angle and geometry of nozzle.

Chapter 3

METHODOLOGY

3.1 Theoretical Hand Analysis Of Heat Transfer In Engine

Theoretical analysis has been performed and hand calculation for rocket equation, combustion analysis and heat transfer has been performed.

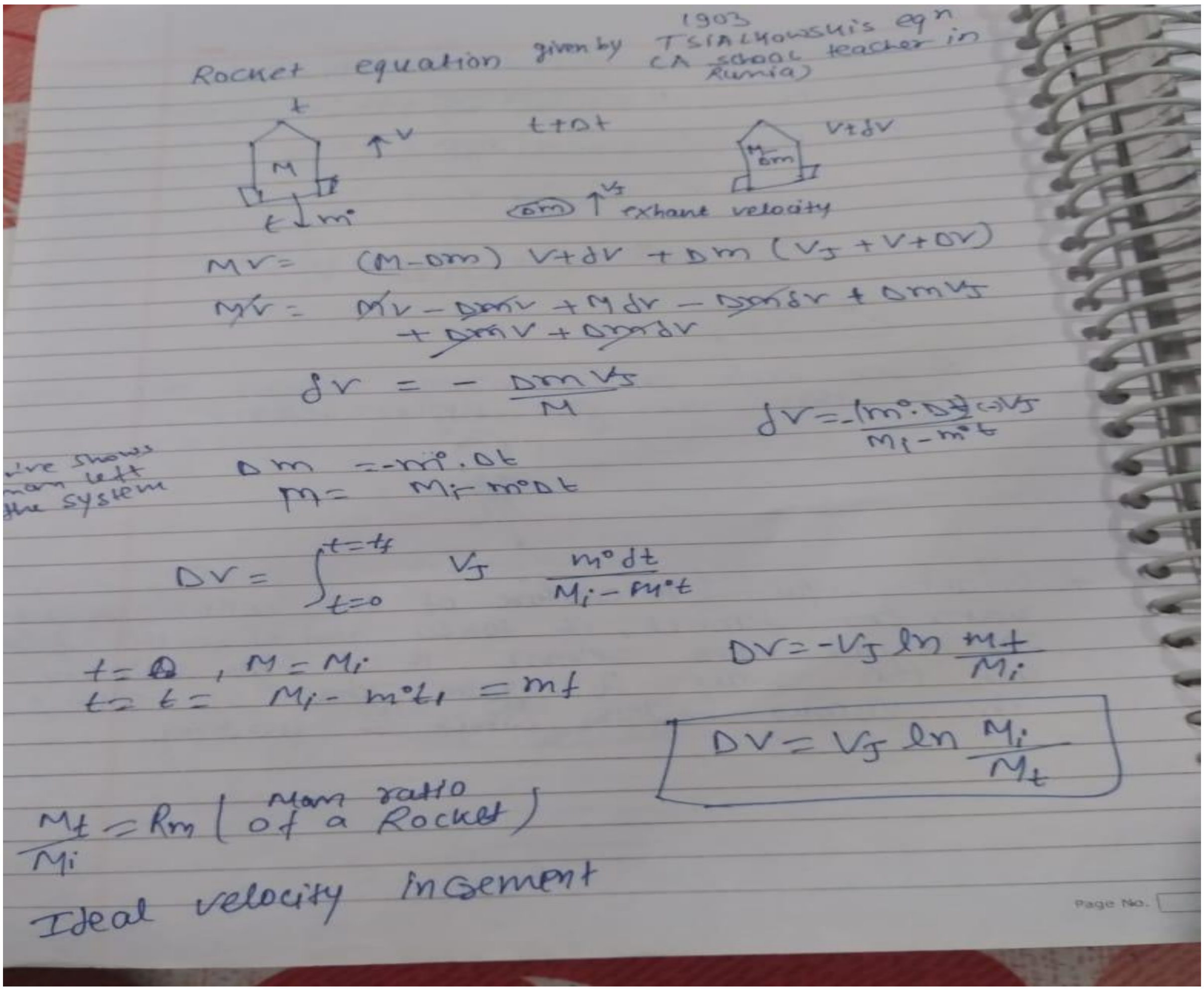
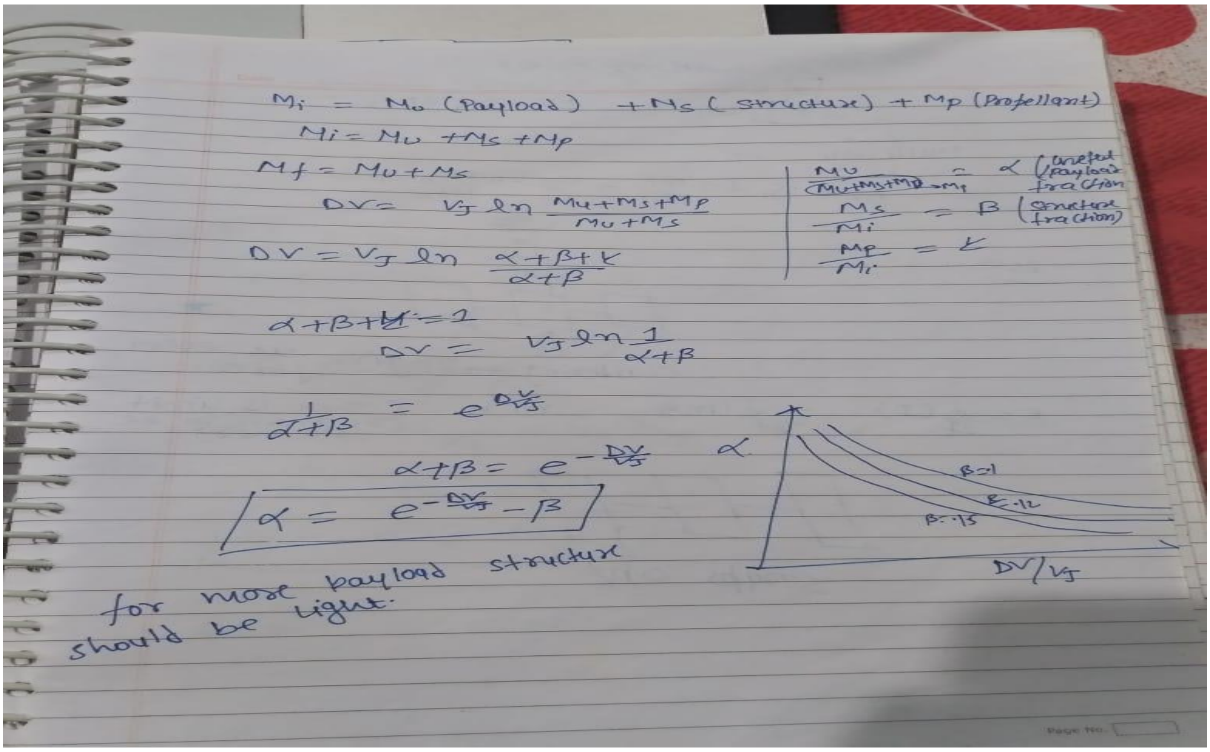


Figure 3.1 Hand calculation of rocket equation



$M_i = M_u\,(\text{Payload}) + M_s\,(\text{structure}) + M_p\,(\text{Propellant})$

$M_i = M_u + M_s + M_p$

$M_f = M_u + M_s$

$\Delta V = V_J \ln \frac{M_u + M_s + M_p}{M_u + M_s}$

$\frac{M_u}{M_u + M_s + M_p = M_i} = \alpha$ (useful payload fraction)

$\frac{M_s}{M_i} = \beta$ (structural fraction)

$\frac{M_p}{M_i} = \varepsilon$

$\Delta V = V_J \ln \frac{\alpha + \beta + \varepsilon}{\alpha + \beta}$

$\alpha + \beta + \varepsilon = 1$

$\Delta V = V_J \ln \frac{1}{\alpha + \beta}$

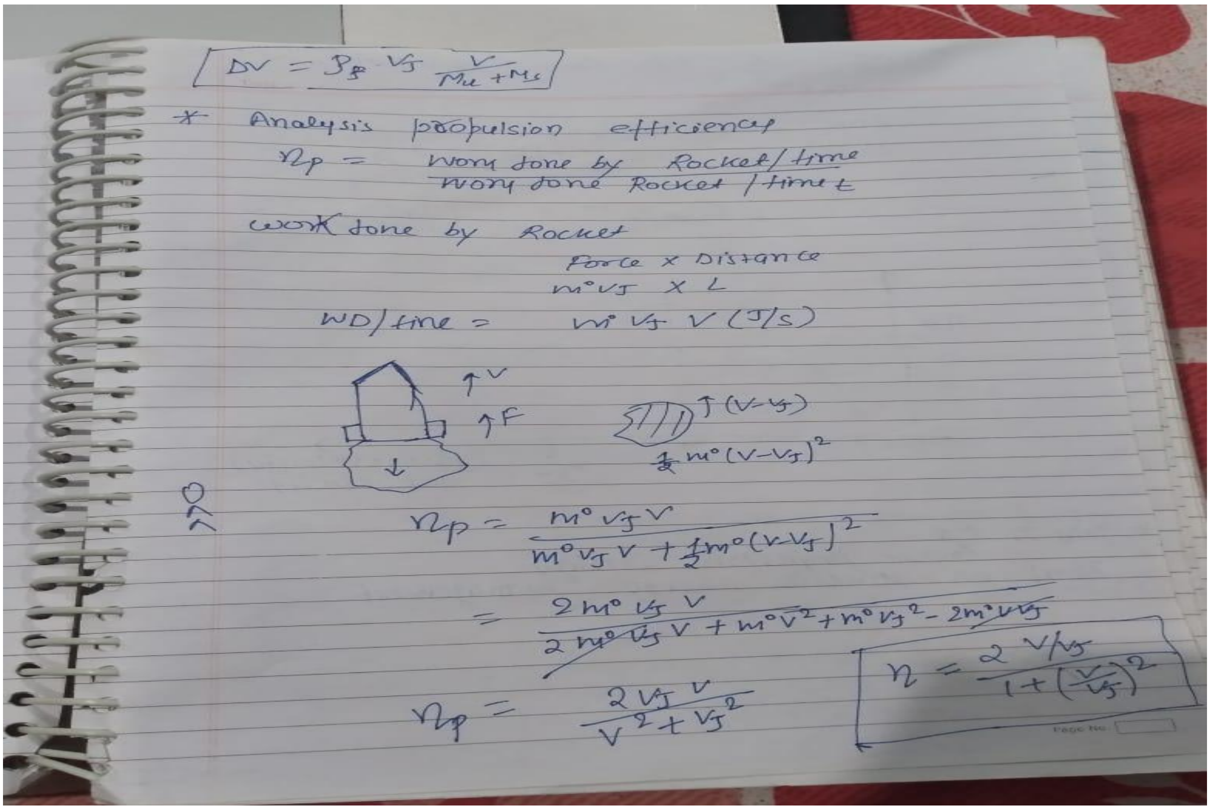
$\frac{1}{\alpha + \beta} = e^{\Delta V / V_J}$

$\alpha + \beta = e^{-\Delta V / V_J}$

$$\boxed{\alpha = e^{-\Delta V / V_J} - \beta}$$

for more payload structure should be light.

Figure 3.2 Hand calculation of payload, structure & propellant.



$$\boxed{\Delta V = \beta_p V_J \frac{V}{M_u + M_s}}$$

* Analysis propulsion efficiency

$\eta_p = \frac{\text{work done by Rocket / time}}{\text{work done Rocket / time } t}$

work done by Rocket

Force × Distance

$\dot{m} V_J \times L$

WD/time $= \dot{m} V_J V$ (J/s)

$\frac{1}{2}\dot{m}(V - V_J)^2$

$\eta_p = \frac{\dot{m} V_J V}{\dot{m} V_J V + \frac{1}{2}\dot{m}(V - V_J)^2}$

$= \frac{2\dot{m} V_J V}{2\dot{m} V_J V + \dot{m} V^2 + \dot{m} V_J^2 - 2\dot{m} V V_J}$

$\eta_p = \frac{2 V_J V}{V^2 + V_J^2}$

$$\boxed{\eta = \frac{2\, V/V_J}{1 + \left(\frac{V}{V_J}\right)^2}}$$

Figure 3.3 Hand calculation of propulsion efficiency

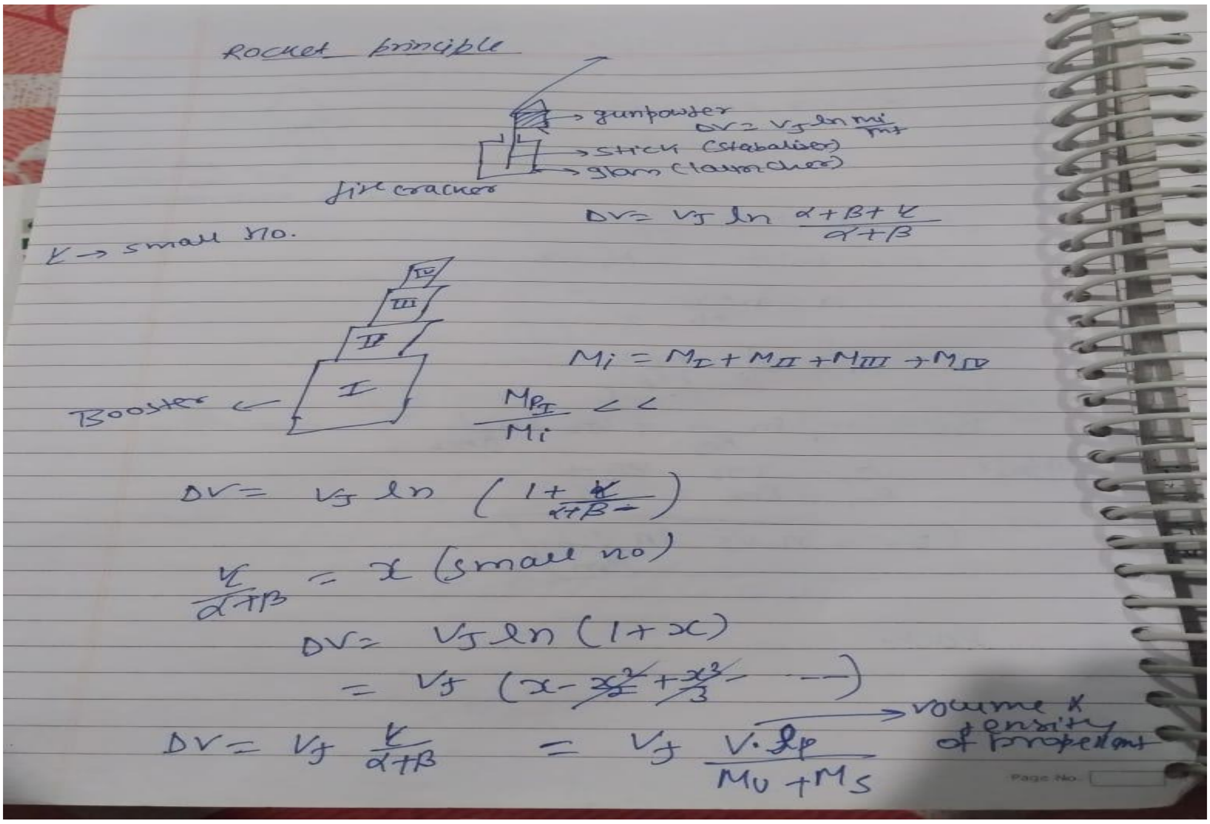


Figure 3.4 Hand calculation of heat transfer analysis in 4 stages of rocket

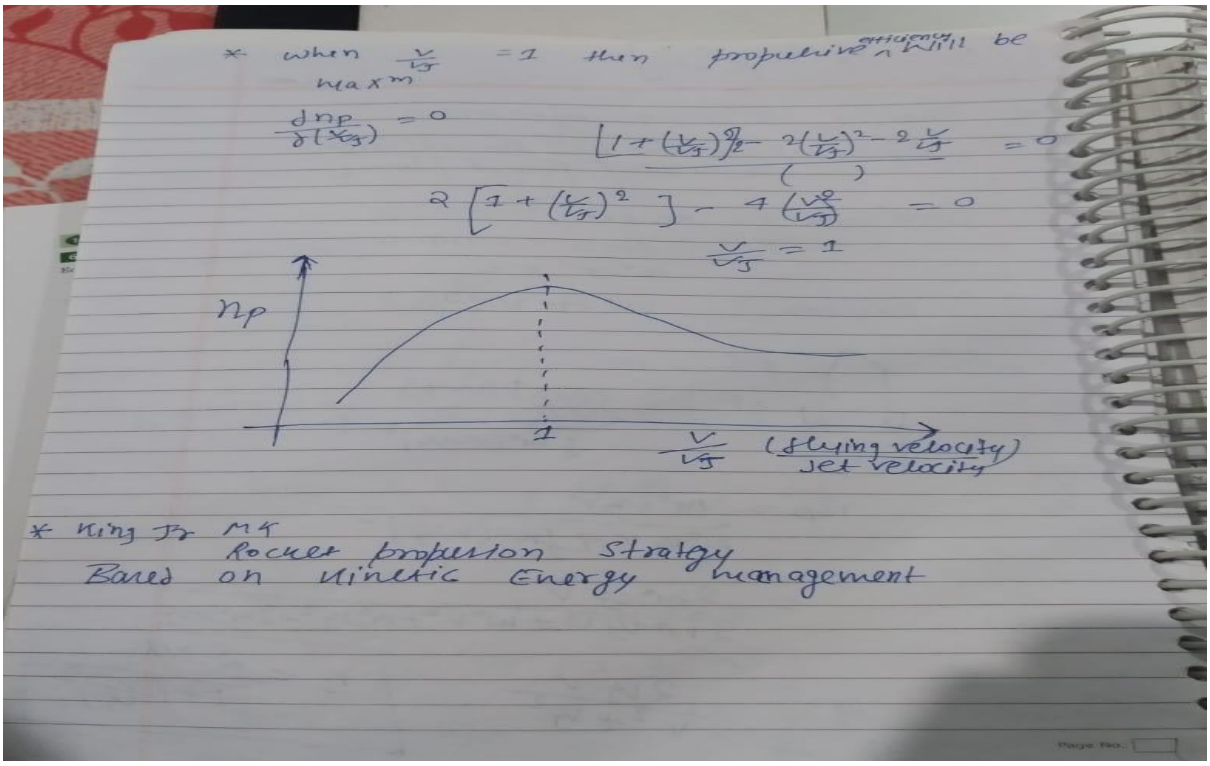


Figure 3.5 Hand calculation and variation curve of kinetic energy

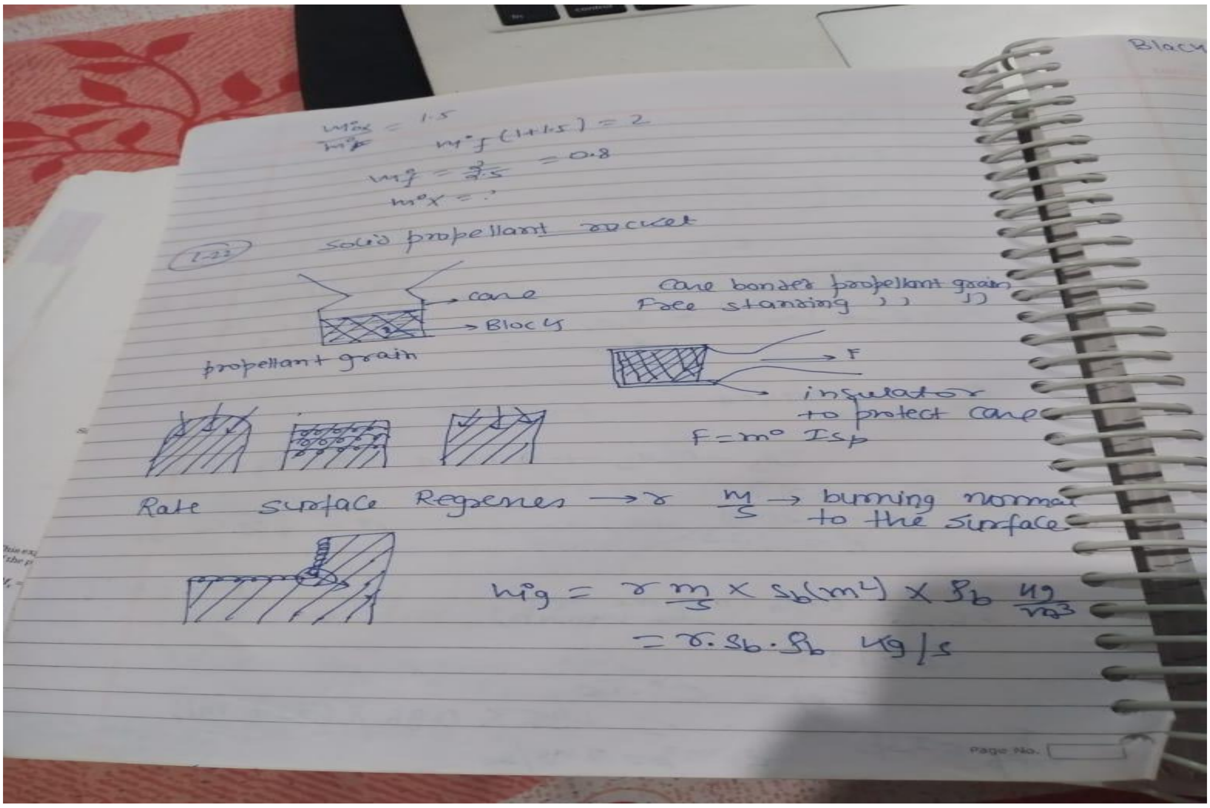


Figure 3.6 Surface regression of solid propellant rockets

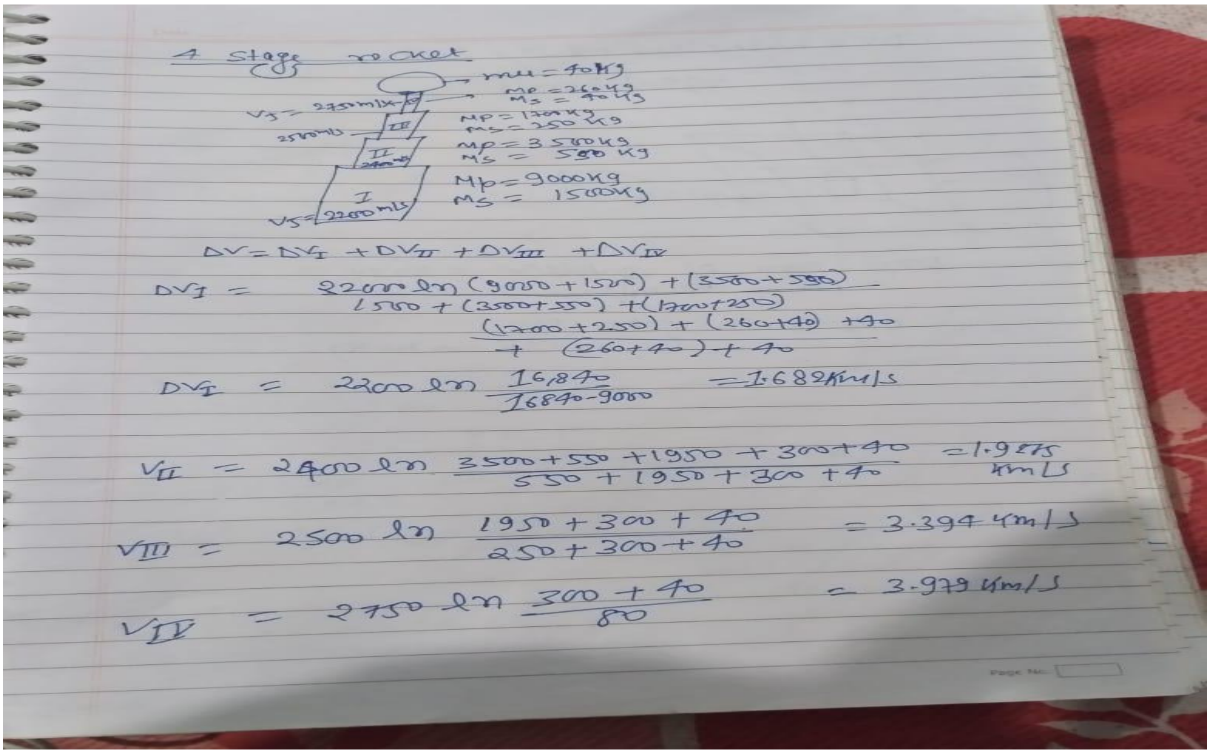


Figure 3.7 Hand calculation of propulsive speed of 4 stage booster rockets

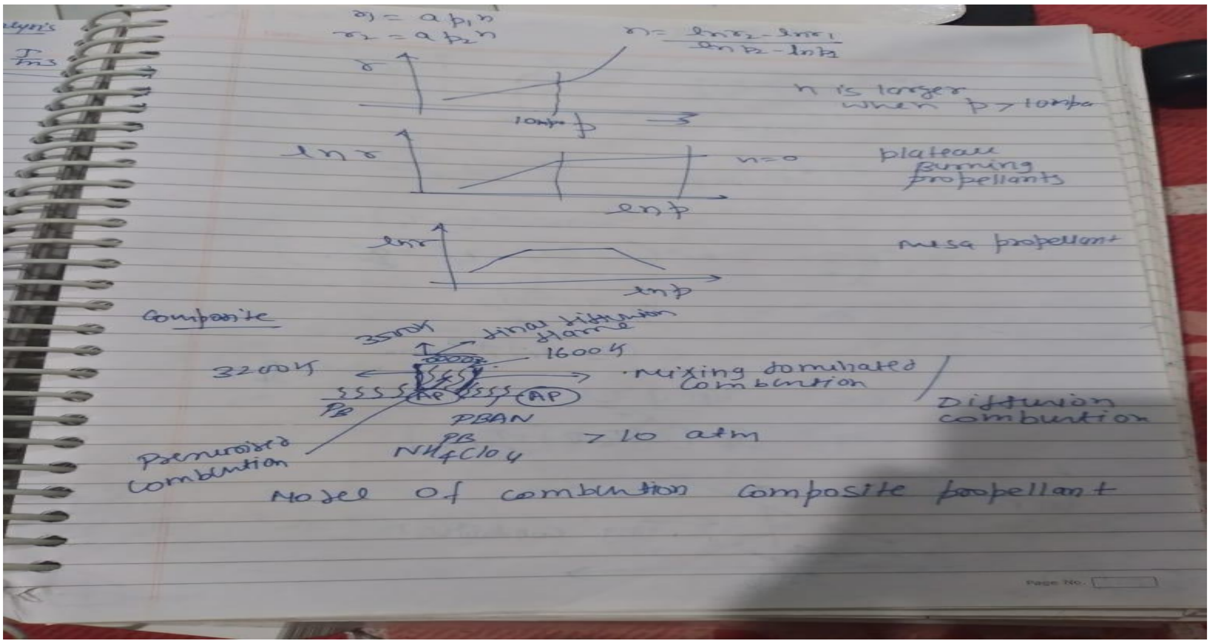


Figure 3.8 Characteristics curve plotted for composite propellant burning rate vs pressure.

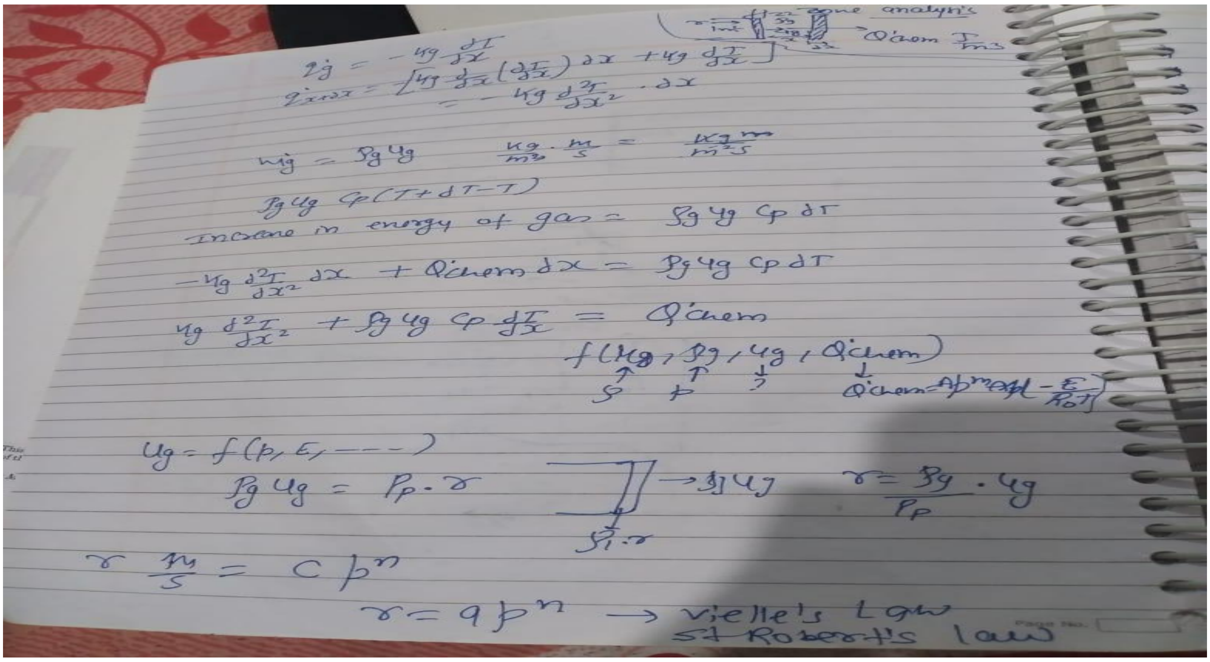


Figure 3.9 Hand calculation of gas energy according to st roberts law

Generally if chamber pressure is small, mass flow rate is small and Isp {Specific Impulse}is not much affected, gg cycle is more suited for low pressure engine and scc cycle is applicable for high pressure engine.

CHAPTER 4
RESULTS & DISCUSSION

Simulation Results

4.1 Nozzle design

A converging diverging Nozzle has been designed as cad modelling for heat transfer analysis in combustion chamber and in throat area and entire nozzle area. Inspiration has been taken from spacex merlin vacuum 1d engine. Nozzles are the most important part of all kinds of rockets whether solid propellant, liquid propellant and hybrid propellant rockets. Design of nozzle is very important as the small change in parameters could bring a lot of impact in heat transfer. The converging side has been taken 0.37 m and the length of the diverging section has been taken 0.63 m. The length of the converging side is smaller as compared to the diverging side to facilitate greater thrust in rockets. The throat has minimum radius because to increase mach number greater than 1 and room for diverging section. It is also the critical part of the nozzle during heat transfer analysis.

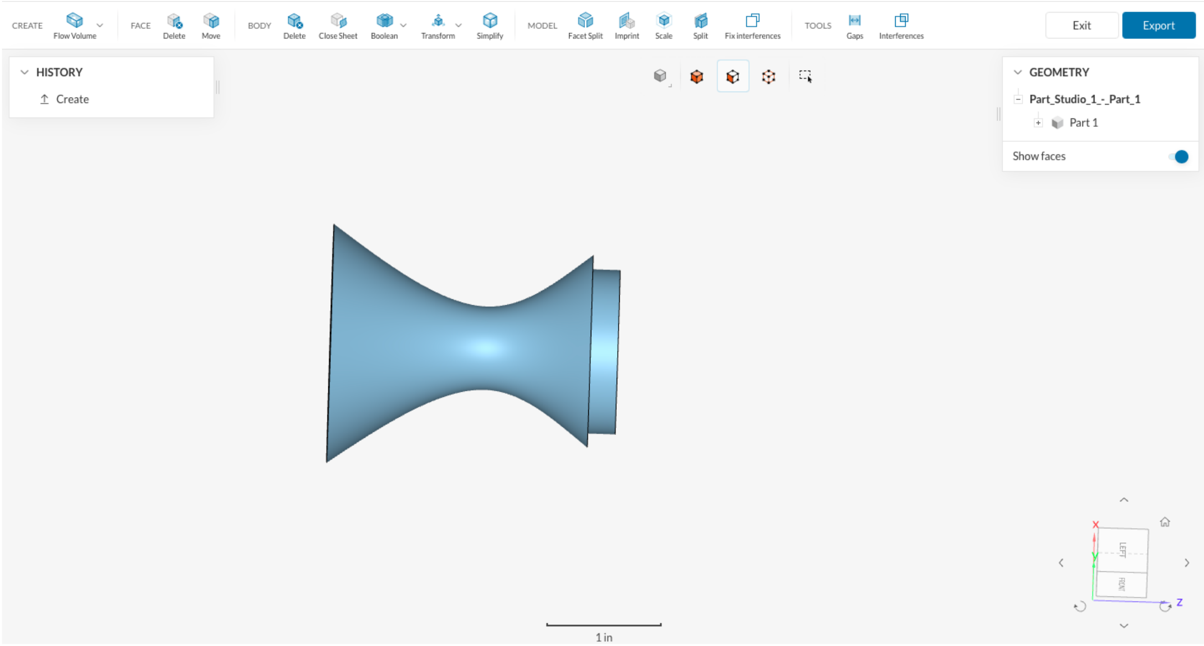


Figure 4.1 Diagram showing full cad model of converging diverging nozzle.

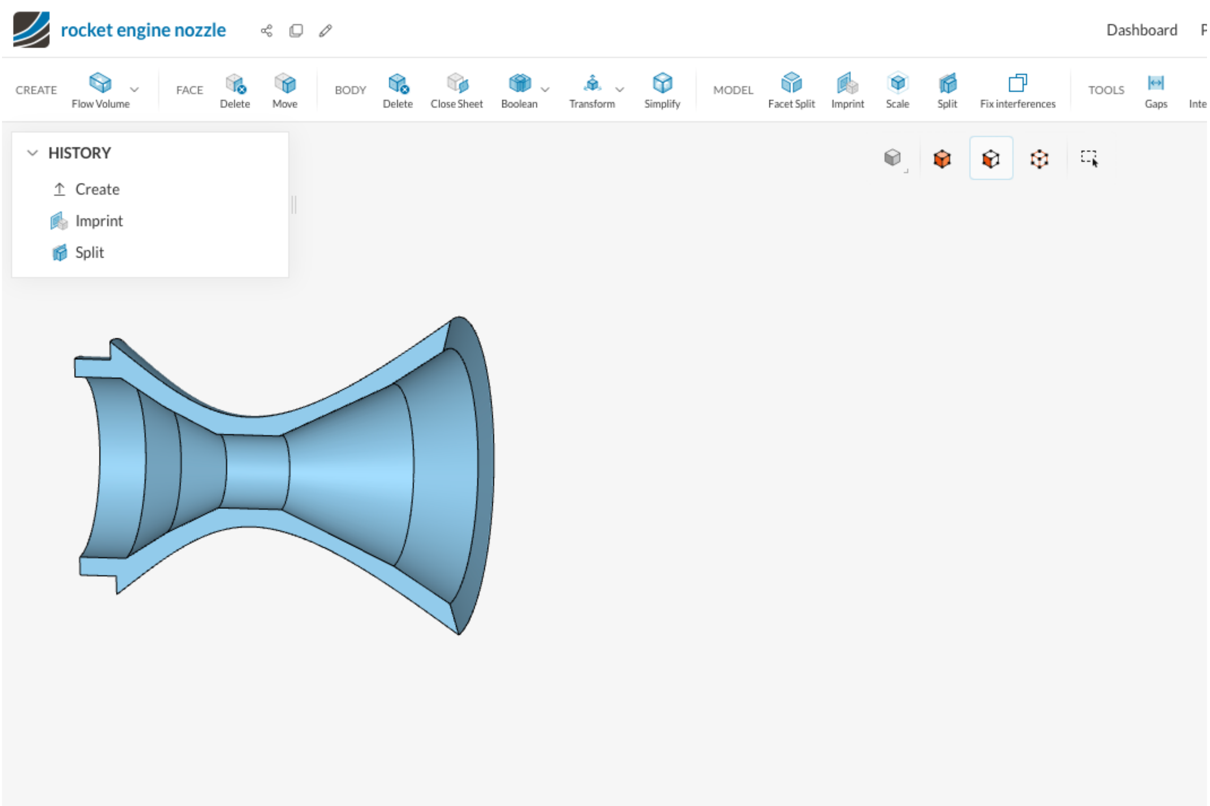


Figure 4.2 Diagram showing cutout of converging diverging nozzles

The radius can be seen decreasing first then decreasing which is critical for proper functions of rocket engines. The nozzle temperature can go up to 3300 k so it's very important to use proper cooling mechanisms.

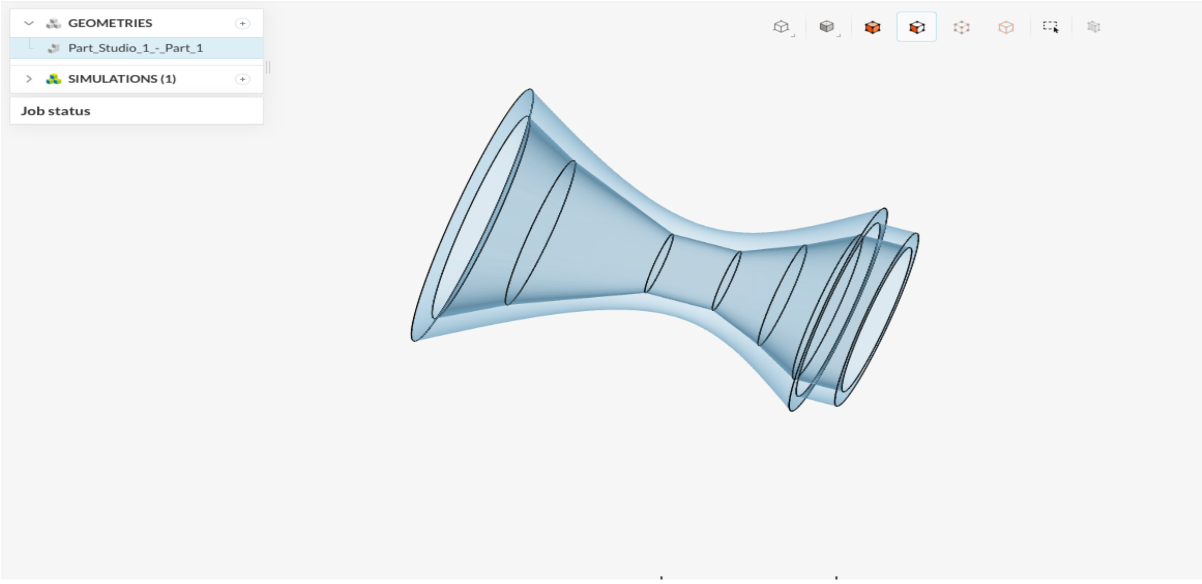


Figure 4.3 Proper visualisation of converging diverging nozzles with radius.

4.2 Simulation Conditions

When nozzle geometry is finalised then for simulations initial condition is set and for cooling techniques effectiveness heat transfer analysis is done and then compared for validation.

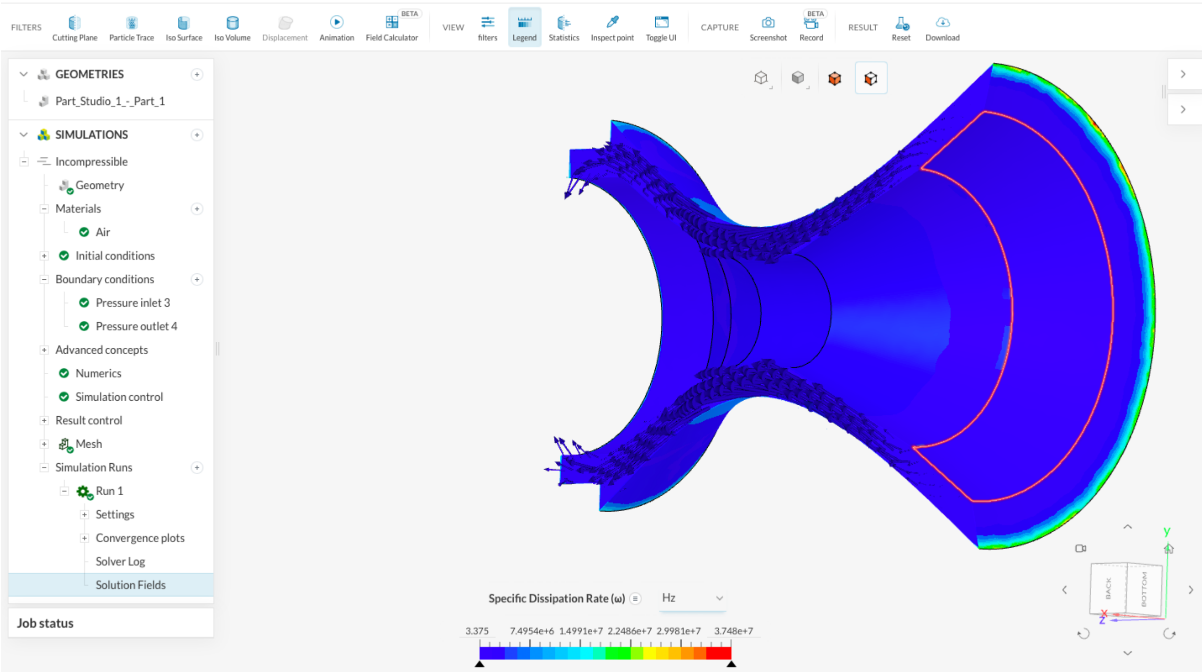


Figure 4.4 Proper visualisation of specific dissipation rate

The figure 3.4 shows the turbulence dissipation using the Low-Re k-ε model; it generally shows amount of energy is lost when flow is nearly turbulent due to viscosity. The model is very useful in heat transfer analysis with flow separation in turbulent flows. When Reynolds number increases the eddies increase sharply so does the losses. It affects heat transfer greatly as kinetic energy is converted into a low form of energy which is heat. We chose incompressible analysis, then we went to turbulence modeling and chose the k-ε model then we specified the boundary conditions with respect to the nozzle domain of our rocket engine cooling.

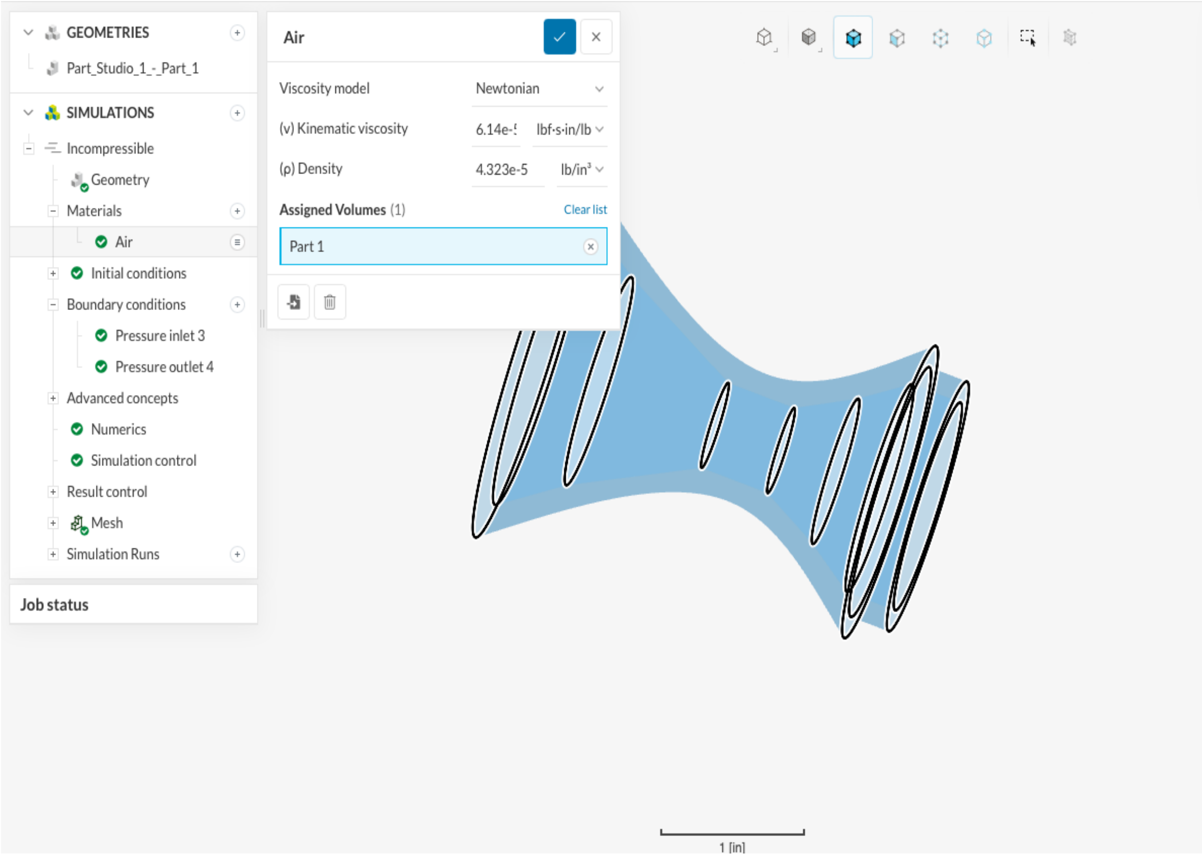


Figure 4.5 Proper visualisation of medium selection for analysis.

In previous figure we have chosen the turbulence modeling equation now we chose medium which is to be air and the viscosity modeling will be newtonian and we specify kinematic viscosity and density, once this is done then we specify the assigned volumes where this compressed air or gas passes through for proper modeling and mathematical analysis.

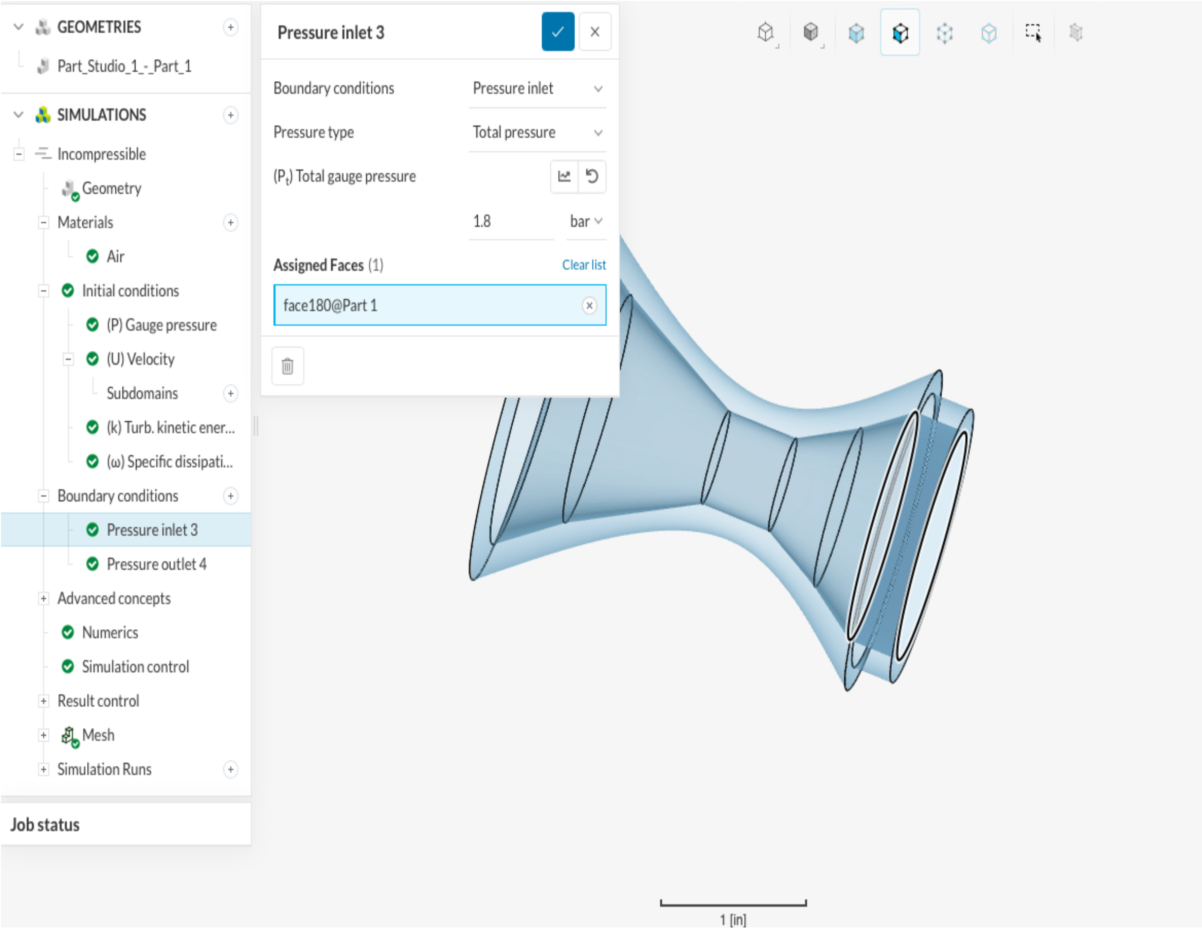


Figure 4.6 Proper visualisation of pressure boundary condition.

Now after medium we start setting boundary conditions and first it comes to pressure inlet 3 and pressure inlet 4. The pressure type is total pressure and total gauge pressure is 1.8 bar and also faces have been assigned. The pressure is directly linked to temperature and heat transfer analysis as pressure is very high in the combustion chamber and especially in the region of the throat. Now the rocket functions in extreme temperatures and pressures, the simulation modelling is also set on the same inputs to really bring the same environment to read and put exact data to be used by researchers in future.

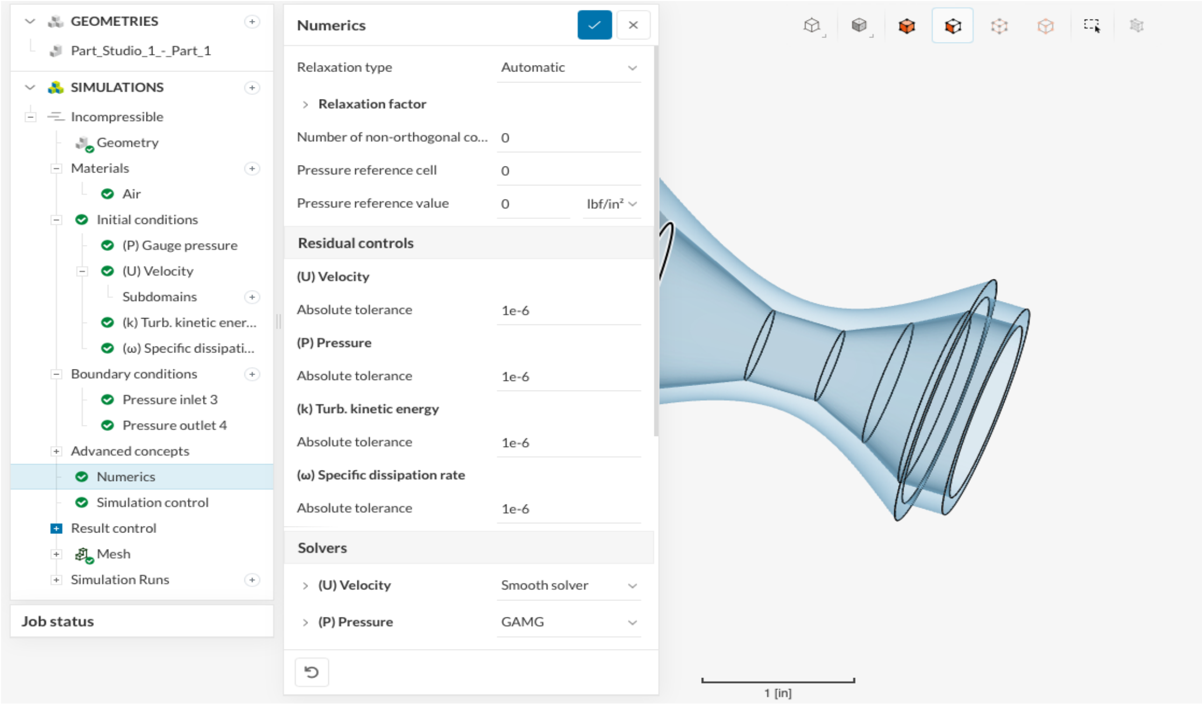


Figure 4.7 Proper visualisation of numerics.

The normal relaxation parameter is selected and at the same time for residual controls tolerance, turbulence kinetic energy and specific dissipation rate is filled. The important parameters which solver used are smooth for velocity and GAMG for pressure.

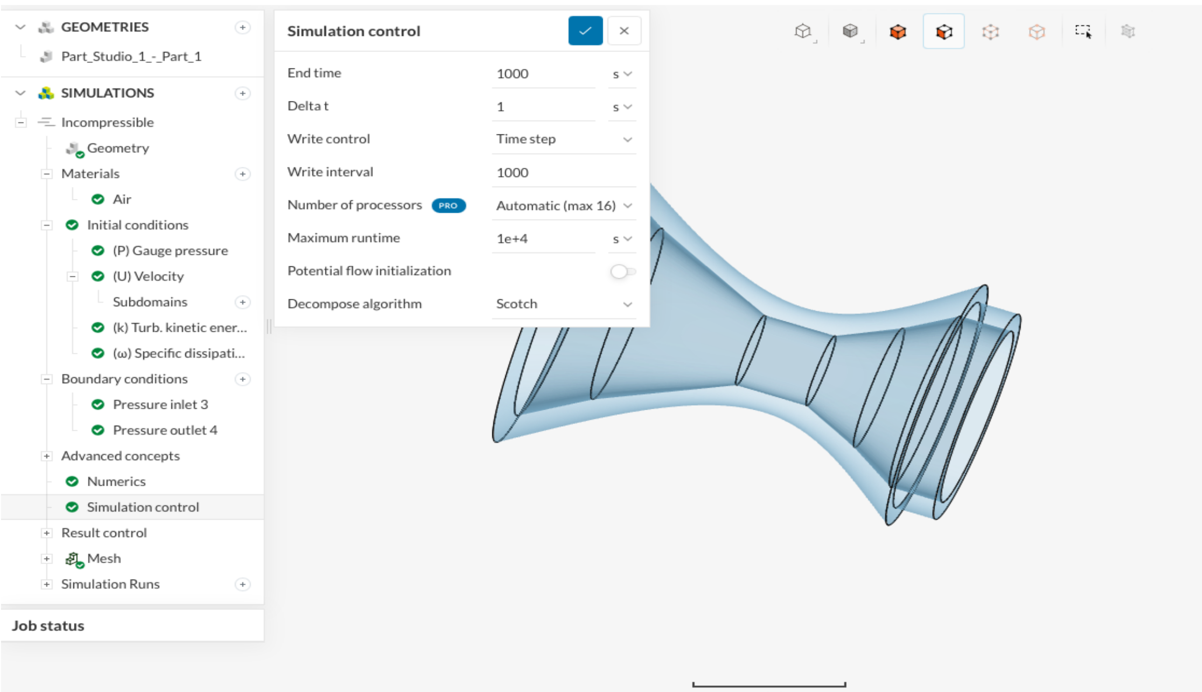


Figure 4.8 Proper visualisation of simulation control

In advance concepts time set is 1000 sec with timestamp and decompose algorithm is scotch.

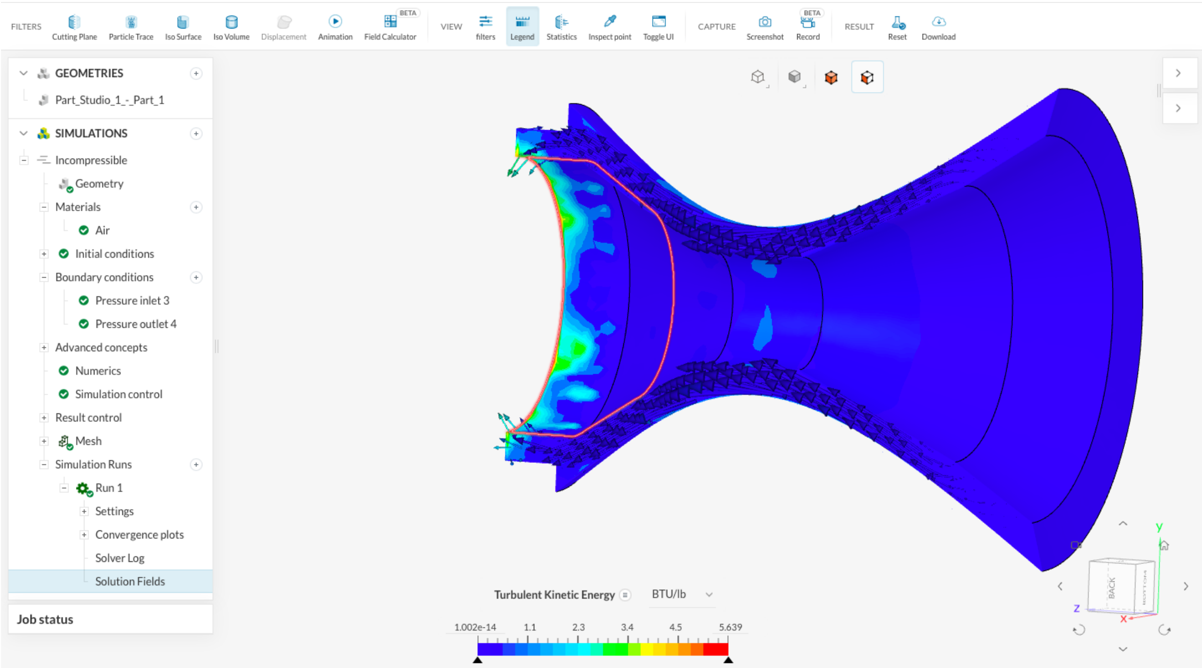


Figure 4.9 Proper visualisation of turbulent kinetic energy

Turbulent kinetic energy is associated with the number of eddies in flow.

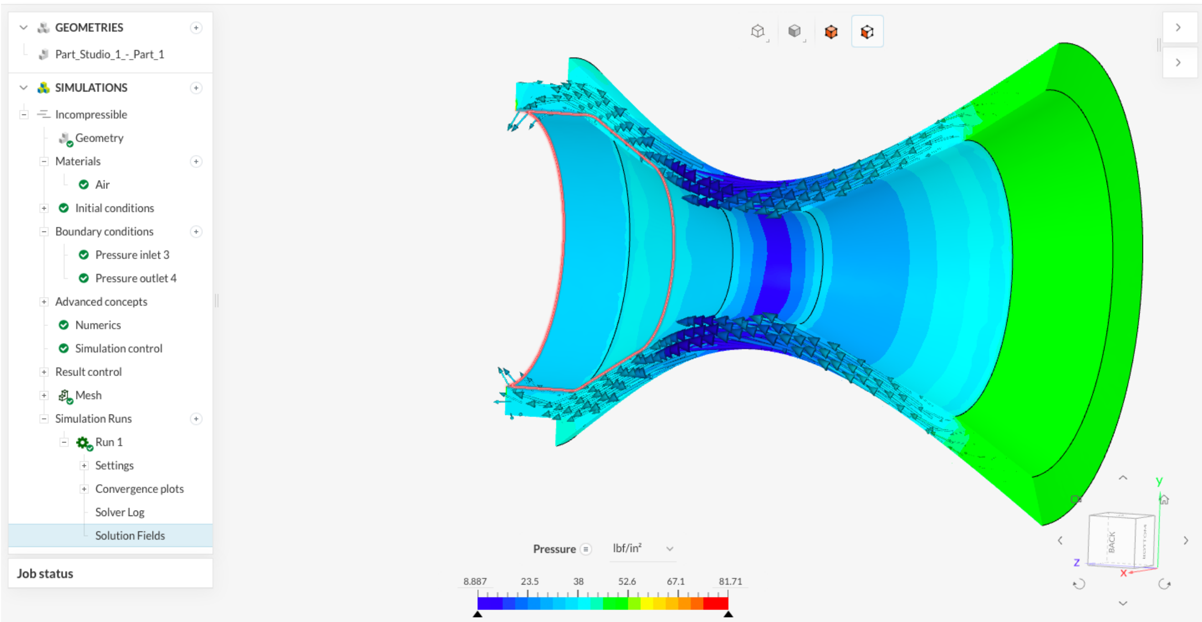


Figure 4.10 Proper visualisation of pressure in entire nozzle

The pressure is the reason gases flow inside the nozzle, key parameter for jet kinetic energy.

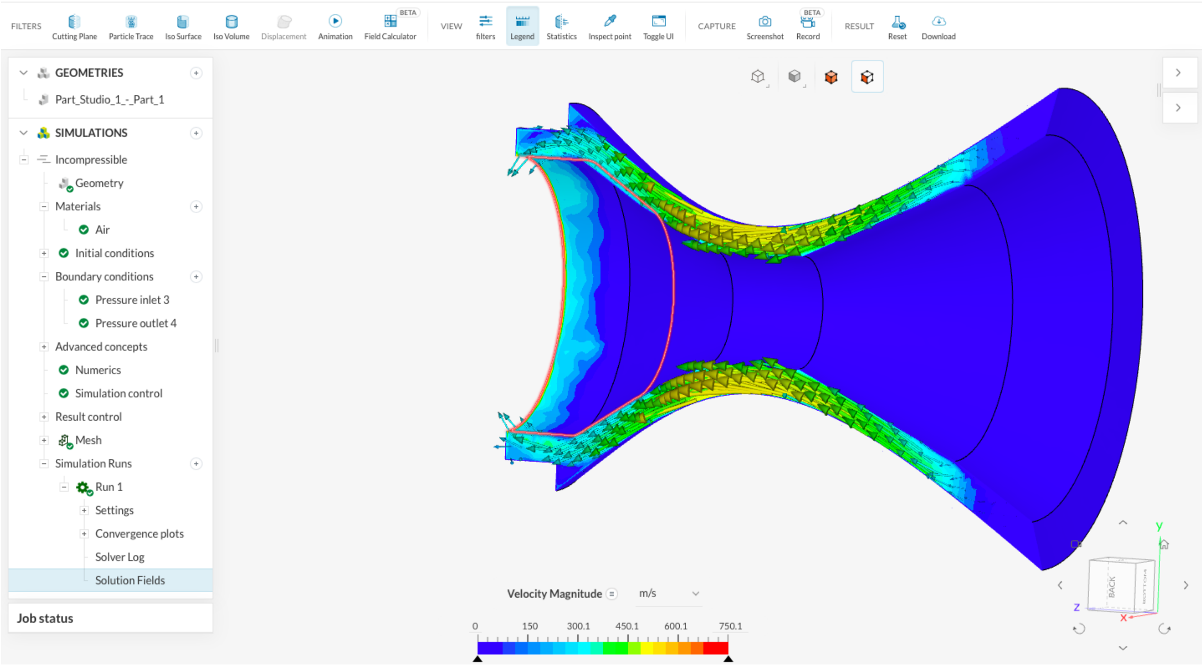


Figure 4.11 Proper visualisation of velocity magnitude.

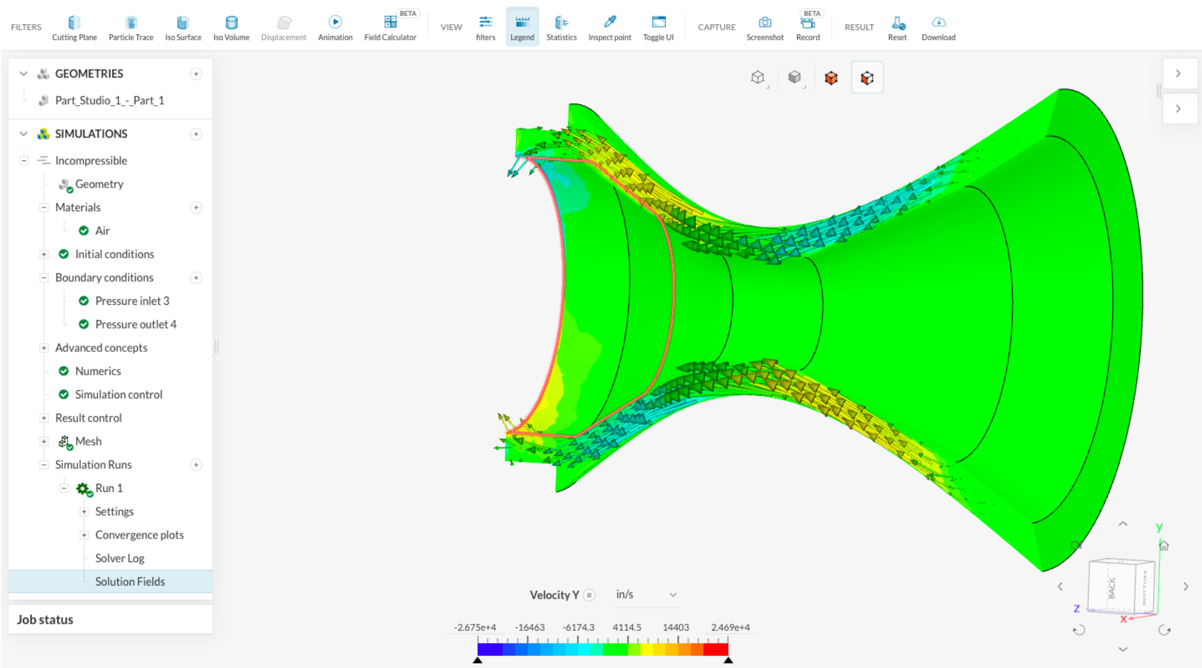


Figure 4.12 Proper visualisation of velocity magnitude in Y direction.

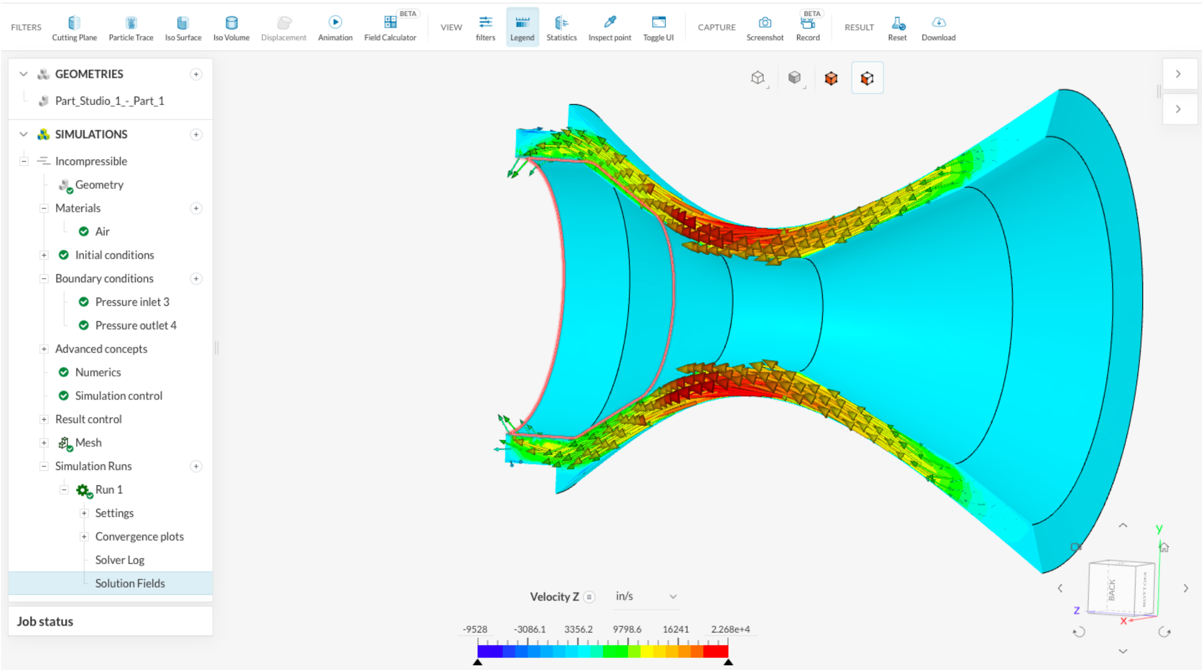


Figure 4.13 Proper visualisation of velocity in Z direction.

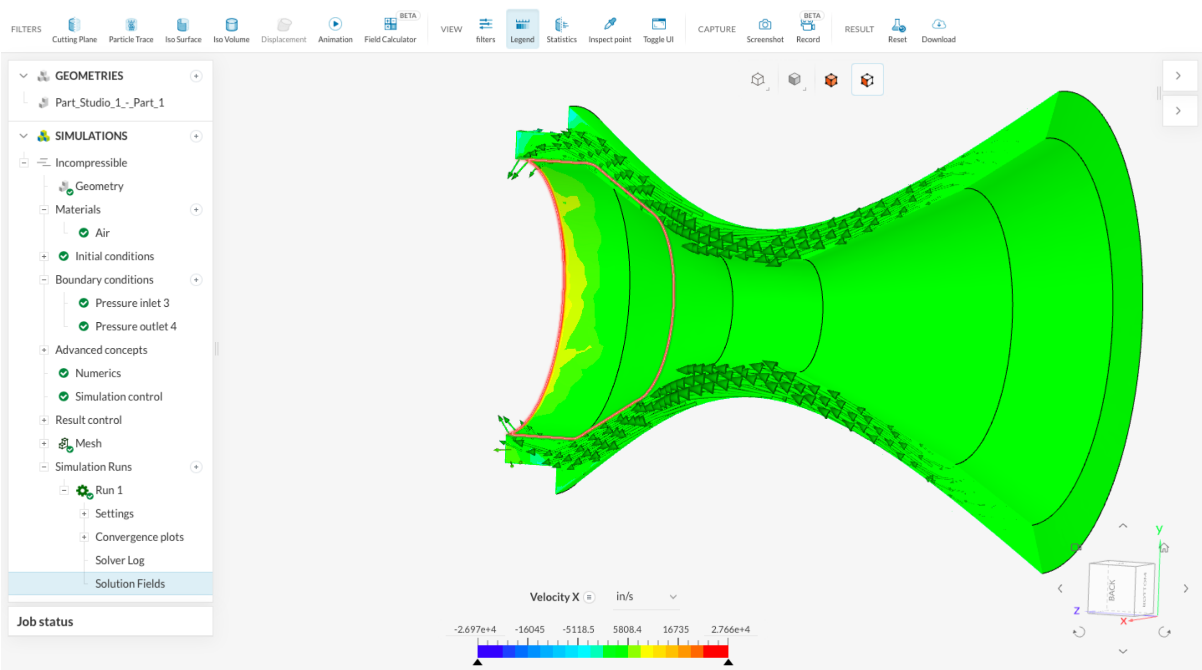


Figure 4.14 Proper visualisation of velocity in X direction.

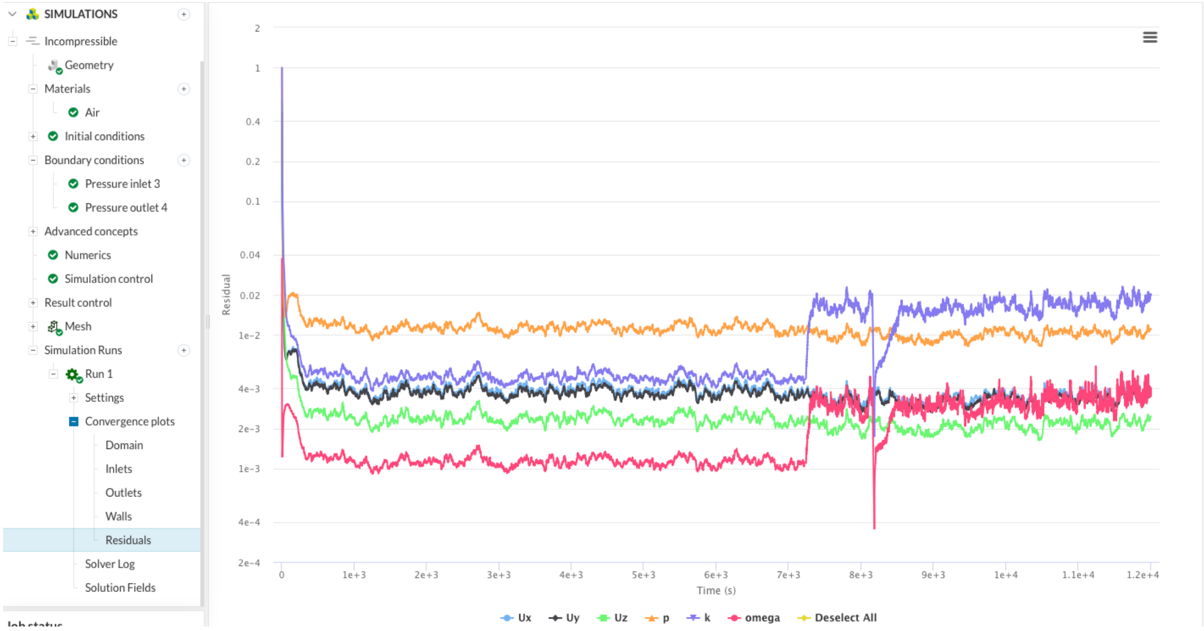


Figure 4.15 Proper visualisation of residual velocity, pressure, omega vs time in X, Y, Z directions.

The residuals are important because it shows variation, say imbalances in value theoretical according to mathematical conditions and practical simulations in extreme environments.

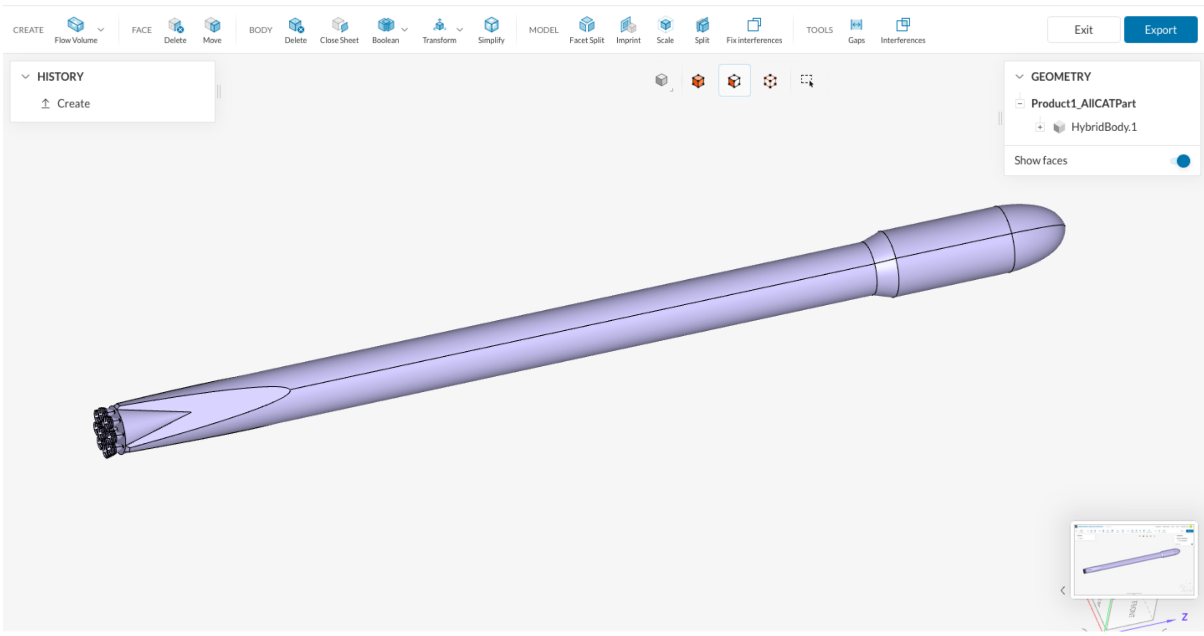


Figure 4.16 CAD model of spacex rocket for purpose of full scale temperature profile further

We took a full spacex falcon 9 rocket to visualize the temperature profile for heat transfer analysis in rockets. It will also visualize temperature in the entire volume region. The idea is to show temperature ranges which is shown in figure below

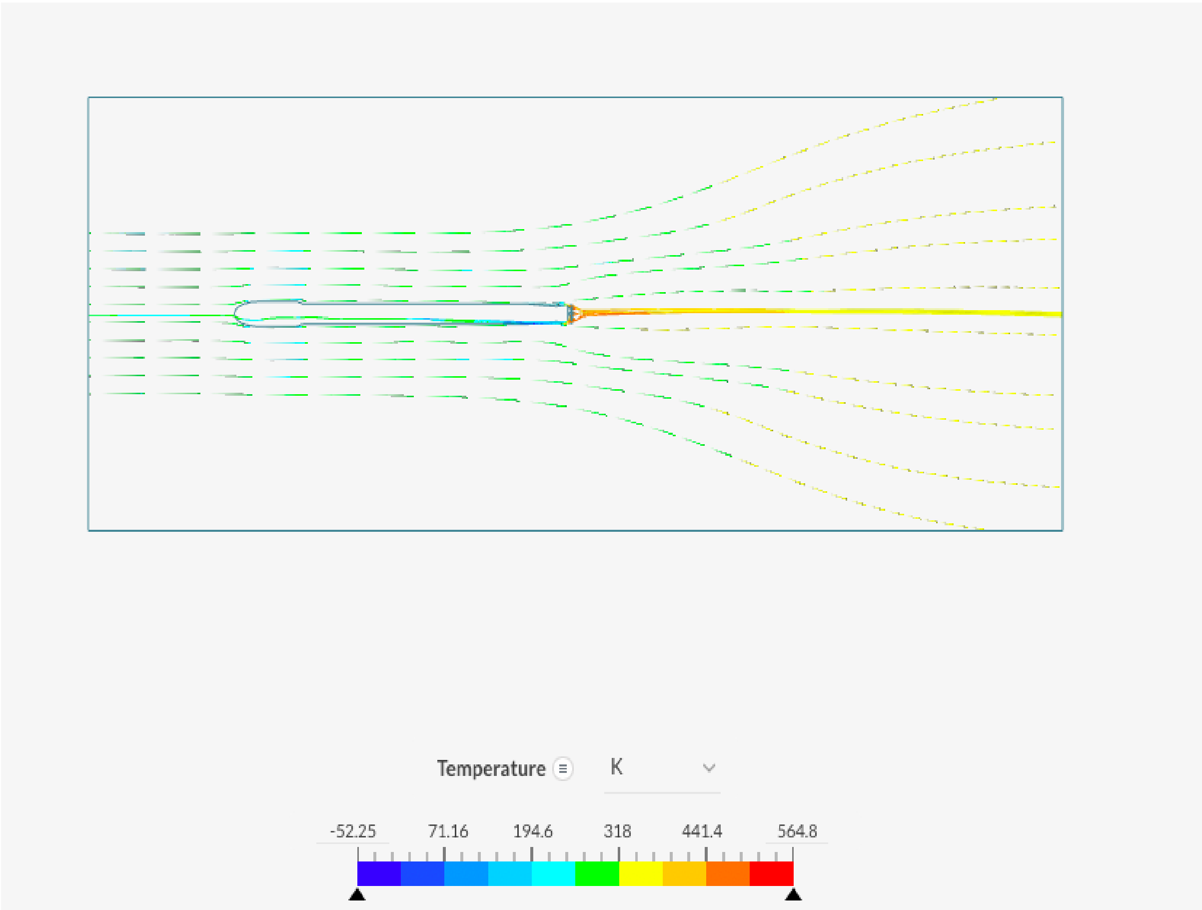


Figure 4.17 Temperature profile of outer surface of entire falcon 9 rockets in enclosed volume where heat transfer is taking place.

In the above figure we can see that the temperature around the front body is between 240 - 318 k but around the nozzle and when flue gases come out the temperature is too high, more than 441.4 k due to cooling techniques used otherwise it can reach up to 3200-3400k in throat of nozzles. External factors are also responsible like solar irradiation, electric equipment used in rockets and also the vacuum environment in space also induces more heat and that heat transfer has to be transferred out of the chamber for that purpose we used combined more regenerative and little film cooling effect for maximum output. This reduces our fuel consumption and our rockets can travel much farther than other methods.

Effectiveness = $\sqrt{kp/ hA_c}$
Efficiency = $\tanh mL/mL$
$m2 = hP/kA_c$
$\Delta p_{major\_loss} = \lambda (l / d_h) (\rho_f v^2 / 2)$

Table 4.1 Heat transfer analysis formula used shown in table.

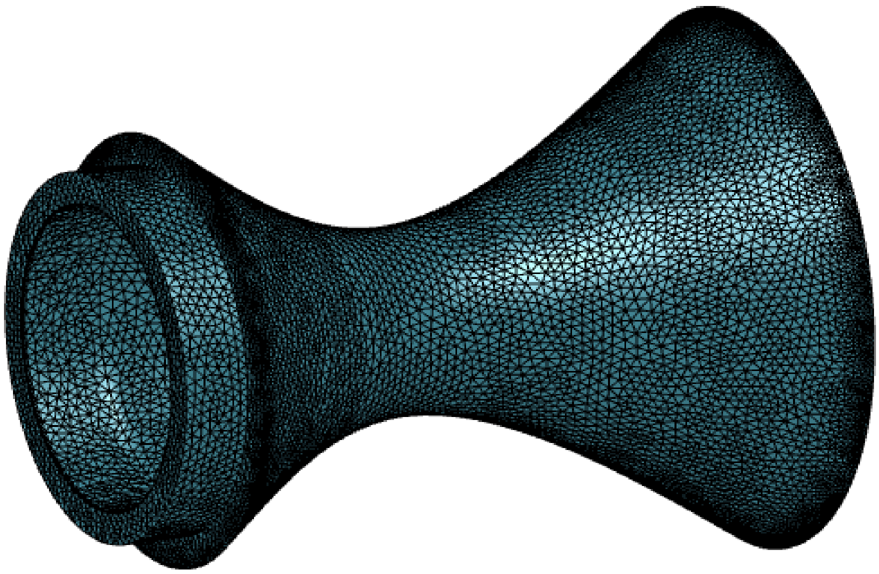
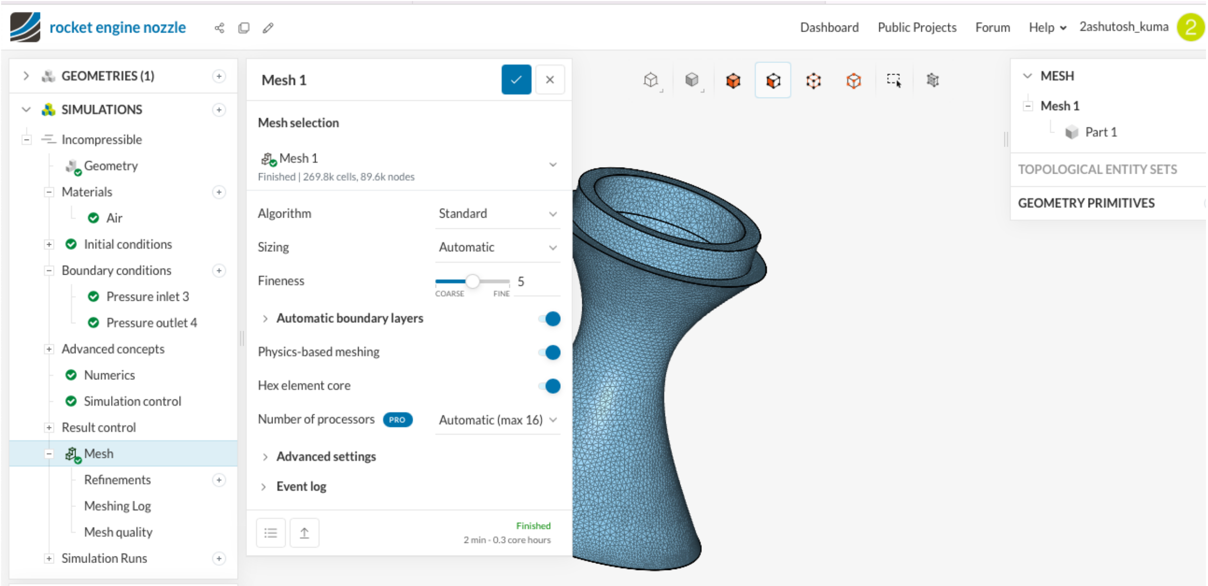


Figure 4.18 Figure showing fully meshed nozzles used for simulations of heat transfer analysis.




```
Maximum precision of model and its entities: 1e-08 m.
Absolute small feature tolerance: 9.99998e-06 m.
Gap refinement size is reaching lower cell size cap for 11 faces. Consider lowering model tolerance if resulting mesh fineness is too coarse.
Gap refinement size is reaching lower cell size cap for 1 regions. Consider lowering model tolerance if resulting mesh fineness is too coarse.
Mesh quality metrics:
tetEdgeRatio
        min: 1.000075508551453
        max: 17.18992231572657
        average: 1.73404732271161988
        standard deviation 0.4141243831797385
        median: 1.6996967745794362
                99.9-th percentile: 6.948421130376145
                99.99-th percentile: 14.184408825315797
                99.999-th percentile: 15.491782227414564
quadMaxAngle
        min: 89.9986443935383
        max: 172.49001777943457
        average: 104.49151569018396
        standard deviation 18.04218724170911
        median: 94.27910180723741
                99.9-th percentile: 163.98932422459785
                99.99-th percentile: 169.9788950122911
                99.999-th percentile: 172.48744233694558
triMaxAngle
        min: 60.00499566842345
        max: 159.01053825964098
        average: 78.0198744397219
        standard deviation 11.037710235229184
        median: 75.43295881031733
                99.9-th percentile: 120.3452243666937
                99.99-th percentile: 140.87921148604315
                99.999-th percentile: 147.14718532438565
triMinAngle
        min: 3.334881014966649
        max: 59.99750216578825
        average: 44.849452454160165
        standard deviation 7.382573788237353
        median: 45.789331025854494
                99.9-th percentile: 59.997502165787964
                99.99-th percentile: 59.997502165788234
                99.999-th percentile: 59.997502165788234
volumeRatio
        min: 1
        max: 14.516325703099278
        average: 1.330740608377294
        standard deviation 0.3208891343033776
        median: 1.2547206086326999
                99.9-th percentile: 3.558369901575678
                99.99-th percentile: 8.146224205845945
                99.999-th percentile: 10.416902684231566
```

Figure 4.19 Temperature profile of outer surface of entire falcon 9 rockets in enclosed volume where heat transfer is taking place.

In meshing 16 core advance processor has been used with hex element in core and also automatic boundary layer, physics based meshing is selected and algorithm used is hex dominant parametric. Fineness level selected is 5. The number of nodes in the event log is 89643, the number of edges is 1327, the number of faces is 39042, the number of volumes 269769. the number of triangles is 38172, the number of quadrangles is 870 the number of prism is 99805, the number of pyramids is 1163, the number of tetrahedral is 168781 and the number of hexahedral is 20.

```
volumeRatio
        min: 1
        max: 14.516325703099278
        average: 1.330740608377294
        standard deviation 0.3208891343033776
        median: 1.2547206086326999
                99.9-th percentile: 3.558369901575678
                99.99-th percentile: 8.146224205845945
                99.999-th percentile: 10.416902684231566
tetAspectRatio
        min: 1.0000503472711681
        max: 7.648661589181014
        average: 1.5417064086596326
        standard deviation 0.25644217376368467
        median: 1.5297188151620866
                99.9-th percentile: 4.054128433750193
                99.99-th percentile: 6.379350495031744
                99.999-th percentile: 7.085628027171181
nonOrthogonality
        min: 0
        max: 84.13814572337098
        average: 19.678915283786885
        standard deviation 14.825940534148762
        median: 16.828345456801358
                99.9-th percentile: 72.68358435528485
                99.99-th percentile: 81.1823368141777
                99.999-th percentile: 82.97636436844874
skewness
        min: 0
        max: 9.180081869944916
        average: 0.12716503277196592
        standard deviation 0.1828951831492877
        median: 0.098105650208181
                99.9-th percentile: 2.662362446040738
                99.99-th percentile: 4.9782403588215995
                99.999-th percentile: 6.394748933262619
aspectRatio
        min: 0.05543729532916604
        max: 7.648661589181014
        average: 1.0239266025700684
        standard deviation 0.7039196884389293
        median: 1.3283469518049502
                99.9-th percentile: 2.9249160065738584
                99.99-th percentile: 6.1937068377131075
                99.999-th percentile: 7.028762983344996
Overall mesh quality (based on the 99-percentile): 0.481794
Overall mesh quality is computed from:
        Non Orthogonality: 64.047862 (normalized value: 0.43762, weight: 0.60)
        Skewness: 0.497475 (normalized value: 0.608064, weight: 0.30)
        Aspect Ratio: 1.999609 (normalized value: 0.368023, weight: 0.10)
```

Figure 4.20 Figure showing mesh log event for volume ratio, aspect ratio, non orthogonality and skewness.

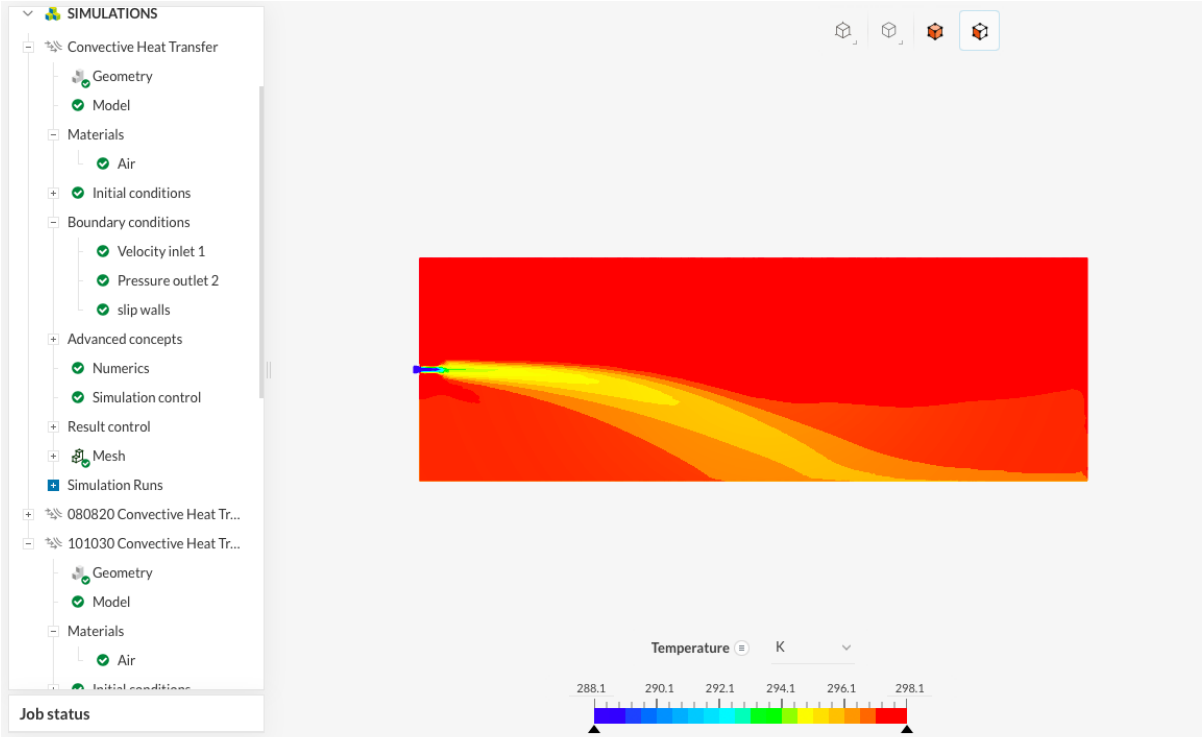


Figure 4.21 Figure showing convective heat transfer simulations and temperature profile of falcon 9 merlin 1 d vacuum engine rocket.

While doing convective heat transfer analysis in modelling radiation has also been taken, flow has taken to be compressible, k-sst omega has been chosen as turbulence modeling and then we get the desired results.

CHAPTER 5
CONCLUSIONS

Future Works

Different cooling techniques can be blended together and can be simulated to find the effect mostly temperature inside nozzle of not just merlin engine but various other engines and also the simulation platform can be advance like that used in ISRO, Spacex, NASA, JAXA, ROSCOSMOS or other commercial companies so that results coming could be more accurate. The author in the future in PHD will continue expansion on this work.

References

[1] Brief history of rockets published on NASA website https://www.grc.nasa.gov/www/k-12/TRC/Rockets/history_of_rockets.html

[2] Indian Space Research Organisation {ISRO} published on wikipedia https://en.wikipedia.org/wiki/Indian_Space_Research_Organisation

[3] George P. Sutton and Oscar Biblarz, “Rocket Propulsion Elements”, Eighth Edition, John Willey & Sons Inc, 2010

[4] S.R. Shine, S.Shri Nidhi, Review on film cooling of liquid rocket engines, Propulsion and Power Research, Volume 7, Issue 1, 2018, Pages 1-18, ISSN 2212-540X, https://doi.org/10.1016/j.jppr.2018.01.004.

[5] KANDA, Takeshi & Sato, Masaki. (2016). Radiative Heating in Combustion Chamber of Liquid Propellant Rocket Engines. TRANSACTIONS OF THE JAPAN SOCIETY FOR AERONAUTICAL AND SPACE SCIENCES. 59. 332-339. 10.2322/tjsass.59.332.

[6] Mustafa emre boysan, Msc thesis submitted on“Analysis of regenerative cooling in liquid propellant rocket engines”, Middle east technical university, 2008

[7] Miranda, Alexander & Naraghi, Mohammad. (2011). Analysis of Film Cooling and Heat Transfer in Rocket Thrust Chamber and Nozzle. 10.2514/6.2011-712.

[8] Ciasullo, Kevin J. “Design of a Regenerative Nozzle for Aluminum-water Combustion.” (2016).

[9] Buongiorno, Jacopo. “MIT22_06F10_lec13.pdf.” MIT Department of Nuclear Science and Engineering, n.d

[10] Amirhossein Adami, Mahdi Mortazavi, Mehran Nosratollahi, Mohammadreza Taheri, Jalal Sajadi, "Multidisciplinary Design Optimization and Analysis of Hydrazine Monopropellant Propulsion System", *International Journal of Aerospace Engineering*,vol. 2015, Article ID 295636, 9 pages, 2015. https://doi.org/10.1155/2015/295636

[11] Bai, Mo & Chung, J.. (2011). Analytical and numerical prediction of heat transfer and pressure drop in open-cell metal foams. International Journal of Thermal Sciences. 50. 869-880. 10.1016/j.ijthermalsci.2011.01.007.

[12] J. Stoll, J. Straub, Film cooling and heat transfer in nozzles, J. Turbomach. 110 (1988).

[13] C.J. Marek, R.R. Tacina, Effects of free-stream turbulence on film cooling, NASA TN D-7985, 1975.

[14] Fan, Chuan & Wang, Jing & Fan, Xuejun. (2017). Experimental Studies of Film Cooling in Supersonic Combustors. 10.2514/6.2017-2267.

[15] Juhany, Khalid A., Melany L. Hunt and Joseph M. Sivo. "Influence of injectant Mach number and temperature on supersonic film cooling." Journal of Thermophysics and Heat Transfer 8 (1994): 59-67.

[16] Hombsch, Maximilian & Olivier, Herbert. (2013). Film Cooling in Laminar and Turbulent Supersonic Flows. Journal of Spacecraft and Rockets. 50. 742-753. 10.2514/1.A32346.

[17] König, Valentina & Rom, Michael & Mueller, Siegfried. (2020). A Coupled Two-Domain Approach for Transpiration Cooling. 10.1007/978-3-030-53847-7_2.

[18] Peter, Johannes & Kloker, Markus. (2020). Direct Numerical Simulation of Supersonic Film Cooling by Tangential Blowing.

<u>List Of Publications [Certificates]</u>

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