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INVESTIGATION OF SUPERSONIC FLOW THROUGH CONICAL NOZZLE WITH VARIOUS ANGLES OF DIVERGENCE

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ABSTRACT

CFD studies are carried out on conical nozzles with different divergence angles using supersonic gas flows though the nozzle. In the present investigation the nozzles are modeled with axi symmetric condition and modeling is carried out using Gambit software. Fluent solver is adopted in the present study to investigate the flow characteristics by considering the throat diameter and exit diameter of nozzle is same for all cases.

The flow parameters, such as pressure ratio, Mach number of the flow at the nozzle exit, and the area of nozzle exit ratio are considered for the simulation studies. The result shows the variation in the Mach no., Pressure, Temperature, Velocity of flow, Turbulence Intensity at different divergence angles.

KEYWORDS: Conical, Pressure Ratio, Temperature, Turbulence Intensity, Mach Number

INTRODUCTION

A Nozzle is a device designed to control the direction or characteristics of a fluid flow (especially to increase velocity) as it exits (or enters) an enclosed chamber or pipe via an orifice.

A nozzle is often a pipe or tube of varying cross sectional area and it can be used to direct or modify the flow of a fluid (liquid or gas). They are frequently used to control the rate of flow, speed, direction, mass, shape, and/or the pressure of the stream that emerges from them.

The aim of a nozzle is to increase the kinetic energy of the flowing medium at the expense of its pressure and internal energy. Nozzles can be described as convergent (narrowing down from a wide diameter to a smaller diameter in the direction of the flow) or divergent (expanding from a smaller diameter to a larger one). A de Laval nozzle has a convergent section followed by a divergent section and is often called a convergent-divergent nozzle.

Convergent nozzles accelerate subsonic fluids. If the nozzle pressure ratio is high enough the flow will reach sonic velocity at the narrowest point (i.e. the nozzle throat). In this situation, the nozzle is said to be choked. Convergent-divergent nozzles can therefore accelerate fluids that have choked in the convergent section to supersonic speeds. This CD process is more efficient than allowing a convergent nozzle to expand supersonically externally.

The shape of the divergent section also ensures that the direction of the escaping gases is directly backwards, as any sideways component would not contribute to thrust.

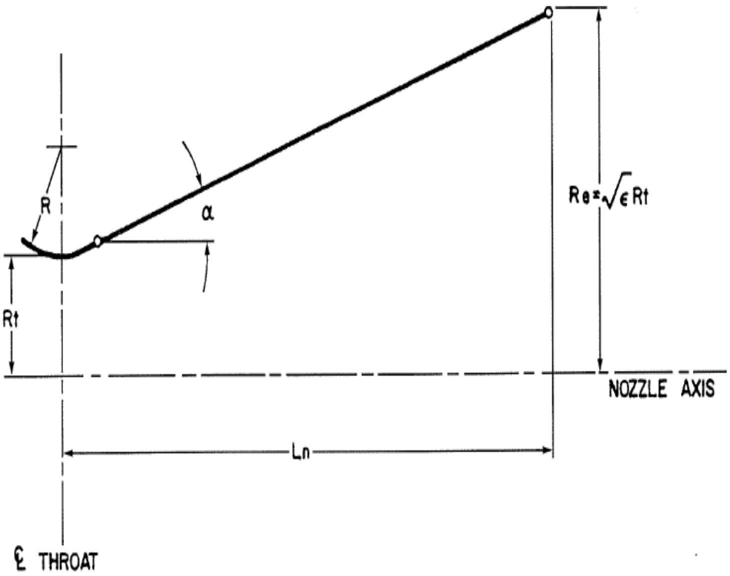


Fig 1: Conical nozzle **Fig 2: Conical Nozzle Design with a Rounded Throat**

Rocket motors use convergent-divergent nozzles with very large area ratios so as to maximize thrust and exhaust velocity and thus extremely high nozzle pressure ratios are employed. Mass flow is at a premium since all the propulsive mass is carried with vehicle, and very high exhaust speeds are desirable. B.E. Milton and K Pianthong [2] work on pulsed, supersonic liquid fuel jets. These have been studied experimentally for low and high range supersonic Mach numbers and simulated for the low Mach numbers only. Both experiments and simulation is complex. The latter show some similarity to the trends calculated from empirical formulas used for high subsonic diesel jets although the specific values need to be reassessed. Heated air is used as a reservoir gas of supersonic nozzle. Hydrogen is injected transversely through circular hole into free stream of Mach 2. Flow duration is 300 μs. Schlieren method and CCD UV camera are used to obtain information on the shock structures and the region of combustion. The effects of total pressure of injection gas to the fuel penetration and the region of combustion have been obtained. Laser-induced fluorescence of OH and CH_2O [4] was imaged to investigate the flame stabilization mechanism in a flame holder with a Mach 2.4 free stream. Ethylene was burned in a rectangular cavity with two points of injection: the aft wall and the cavity floor. The processes of improving the implementation of the combustion process in the thruster duct remain topical: [5] In the flight range of the flying vehicle M<8, the most efficient combustion is realized in a pseudo shock flow structure. The localized, intense heat supply may cause the duct choking prior to the thermal crisis; the combustion process deterioration because of the dissociation at flight speeds over M = 8 is one of the main problems, which require their solution.

New experimental data were obtained for the study of kinetic coupling between DME and methane and for the validation of existing kinetic mechanisms. [6] The following conclusions can be drawn from the present work. The present pressure-based method successfully predicts the [7] liquid kerosene-oxygen spray combustion in the liquid fuel rocket engine at all speeds. In this study, soot formation at high temperatures and in a high pressure combustor is also investigated using a two-step global model. A survey of physical locations of the [8] optimal discharge placement and their common mixture fraction was performed for a JICF. A numerical study of mixing and combustion enhancement has been performed for a Mach 2 model scramjet (supersonic combustion ramjet) combustor. [9] Fuel (hydrogen) is injected at supersonic speed through the rear of a lobed strut located at the channel symmetry axis. The shape of the strut is chosen in a way to produce strong stream wise vortices and thus to enhance the hydrogen/air mixing. In this numerical study the influence of chemistry models on the predictions of supersonic combustion in a model combustor is investigated.[10] To this end, 3D, compressible, turbulent, reacting flow calculations with a detailed chemistry model (with 37 reactions and 9 species) and the Spalart–Allmaras turbulence model have been carried out. These results are compared with earlier results obtained using single step chemistry. A large eddy simulation (LES) model with a new localized dynamic sub grid closure for the magneto hydrodynamics (MHD) equations is used to investigate plasma-assisted combustion in supersonic flow. [11] A 16-species and 74-reactions kinetics model is used to simulate hydrogen-air combustion and high-temperature air dissociation. Spontaneous ignition of pressurized hydrogen released [12] through a tube into air is investigated using a modified version of the KIVA-3V CFD code .A mixture-averaged multi-component approach is used for accurate calculation of molecular transport. Auto ignition and combustion chemistry is accounted for using a21 step kinetic scheme. Problems of methane-containing hydrocarbon fuel preparation [13] on board a launch vehicle, organization of combustion, control of a burn-out zone length and a combustion effect on thermo gas dynamic parameters in the combustion chamber of a rocket-ramjet engine are discussed in the paper. A possibility of a considerable enhancement of ignition in a diffusion [14] mode in a supersonic non-premixed H_2 – air flow by means of the laser- induced excitation of O_2 molecules to the $b^1\Sigma_g^+$ electronic state is investigated on the base of numerical simulation. An investigation of the thrust characteristics and internal pressure distributions of two convergent-divergent 15 deg. half-angle exhaust nozzles [16] having area ratios of 6 and 9 was made in the NASA Lewis 10- by 10-foot supersonic wind tunnel. Supersonic Combustion of hydrogen has been presented with strut flat duct length. The combustor has a single fuel injection parallel to the main flow from the base. [17]. Finite rate chemistry model with S-A model have been used for modeling of supersonic combustion. In this paper strut at 60 with flat duct length analyzed for without hydrogen injection, with hydrogen injection and hydrogen injection with combustion. Rockets typically use a fixed convergent section followed by a fixed divergent section for the design of the nozzle. This nozzle configuration is called a convergent-divergent [18], or CD nozzle. The Function of the rocket nozzle is to convert the thermal energy in the propellant into kinetic energy as efficiently as possible, in order to obtain high exhaust velocity along the desired direction.

CALCULATION OF PARAMETERS IN CONICAL NOZZLE

A typical conical nozzle design with a rounded throat is shown in figure2.Two parameters will be of interest in calculating the losses for, and optimizing the design of straight- cut throats; the thrust coefficient and specific impulse. The nozzle thrust coefficient is defined as the thrust divided by the product of the chamber pressure and the nozzle throat area.

$$C_F \equiv \frac{F}{p_c\, A_{th}}$$

$$F = C_F A_{th}\, p_c$$

Where, A_{th} = nozzle throat area, m^2 (in^2), C_F = thrust coefficient, F = thrust, N (lb), P_c= chamber pressure, Pa (lb/in^2). The specific impulse of the rocket motor is the thrust of the motor divided by the propellant flow rate.

$$I_{sp} = \frac{F}{\dot{m} g_0}$$

Where: F = thrust, N, g_o = acceleration due to gravity at sea level=9.8066 m/s^2(32.174 ft/sec^2) , I_{sp}, = specific impulse, N-sec/kg (sec), $\dot{m}$ = propellant mass flow rate, kg/sec. As will be seen, the thrust coefficient and specific impulse are interrelated. If a nozzle produces a higher thrust by having a higher thrust coefficient for a particular propellant flow rate and chamber pressure, the specific impulse of the rocket motor will also be increased. Another parameter of interest for understanding losses in specific impulse from combustion losses upstream of the nozzle throat is the characteristic velocity (c*). The characteristic velocity is defined by

$$c^* \equiv \frac{p_c \, A_{th}}{\dot{m}}$$

Where,c* = characteristic velocity, m/sec (ft/sec). A theoretical expression for characteristic velocity is derived showing that the characteristic velocity is a function of the combustion conditions and the ratio of specific heats.

$$c^* = \left[\frac{1}{\gamma}\left(\frac{\gamma+1}{2}\right)^{\frac{\gamma+1}{\gamma-1}} \frac{RT_c}{\overline{M}}\right]^{1/2}$$

The thrust of the rocket motor as a function of mass flow, characteristic velocity, and thrust coefficient.

$$F = \dot{m} c^* C_F$$

and Isp = (c^* C_F) / g_o

To determine the thrust coefficient for the nozzle of a rocket motor the ideal thrust coefficient is calculated first. Corrections are then made to the ideal thrust coefficient to take into account departures from the ideal performance assumptions used to derive the ideal thrust coefficient equation. The derivation of the ideal thrust coefficient equation presented is based on a control volume drawn around a liquid rocket engine thrust chamber and nozzle. The assumption is made in the derivation that the stream thrust from injecting the propellants into the combustion chamber is zero, due to the velocity of the propellant flow into the chamber being much lower than the nozzle exhaust velocity. This same assumption can be made for a hybrid rocket motor.

MODELING AND MESHING CFD ANALYSIS

The geometry of the nozzle and meshing was created in Gambit software and analysis is done in fluent software. Pressure far-field conditions are used in **FLUENT** to model a free-stream condition at infinity, with free-stream Mach number and static conditions being specified. The pressure far-field boundary condition is often called a characteristic boundary condition, since it uses characteristic information (Riemann invariants) to determine the flow variables at the boundaries.

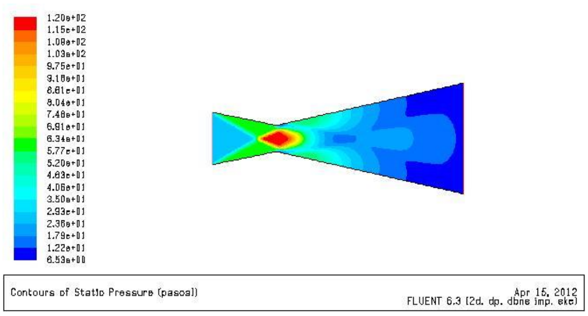
RESULTS AND DISCUSSIONS

After the completion of CFD simulations to study the characteristics of the Conical Nozzle, the obtained results are shown in table1.

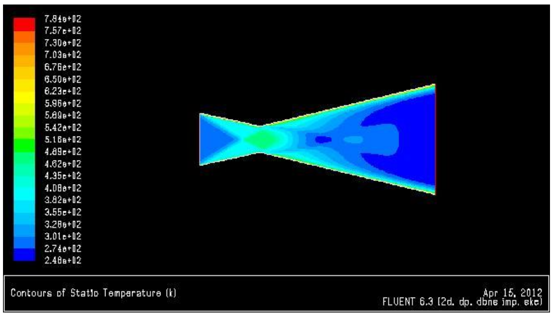
Table1: The Variation of the Temperature, Pressure, Velocity and Turbulence with Respect to Divergence Angle of the Conical Nozzle

S.No	Divergence Angles, (deg)	Pressure, Pa	Temperature,k	Velocity Magnitude, m/s	Turbulence Intensity,m2/s2
1	7.21	1.02E+02	7.84E+02	1.08E+03	1.02E+04
2	20	9.58E+01	7.85E+02	1.08E+03	1.02E+04
3	30	9.62E+01	7.83E+02	1.11E+03	1.18E+04
4	40	8.73E+01	7.87E+02	1.13E+03	1.31E+04

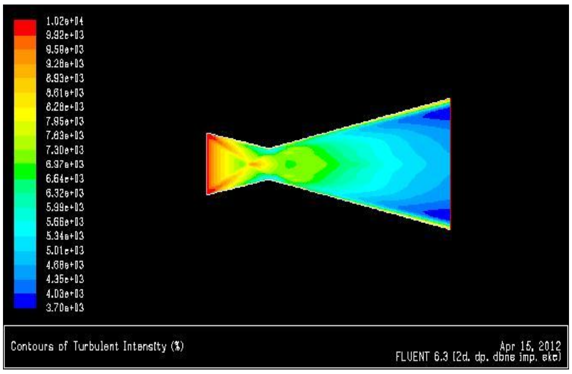
It is observed from the above results that the maximum pressure of 1020pa is observed at the divergence angle of 7.21^0 then it decreased to 95.8pa at an angle of 20^0 then it gradually decreases to 96.2pa at angle of 30^0 and 87.3pa at a divergence angle of 40^0. As the divergence angle increases the area of the divergent portion enlarged then the pressure decreases. It is observed from the obtained results that the maximum temperature of 785^0k is obtained at 20^0 and it gradually decreases to 783^0 k at 30^0 from there it increases to787^0k at 40^0 of divergence angle. It is observed from the obtained results that the maximum Velocity Magnitude of 1.13m/s is obtained at 40^0 and it gradually decreases to 1.11 at 30^0 and there is a constant decrease to 1.08 at 20^0 & 10^0 of divergence angle. Here, as the divergent angle increases the area increases and the pressure is inversely proportional to the velocity. Therefore, as the pressure decreases the velocity increases. It is observed from the obtained results that the maximum Turbulence Intensity of $1.31m^2/s^2$ is obtained at 40^0 and it gradually decreases to 1.18 m^2/s^2 at 30^0 and there is a constant decrease to 1.02 m^2/s^2 at 20^0 & 10^0 of divergence angle. As the divergence increases the turbulence increases. Due to turbulence the acceleration of the fluid increases. So there is a maximum discharge in the nozzle.



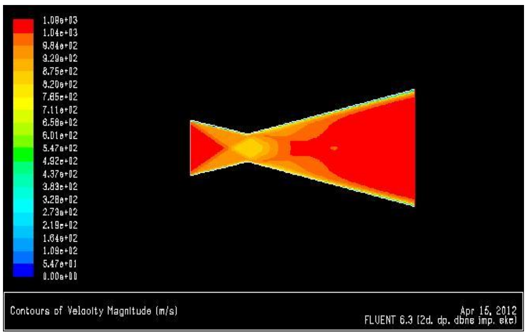
Contours of Static Pressure



Contours of Static Temperature

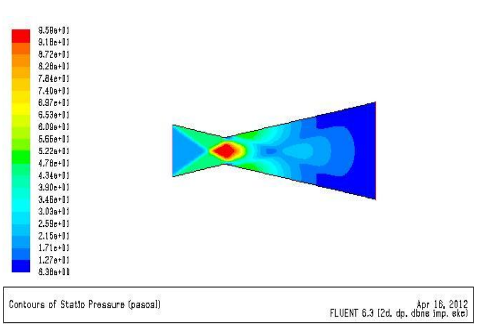


Contours of Turbulence Intensity

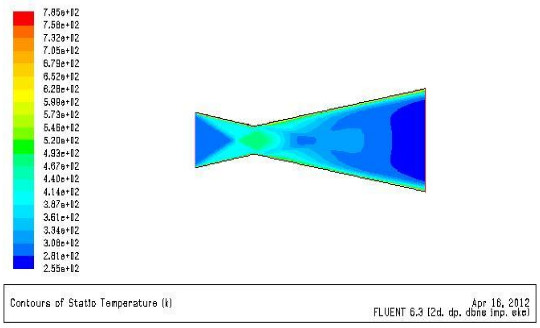


Contours of Velocity Magnitude

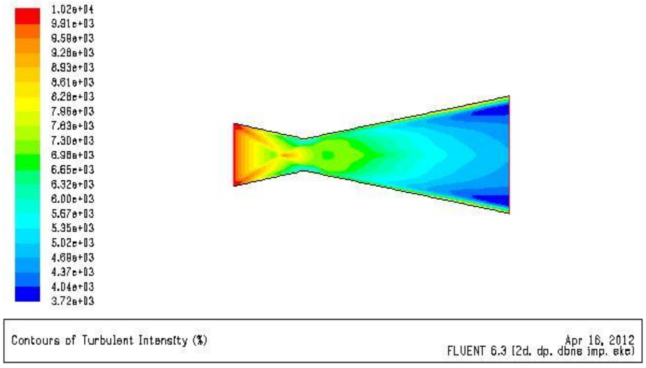
Fig. 3: Variation of Pressure, Temperature, Turbulence Intensity, Velocity Magnitude at 7.21^0 Divergence Angle

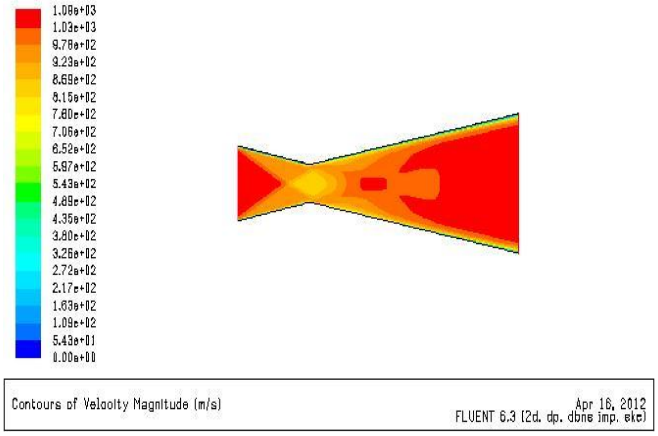


Contours of Static Pressure



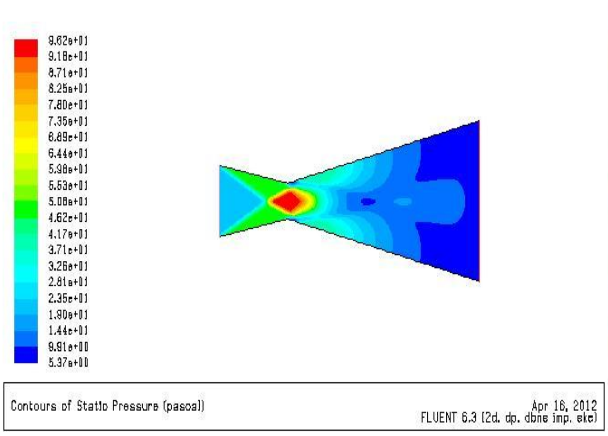
Contours of Static Temperature

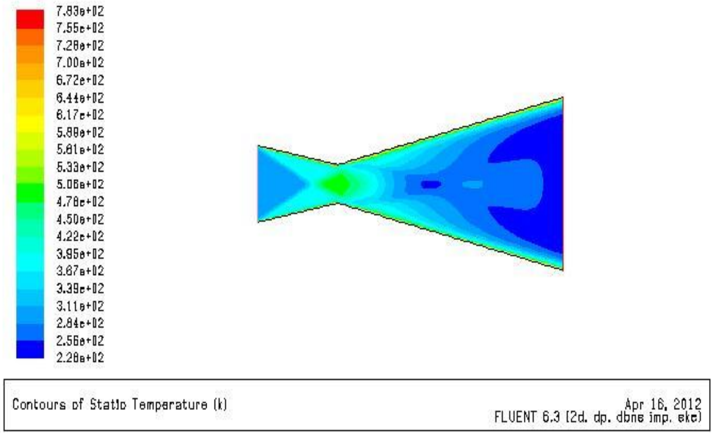




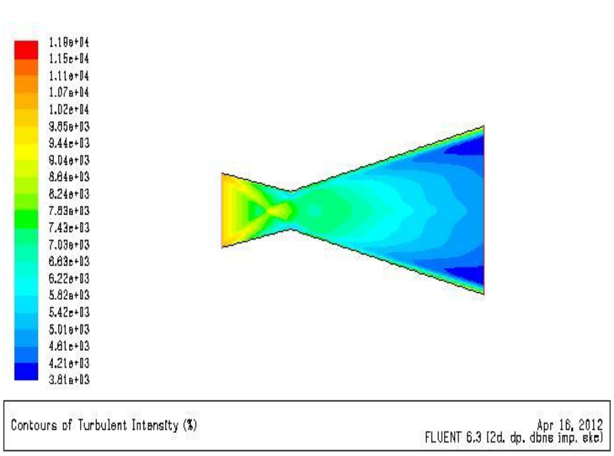
Contours of Turbulence Intensity **Contours of Velocity Magnitude**

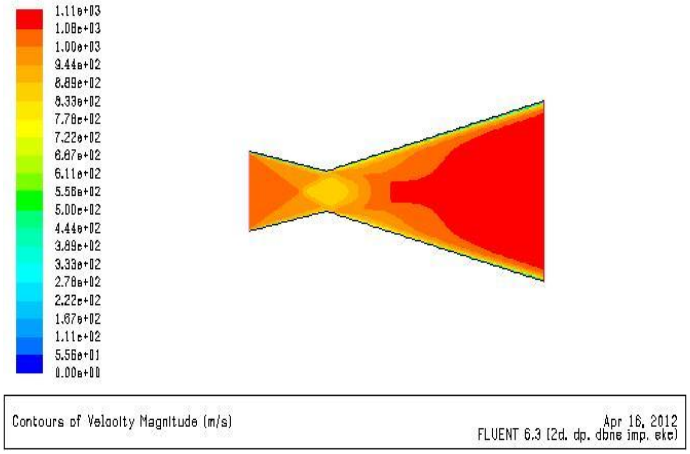
Fig. 4: Variation of Pressure, Temperature, Turbulence Intensity, Velocity Magnitude at 20^0 Divergence Angle





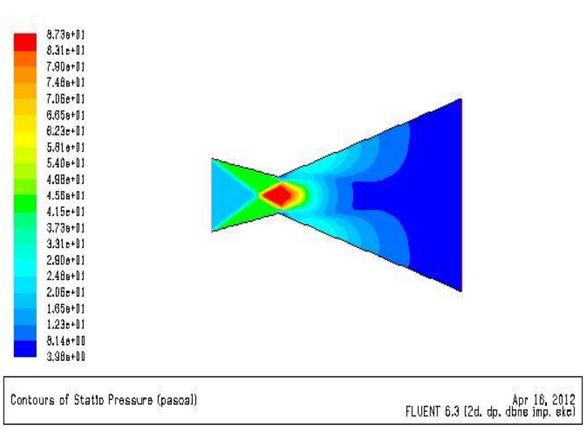
Contours of Static Pressure **Contours of Static Temperature**



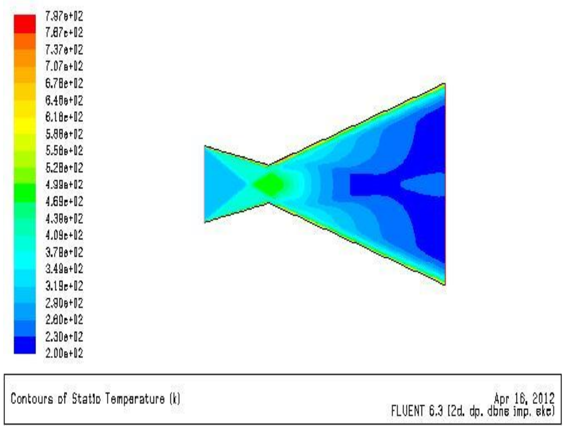


Contours of Turbulence Intensity **Contours of Velocity Magnitude**

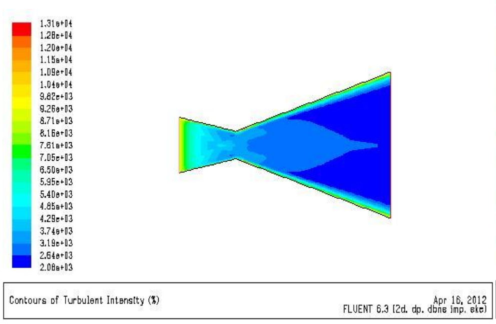
Fig. 5: Variation of Pressure, Temperature, Turbulent Intensity, Velocity Magnitude at 30^0 Divergence Angle



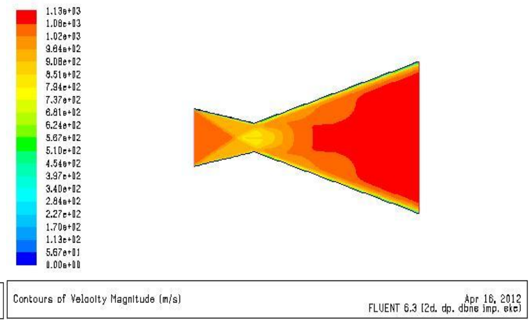
Contours of Static Pressure



Contours of Static Temperature



Contours of Turbulent Intensity



Contours of Velocity Magnitude

Fig.6: Variation of Pressure, Temperature, Turbulent Intensity, Velocity Magnitude at 40^0 Divergence Angle

CONCLUSIONS

Based on the results obtained, the following conclusions are drawn for the flow through conical nozzle. The flow characteristics behave differently at different divergent angles. At 7.21º divergent angle the maximum pressure in the nozzle which is obtained to 120 Pa. Whereas at 40^0 divergence angle the temperature is increased to 787 ^{0}k, the turbulent intensity of the nozzle increased to 13100 m^2/s^2 and the velocity magnitude of the nozzle increasing to 1130 m/s.

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