DESIGN OF NITROGEN-TETROXIDE / MONOMETHYL-HYDRAZINE THRUSTER FOR UPPER STAGE APPLICATION

Noor Muhammad Feizal B Muhalim and Subramaniam Krishnan
Department of Aeronautical Engineering, Faculty of Mechanical Engineering
Universiti Teknologi Malaysia, 81310 Skudai, Malaysia

Keywords: spacecraft propulsion / liquid rocket engine design / contoured nozzle

CONTENTS

ABSTRACT

Nitrogen tetroxide (N_2O_4) is a high density storable oxidizer. For many space applications, monomethyl hydrazine (MMH) and N_2O_4 are frequently adopted hypergolic propellant-combinations. Although N_2O_4 and MMH are toxic, because of their high specific impulse, extreme storage stability, and hypergolic nature, the combination is extensively used in orbital maneuvers, reaction controls, and launch vehicle propulsion. The present study details the design procedure for a 450-N and 3600-s reaction control rocket engine using N_2O_4 and MMH. The restartable engine should be capable of producing short thrust pulses of varying thrust levels. By using NASA CEC71 complex chemical equilibrium program the optimum oxidizer to fuel ratio is calculated to be 1.69. The procedures to configure the combustion chamber and propellant tanks are detailed. The propellants are pressurized by helium under blow down mode. A bell shaped nozzle contour with area ratio of 375 is adopted.

NOMENCLATURE

$ACAT$	= chamber nozzle-throat contraction ratio
C_F	= thrust coefficient
d	= diameter (m)
F	= thrust (N)
FS	= factor of safety

I_{sp}	= specific impulse (N-s/kg)
m	= mass (kg)
$\dot{m}$	= mass-flow rate (kg/s)
OF	= oxidizer to fuel mass ratio
P_c	= combustion chamber pressure
P_T	= tank pressure (MPa)
R	= radius (m)
t	= thickness (m)
V	= volume; tank volume (m^3)
Δt	= thrusting time (s)
ε	= nozzle area ratio
$\eta_{I_{sp}}$	= specific-impulse efficiency
ρ	= density (kg/m^3)
σ_y	= yield strength (MPa)

Subscripts

e	= nozzle exit
exp	= experimental estimated
F	= final
f	= fuel
g	= pressurizing gas
I	= initial
o	= oxidizer
p	= propellant
t	= nozzle throat

1.1 INTRODUCTION

Small thrusters adopting liquid propellants are used to provide attitude control, to align a satellite for a mid-course corrective or terminal maneuver, for rendezvous with another spacecraft, or to stabilize the spacecraft after separation from its stage. The most widely used propellant combination for such applications is nitrogen-tetroxide (N_2O_4) and monomethyl-hydrazine (MMH, CH_6N_2). Some of the upper stage rocket engines that use N_2O_4/MMH are the Orbital Maneuvering System used on all Space Shuttles and Tangential Injection and Rotational Combustion used on most of the classified satellites of USA. Typical thrust produced by satellite propulsion-systems is in the range of 400N to 900N.

N_2O_4/MMH is frequently used as a propellant combination for the propulsion of upper stages and spacecraft because of its higher specific impulse, hypergolic nature, and extreme storage stability. The hypergolic nature removes the need for an external ignition system and its extreme storage stability is ideal for spacecraft applications. The main drawback of N_2O_4 and MMH is their toxicity. The properties of these propellants are given in Table 1.

The purpose of this study is to draw out a design methodology for an upper stage liquid propellant rocket engine using N_2O_4/MMH propellant combination. An upper stage rocket engine producing typical thrust with gas pressurizing propellant feed system is taken for the study. The sketch of the rocket engine with its main features is given in Fig. 1.

1.2 DESIGN

The rocket engine is to deliver thrust in pulse mode with a nominal thrust of 450N. The total thrusting time is chosen to be 3600 seconds. The engine propellant feed system is helium gas pressurized one of blowdown type. Generally, the nominal chamber pressure for N_2O_4/MMH space propulsion systems is around 1MPa. Therefore with the chamber pressure of 1MPa, allowing the pressure drop across the injector as 0.7MPa, and some pressure loss across the on-off valve, the propellant tank nominal pressure is 1.8MPa. The required feed pressure range is chosen to be from 3MPa to 0.9MPa. Area ratio of 375 is chosen for the engine based on the average area ratio values of the upper stage engines in operation under this class.

Table 1 Properties of N_2O_4 and MMH propellant combination

Details	Values
Propellant combination	
Optimum *OF* ratio	~1.6
Combustion Temperature	3385 K
Density	1200 kg/m^3
I_{sp} (sea level)	2825N-s/kg
I_{sp} (vacuum)	3296 N-s/kg
N_2O_4	
Density	1431 kg/m^3
Freezing temperature	11°C
Boiling temperature	21°C
MMH	
Density	874 kg/m^3
Freezing temperature	-52°C
Boiling temperature	87°C

1.2.1 Choice of Oxidizer to Fuel Ratio

A fuel-rich oxidizer-to-fuel ratio (OF) in a bipropellant combustion chamber will yield a maximum specific impulse and this ratio is called the optimum mixture ratio. Using CEC71 program of NASA [Gordon and McBride, 1971], values of the specific impulse, I_{sp} for various OF ratios are obtained for both frozen and equilibrium flows at vacuum conditions and are plotted in Fig. 2.

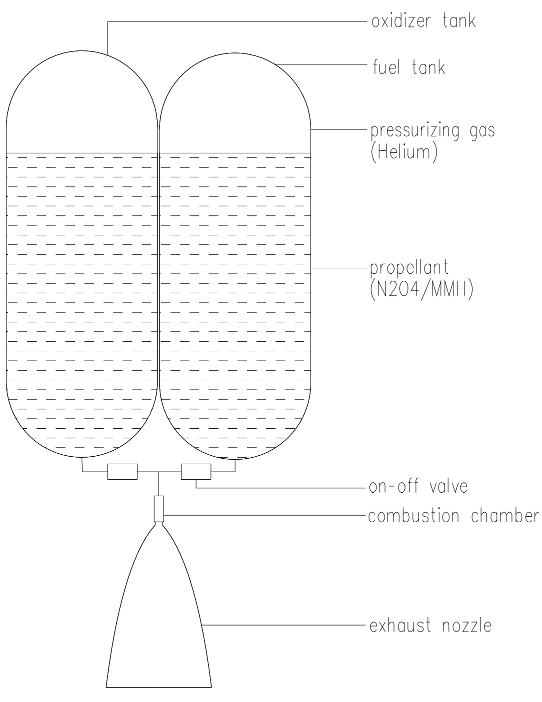

Fig. 1 Gas pressurized N_2O_4/MMH liquid propellant rocket engine.

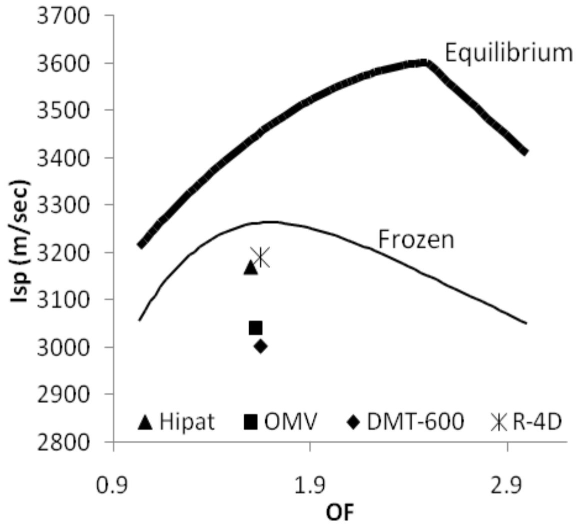

Fig. 2 Variation of vacuum specific impulse with OF ratios; Hipat, OMV, DMT-600, and R-4D are the N_2O_4/MMH upper-stage / spacecraft engines in operation.

The optimum OF ratio calculated is 1.69 for frozen flows and 2.46 for equilibrium flows. Hipat, OMV, DMT-600, and R-4D are the N_2O_4/MMH upper-stage / spacecraft engines in operation and the OF ratios adopted in these engines are also shown in Fig. 2. It is seen that the standard practice in industry is to adopt the optimum OF ratio of frozen flow assumption. The typical OF ratios adopted vary from 1.6 to 1.65. Nevertheless, in order to size the engine in the present study, the optimum OF ratio of 1.69 obtained through the theoretical calculations is adopted. The combustion chamber propeties calculated for the optimum OF ratio are given in Table 2.

Table 2 Propulsion parameters at the optimum *OF* ratio

Details	Values
Optimum OF ratio	1.69
Specific impulse – frozen (sea level)	3221.0 N-s/kg
Specific impulse – frozen (vacuum)	3264.6 N-s/kg
Thrust coeficient	1.892
Charateristic velocity	1703 m/s

1.2.2 Propellant Tank Design

The propellant tanks contain both the propellants and the pressurizing gas. The amount of pressurizing gas required depends on the amount of propellant to be expelled from the tanks.

1.2.2.1 Propellant Volume

The propellant volume required is determined using the calculated specific impulse value (Table 2). To arrive at an estimated experimental specific impulse, a conservative value of 0.98 is assumed for I_{sp} efficiency, $\eta_{I_{sp}}$. Thus the estimated experimental specific impulse value is 3199.31 N-s/kg. To produce a thrust of 450N the propellant flow rate,

$$\dot{m}_p = \frac{F}{I_{sp}} = 0.14066\ kg / s \tag{1}$$

The fuel flow rate is given by

$$\dot{m}_f = \frac{\dot{m}_p}{1+OF} \tag{2}$$

Hence, $\dot{m}_f = 0.0523\ kg/s$ and $\dot{m}_o = \dot{m}_p - \dot{m}_f = 0.0884\ kg/s$. For the total thrusting time of 3600 seconds, the total mass of fuel, $m_f = \dot{m}_f \Delta t = 188.2\ kg$ and total mass of oxidizer, $m_o = \dot{m}_o \Delta t = 318.1\ kg$. The volume of the fuel is then, $V_f = m_f / \rho_f$. For the values of $\rho_f = 874\ kg/m^3$ (Table 1), $V_f = 0.215\ m^3$. Adopting the same procedure, with $\rho_o = 1431\ kg/m^3$ (Table 1), the required oxidizer volume $V_o = 0.222 m^3$. In order to include some losses of propellants trapped in on-off valve and tubing, both the fuel and oxidizer volumes are increased by 5% and therefore the required propellant volumes are, $V_f = 0.226\ m^3$ and $V_o = 0.233\ m^3$.

1.2.2.2 Pressurizing Gas Volume

In the gas pressurization, the propellants can be pressurized by having a separate pressurization tank. But for a single thruster operation operating essentially with pulse modes, gas pressurization is adopted within the propellant tanks. This arrangemt known as blow down mode will result in reducing tank pressures as propellants are being consumed. However, the required impulse-pulse for correction ($F\Delta t$) can be given by the increased Δt as the thrust F falls with the reduction in tank pressures. Thus, for this upper stage rocket engine design, the pressurizing gas is included in the propellant tank itself. The required pressurizing gas volume is calculated by assuming that the gas expansion process in the tank is isothermal. Thus in order to completely expel all the fuel from its tank, the required pressurizing gas initial volume in the fuel tank,

$$Vg_{fI} = \frac{\left(P_{T_F}/P_{T_I}\right)Vf}{\left(1 - P_{T_F}/P_{T_I}\right)} \tag{3}$$

For $P_{T_F} = 0.9\ MPa$ and $P_{T_I} = 3\ MPa$, the required pressurizing gas initial volume is $V_{g_{fI}} = 0.097\ m^3$. Similarly the required pressurizing gas volume in the oxidizer tank,

$$Vg_{oI} = \frac{\left(P_{T_F}/P_{T_I}\right)V_o}{\left(1 - P_{T_F}/P_{T_I}\right)} \tag{4}$$

For the same values of P_{T_F} and P_{T_I} used previously, the required pressurizing gas initial volume in oxidizer tank is $V_{g_{oI}} = 0.1\ m^3$. Therefore the calculated tank volumes of fuel and oxidizer are $0.323\ m^3$ and $0.333\ m^3$ respectively. To simplify the engine system design, volumes of both the tanks are rounded off to $0.333\ m^3$.

1.2.2.3 Propellant Tank Materials

The materials used for both the fuel and oxidizer tanks is aluminium alloy 7075. Aluminium alloy is used because it is compatible with N_2O_4 as well as MMH. The alloy has a high strength to weight ratio and is resistant to corrosion. Mechanical properties of Aluminium 7075 are given in Table 3 [Bray, 2007].

1.2.2.4 Propellant Tank Configuration and Dimension

Propellant tank configuration is chosen based on the space available in the launch vehicle or the spacecraft. Spherical shape will result in large cross-section and should be accetable to the space provided in a lauch vehicle stage. In spacecraft spherical and other odd shapes suitable for the available space can be adopted. However, in the present study a cylindrical shape is chosen for the propellant tanks. The fuel tank and oxidizer tank internal dimensions are as shown in Fig. 3. The tank wall-thickness is,

$$t = \frac{P_{T_I} R}{\sigma_y} FS \tag{5}$$

For the safety factor, $FS = 1.3$ and tank radius, $R = 0.25\,m$ the required fuel and oxidizer tank thickness is 1.93 mm. This is rounded off to 2mm.

Table 3 Aluminium 7075 mechanical properties

Details	Values
Tensile strengh	570 MPa
Yield strengh	505 MPa
Modulus of elasticity	72 GPa
Density	2810 kg/m^3

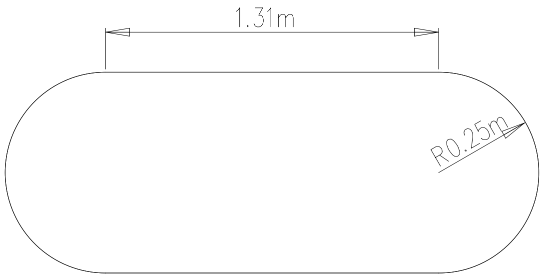

Fig. 3 Fuel tank and oxidizer tank internal dimensions.

1.2.3 Thrust Chamber Design

There are four types configurations used for the combustion chamber. These are spherical-, near-spherical-, pear-, and cylindrical-shape. The spherical shape is the best in terms of weight and heat transfer but is not easy to manufacture. Therefore, the classical cylindrical combustion chamber is adopted in the engine design.

1.2.3.1 Thrust Chamber Dimension

The experimental thrust coefficient $C_{F_{exp}}$ is estimated by multiplying the theoretical thrust coefficient, C_F with an efficiency factor of 0.98. Thus the estimated experimental thrust coefficient value is 1.8542.

To produce 450 N of thrust with chamber pressure of 1MPa, the throat diameter is calculated as,

$$d_t = \sqrt{\frac{4F}{\pi C_{F_{exp}} Pc}} = 0.018\,m \tag{6}$$

For the area ratio, $\varepsilon = 375$, the exit diameter is, $d_e = d_t\sqrt{\varepsilon} = 0.34\ m$.

The ratio of the chamber area to throat area is called contraction ratio, $ACAT$ and its values depend strongly on the throat diameter, propellant combinations and chamber pressure. Normally, for upper stage rocket engine producing thrust around 400N-450N, the typical contraction ratio values is in the range of 2-2.5 [Huzel and Huang, 2004]. The adopted contraction ratio $ACAT = 2.5$. So, for $d_t = 0.018\ m$, the chamber diameter is $d_c = d_t\sqrt{ACAT} = 0.028\,m$.

As with the contraction ratio, there is no perfect rule in determining the chamber length values that can deliver the highest thrust for given chamber pressure and area. The best method is by conducting experiment. But this is costly and impractical especially during the initial design phase. Huzel and Huang [Huzel and Huang, 2004] have gathered data on chamber length for various throat diameters for liquid propellant rocket engines with chamber pressures around 35 bar. Figure 4 shows the relation between the chamber length and the throat diameter. The rounded chamber-length determined from Fig. 4 is 85 mm.

1.2.3.2 Thrust Chamber Material

Thrust chamber is exposed to extreme heat and pressure during thrusting and thus a special high temperature material must be used in order to avoid catastrophic failure of the thrust chamber. Examples of high temperature metal alloys are those of columbium (or niobium), molybdenum, iridium, rhenium, tantalum, and rhodium. For upper stage engine, the most used metal alloys are of columbium type [Gerardi, 2007]. It is because columbium is relatively light in weight and of high strength at elevated-temperatures. Columbium alloy C-103 is chosen as the thrust chamber materials and its properties are given in Table 4 [Gerardi, 2007].

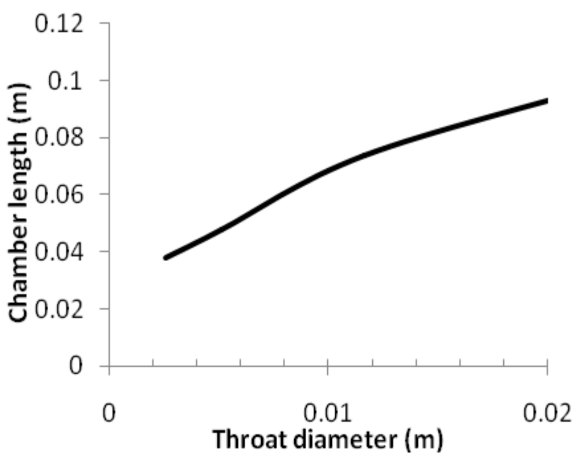

Fig. 4 Relation between chamber length and throat diameter; adopted from [Huzel and Huang, 2004].

1.2.3.3 Combustion Chamber Wall Thickness

The required combustion-chamber wall-thickness is,

$$t_c = \frac{P_c R_c FS}{\sigma y} \tag{7}$$

where, the chamber pressure is taken as the highest chamber pressure value of $P_c = 2.2\ MPa$. So for $FS = 3$, the required chamber wall thickness, $t_c = 0.0012\,m$. The thickness is estimated using simple thin wall pressure vessel theory which does not include stresses due to the thermal loading and hence may not be sufficient. More elaborate analysis that includes both the internal pressure loads combined with high temperature conditions must be made in order to determine the combustion chamber thickness. But for the time being, the wall thickness is taken as 0.002m.

Table 4 Columbium C-103 mechanical properties at various temperature and nominal physical properties

Temperature (°C)	Tensile Strengh (MPa)	Yield Strengh (MPa)
RT	725	660
1095	235	160
1370	90	76

Physical Properties	
Details	Values
Density	8870kg/m^3
Thermal expansion coefficient	8.10 $\mu m/m-K$
Thermal conductivity	41.9W/m-K

1.2.3.4 Exhaust Nozzle Contour

Since the area ratio, $\varepsilon = 375$ is high, the use of conical shape nozzle is not suitable since the exhaust nozzle will be long and heavy. Thus, bell shaped nozzle is used for the engine. The bell shaped nozzle is designed based on the method developed by Rao [Rao, 1958]. Since the nature of the method involves tedious mathematical calculations and iterations, a computer program has been written using FORTRAN 90 language. The program is capable of designing optimum exhaust nozzle contour for a given length and area ratio. Figure 5 shows the contour of the nozzle divergent-portion designed using the program and Fig. 6 is the completed thrust chamber diagram.

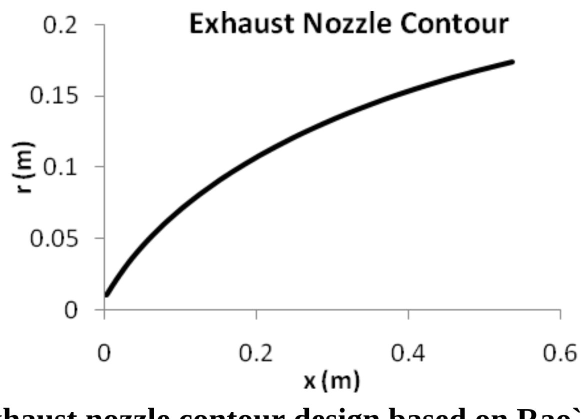

Fig 5 Exhaust nozzle contour design based on Rao`s methods.

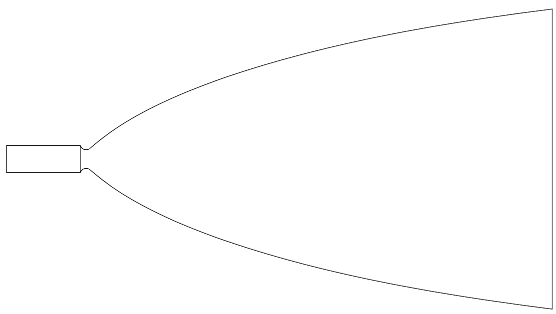

Fig. 6 The thrust chamber diagram.

Table 5 Mass estimations

Details	Values (kg)
Propellant tanks	
Oxidizer tank	16.5
Fuel tank	16.0
Propellants	
N_2O_4	318.1
MMH	188.2
Helium pressurizing gas	1.0
Engine	
Engine	5.4
Wet weight (fully loaded)	545.7
Dry weight	38.4
Thrust to mass ratio	83.33N/kg

Table 6 Charateristics of the rocket engine design

Details	Values
Nominal thrust	450N
Thrusting time	3600 sec
Pressure chamber	3-0.9MPa
Specific impulse (vacuum)	3199.3 N-s/kg
Thrust coeficient	1.8542
Charateristics velocity	1691.93 m/s
Area ratio	375
Contraction ratio	2.5
Throat diameter	0.018m
Exit diameter	0.338m
Chamber diameter	0.028m
Fuel tank volume	0.333 m^3
Oxidizer tank volume	0.333 m^3
Fuel mass	188.2kg
Oxidizer mass	318.1 kg
Thruster length	0.650m
Exhaust nozzle length	0.536m
Combustion chamber length	0.085m

1.2.4 Mass Estimation

The mass-values of propellant tanks and engine are estimated based on their densities, geometries, and dimensions (Table 5).

An engine in operation of similar specification is the HiPAT developed by Aerojet. It has an engine mass of 5.2kg. The estimated mass of the engine of the present study is 5.4kg. This shows that the present design is reasonably on par with an engine in operation. However, the present design is only priliminary that may be used for first cut sizing or detailed analysis.

1.3 RESULTS AND DISCUSSION

The liquid propellant rocket engine taken for the study has been designed covering all the major components. The final envelope size of the engine is 0.65m length by 0.34m wide which are typical values for this class of engine. The engine specific impulse is estimated as 3199 N-s/kg which is quite high compared to other similar engines of the same class. This is because, the value is the one theoretically calculated. The charateristics of the rocket engine are listed in Table 6. The sized propulsion sytem is given in Fig. 1.

1.4 CONCLUSIONS

A design procedure for the upper stage liquid propellant rocket engine has been explained for the first cut sizing or for detailed further analysis. The theoritical performance of the combustion chamber is determined using a widely adopted NASA program and the performance values are used to do sizing of the propellant tanks and the thrust chamber dimensions. The combustion chamber design is based on published experimental data and the final combustion chamber size and mass resonably match with the values of similar engines in operation. A computer program to design the exhaust nozzle contour based on Rao`s method also has been used to design the nozzle.

REFERENCES

Bray, J. W., Aluminium Mill and Engineered Wrought Products, ASM Handbook, Vol. 2, pp. 27-61, ASM International, 2007.

Gerardi, S., Refractory Metals and Alloys (Niobium), ASM Handbook, Vol. 2, pp. 565-571, ASM International, 2007.

Gordon, S. and McBride, B. J., Computer Program for Calculation of Complex Chemical Equilibrium Composition, Rocket Performance, Incident and Refleted Shock and Chapman-Jouguet Detonations, NASA SP-273, 1971.

Huzel, Dieter K. and Huang, David H., Modern Engineering for Design of Liquid Propellant Rocket Engine, Washington, DC:, America Institute of Aeronautics and Astronautics, 2004.

Rao, G. V. R., Exhaust Nozzle Contour for Optimum Thrust, *Jet Propulsion,* Vol 28 (6), pp. 377-382, 1958.