УДК 629.7

RAJESH N. MUNESHWAR[1], AJAY VIKRAM SINGH[2]

[1]***National aerospace university named after Zukovsky «KhAI», Ukraine***
[2]***Golden eye security systems , near shastri nagar , Gaziabad , Uttar Pradesh, New Delhi, India***

PRESSURE FEED WITH GAS GENERATOR UPPER STAGE ENGINE FOR A HYPERSONIC SUBORBITAL CUBE-SAT LAUNCH VEHICLE AND SOUNDING ROCKET "SHIVASTRA-1"

An increasing trend , Low cost space access with Microsatellites, and so the need of an efficient, simple, reliable equally cost effective Carrier was considered with rapid designing and fast prototyping design concept so the small Liquid Propellant Rocket engine with "solid propellant gas generator" pressure feed system is resulted. The abstract discusses the perspective design of the upper stage of the Rocket "Shivastra-1". The design features a co-axially arranged propellant tankage fed with burning a cordite powder charge yielding a rapid initial acceleration and 3 M burn out velocity providing 10KN thrust at 30 km enough to reach 100 -180 km altitude carrying 2 kg of payload. The combustion chamber wall is enameled, protecting from high temperature of combustion. The preliminary design work is performed with CATIA-V5 and was optimized on ANSYS. The results obtained were optimistic for the static test fire on test stand providing a perspective design of a reliable Liquid Rocket Engine.

Key words: *Rocket, Shivastra, solid propellant, gas generator, Sounding Rocket, Upper stage, Microsatellite, Cub-Sat.*

Introduction

At the advent of the 21st century revolutionary discoveries in microelectronics and dual use technologies caused space technology quite accessible for the common civilians and space enthusiasts all over the world. This poised new challenges to them, the most fundamental was Building a less expensive, reliable and simple Rocket vehicle.

This question was considered in the framework of engineering design concept rapid designing and fast prototyping. The main objective of the work was to develop a simple and affordable propulsion system that could easily carry a Nano-satellite weighing no more than

2 kg and loft at the altitude of 100 – 250 km. The pursuit of the objective resulted into the so called "MARUT" propulsion system using hyperbolic propellants Tonka-250 and 97% Nitric acid for the upper stage of the SHIVASTRA-1 Rocket vehicle. It employs solid propellant gas generator which burns 400 gm. of "cordite" powder charge, the gaseous products thus produced is directly impinged into the propellant tanks forcing the propellants in to the combustion chamber. The pressure developed in the chamber is about 5MP. The bell shaped nozzle is optimized for performance at altitude form 30 km to 80 km. the main feature of the engine is it's co-axial arrangement of tankage system and use of throttle valve inside the combustion chamber holding the flow of oxidizer up to 1/10 of its flow at it's initial, which allows to reduce the burning time up to 3 sec. in the result the specific impulse 312 s was produced providing 2200 m/s of burnout velocity of the system, with more than 80 g initial acceleration producing 10 KN of thrust compensating the resulting high atmospheric drag and with booster phase assembly the initial velocity of 2.5 M was added to the system. It means the total velocity at the burnout will be more than 4 M that is somewhat higher than the required velocity of 1114 m/s of Delta-V from 40 km altitude. And so the Rocket Shivastra-1 roars into the space.

This paper presents the design of the "MARUT" Engine for the upper stage as the solution of the problem to reach space frontier carrying the microsatellites or the scientific payloads for upper atmospheric studies.

1. Analyzing the problem of necessary ΔV for adaption of the Required Nozzle

Exit area (Ae/At) for known expansion ratio (Pe/Pc) for isentropic flow (ε_e):

$$\varepsilon_e = \frac{Ae}{A_t} = \frac{\Gamma}{\left(\frac{P_e}{p_t}\right)^{\frac{1}{\gamma}} \cdot \sqrt{\frac{2\gamma}{\gamma-1}\left[1-\left(\frac{Pe}{Pc}\right)^{\frac{\gamma-1}{\gamma}}\right]}}$$

value ε_e depends above all on (Pc/Pe). Adaption of nozzles at high altitude involves high expansion ration Pc/Pe. Increase in Ae increases thrust but also nozzle weight, so it was necessary to compromise between Ae and Nozzle weight. Condition of Adaption (ε_e):

Condition of exit area is very important. It makes possible to select optimum nozzle for a given flight program. A function of type $\varepsilon_e \times$Pe= constant is used to elate values of e and Pe. This expression is valid whatever theory contemplated (γ constant throughout the frozen flow, equilibrium flow) or nature of propellant. It is very useful expression and allowed us to choose very quickly the value of ' ε_e ' best adapted for our altitude with conforming to requirements to avoid flow separation. Gain in velocity of Rocket resulting from compromise between weight of divergent and thrust amounts to several percent is especially important when weight of divergent can no longer be neglected. This gain in velocity depends on value of parameter A/C. once diameters of Dc, Dt, and De are calculated it is necessary to determine shape of convergent and divergent. Nozzle is becoming bigger in modern Rocket's motors. So it became necessary to select the nozzle that could ensure the suitable flow, but also should help the important recombination reactions in case of high performance propellants.

Gain in velocity for adaption of area ratio AC/AT:

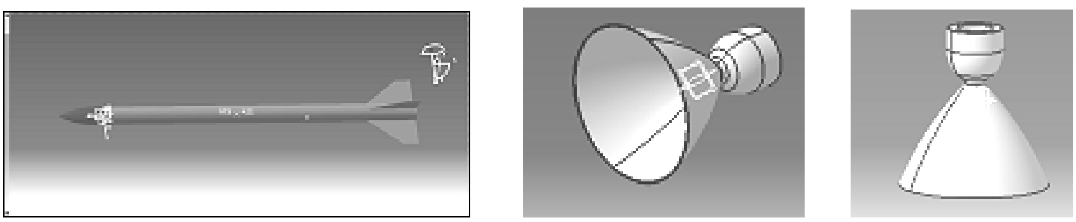
ΔV=go (Is $\log(1/1-\upsilon)$-mp/$\dot{m}$), where, go= standard gravitation acceleration, Isp = specific impulse , υ = mass of propellant/ initial mass of rocket and mp/ $\dot{m}$ = time of burning. Since,in our case of engine nozzle mp/ $\dot{m}$ = 5s , υ=0.56 increase in speed ΔV for one stage is 1041.17 m/s and the initial velocity provided by the Ramjet booster stage of the "SHIVASTRA-1" vehicle 2.8 M will be enough for the Rocket to reach into the space.

2. The propulsion system

The thrust chamber, propellant tankage, solid propellant gas generator forms the propulsion system. It uses hypergolic propellants Tonka-250 as fuel and 97% HNO3 and 3% H2O as oxidizer. The propellant is accommodated into the annularly arranged tanks one inside the other, the outer tank also forms the outer body of the rocket is made up of drawn mild steel. The nozzle is optimized to operate from 30-100 km altitude is bell shaped. The expansion ratio used is 500. Combustion chamber is of tubular geometry made up of stainless steel 20X18H10C2 (ГОСТ1050-88).Instead of the cooling system it uses Enameling insulation coating on inner chamber walls. Injector head is simply a plate with concentric injector orifices. The solid propellant gas generator is of supersonic type that burns a cordite charge, the gases produced forces the fuel and oxidizer initially restricted by burst discs at 1800 lb/sq. in-22000 lb/sq.in into the combustion chamber. The flow is initially partly throttled by expendable valve.

2.1. Bell shaped nozzle

The Nozzle for the required mission has to function from 30 km to 100 km. the atmospheric pressure is too low than that of the sea level so the expansion ration considered was optimized for these altitude and □ = 500 provided the acceptable value of the exit diameter of the nozzle 23 cm. approaching the altitude of 60 km the nozzle will function with maximum specific impulse beyond this the pressure thrust will decrease but the velocity thrust will continue to grow till the burnout.



a b c

Fig. 1. Shivastra Rocket stage and Thrust Chamber Views:
a – Shivastra -1 Rocket stage; b – Thrust chamber-1; c –Thrust chamber-2

2.2. Thrust chamber, propellant tankage and component details

FUEL:

The propellants used in Marut engine are Tonka-250 and HNO3 97% whereas Tonka-250 is a mixture of and HNO3 is largely a fuming nitric acid and added with small percent of sulphuric acid to reduce the corrosion of the tankage etc.

Propellant tanks:

The inner tank is 45.3 inch long and is of light alloy construction. The parallel portion is made up of sheet 2 mm thick rolled and welded; the welds being filled off smooth. The ends which are welded to the parallel part and doomed and flanged. The doomed end is secured to the spare injector at the tail and at the nose to the perforated plate each by 14 small studs.

The tube which forms the body of the rocket and outside of the outer tank is of drawn mild steel 2 mm thick and 45.25 in long. It is welded at each end to a turned and screwed ring over which it fits. The two rings are similar and screwed at one end into the combustion chamber ring (thus clamping the bursting disc and spray injector), and at the other end into a coupling ring. It thus clamps the perforated locating disc, the solid cartridge chock plate and the spun out edge of the solid cartridge container against the head. In operation fuels in the tanks are expelled by gas pressure from the powder charge. The hot gases from this impinge directly on the fuels there being no pistons in the tanks. Channeling of the fuels due to gas is probably small due to the high acceleration of the rocket (60g approx.)

Solid expulsion charge and container:

The latter is a cylindrical mild steel container closed by the chock fitting over one end and fitting over one end and held in place between the head and forward junction ring of the outer fuel container. It contains a small cylinder of solid propellant cordite type which wrapped with paper with two ends being exposed. This is ignited electrically by leads sealed through the forward end of the container and burns at the two exposed flat ends 2400 lb/sq.in. The gas so formed leaves via the chock orifice and pressurizes the fuel tanks. At the forward end of the expulsion charge container a fuze is screwed in which on assembling the rocket projects into the cavity at the rear end of the payload.

Bursting discs:

The double fuel tanks are sealed at each end by single sheet of thin aluminum which also serves as bursting discs. Sheering of bursting discs takes place at the circumference of the hole in the flanged end of the inner tank, and around the outer circumference of the flange. The outer circumference of the disc at the front end is clamped between the junction ring of the body and perforated plate and at the rear end by the spray injector; sealing of the joints is obtained by v'ing the bearing surfaces.

Combustion chamber and spray injection:

Fuel injection:

The fuel and oxidants are injected into the combustion chamber through spray injection plate containing a number of holes arranged in two concentric circles, and impinge on the walls of the chamber. Opposing holes do not necessarily impinge at exactly the same point. The correct ratio of fuel and oxidant flow is obtained approximately by using holes of 1.5 mm diameter in one ring and 2.8 mm diameter in the other.

Expendable valve:

Foreseeing the trouble in obtaining quick ignition of the fuels and the combustion chamber might blow up when ignition is delayed. This can be overcome by the arrangement of a special expendable valve which initially blocks up most of the venturi throat and at the same time throttles the flow of oxidant from the inner tank to about 1/10 of the normal flow. On firing the solid expelling charge (which reaches its full pressure in 1/10 s from the point of ignition) a very fuel rich mixture enter the combustion chamber and the pressure rises to about 10 or 20 atmosphere. During this time thrust is not sufficient to move the rocket. At about this pressure ratio the valve is blown out, opening the venturi throat to its full area and at the same time allowing the correct fuel mixture to enter the chamber. The time from the instant until the full chamber pressure is reached is only 1/100 sec. a fact which gives the rocket a very high initial acceleration. The valve is held in by split circlip ring. The valve is initially jammed in. apparently this gives reproducible results with regards to valve discarding pressure.

Construction of combustion chamber and ventury:

The combustion chamber which is of light construction and enameled inside is made of seamless tube. The venturi can be formed by spinning while hot. The inside in this case had to be enameled and the chamber is expected to last for 5 sec on test stand.

The combined combustion chamber and venturi used in this rocket is extremely light and efficient when compared to other rockets of American designs. Full use has been made of the fact that with such small burning time steady temperature conditions are not reached and the venturi does not burn out. The manufacture of the chamber and venturi in one piece, it's cleanness of line giving the gas a smooth flow out, and for the way it lends itself to giving the venturi a correctly varied thickness along its length has much to recommend it.
Another factor which permits a very light design of venturi to be used is the absence of erosion caused by solid particles in the exhausted gas.

Conclusion

The engine "Marut" has a big commercial potential as a microsatellite launch vehicle and a sounding rocket given that its simplicity of design and operation. The manufacturability from the commonly accessible material is its biggest cause of reason of being affordable. This reduces the launching cost up to several thousands of $$. Though the design is still in embryonic stage and the practical problems of variations in mass flow rates due to the different static pressure heads in fuel tanks caused due to the high acceleration, all it needs to be studied practically. The design has also several building blocks for its further innovation into an anti-aircraft missile.

It is recommended to study practically, the sloshing effects, the effects of difference in propellant mass flow rates. On static test firing on special test stands it also needs to be studied the effects of ignition delays in the combustion chamber and the chugging effect of oxidizer rich flow at the throat cross section.

Reference

1. Dobrovolsky M.V. Liquid Rocket Engines basic designing / M.V. Dobrovolsky. – M., 1968.

2. Sutton George P. Biblarz Oscar Rocket Propulsion elements / G.P. Sutton. – Edition-VII, wiley-interscience, California.

3. Завистовский Д.И. Конструкция и проектирование агрегатов жидкостных ракетных двигателей / Д.И. Завистовский, В.В. Спесивцев. – Х.. НАКУ «ХАИ», 2006.

4.Термодинамические и теплофизические свойства продуктов сгорания: Справочник В 10 т. / Под ред. В.П. Глушко –М.: АН СССР, 1971-1979. –Т. 5. – Р. 375-400.

Поступила в редакцию 30.04.2010

Рецензент: д-р техн. наук, проф., проф. кафедры Д.И. Завистовский, Национальный аэрокосмический университет им. Н.Е. Жуковского “ХАИ”, Харьков.

ВИШТОВХУВАЛЬНА СИСТЕМА ПОДАВАННЯ З ГАЗОГЕНЕРАТОРОМ ДЛЯДВИГУНА ВЕРХНЬОГО СТУПЕНЯ ГІПЕРЗВУКОВОЇ СУБОРБІТАЛЬНОЇРАКЕТИ-НОСІЯ ДЛЯ СУПУТНИКА CUB-SAT ТА ЗОНДУЮЧИХ РАКЕТ

Раджеш Н. Мунешвар, Аджай Вікрам Сінгх

Зростаючий інтерес, низька вартість та доступність мікросупутників і, як наслідок, потреба в ефективних, простих, надійних і рівно недорогих носіях призвели до швидкої розробки та створення конструкторських прототипів малого ракетного двигуна на рідкому паливі із виштовхувальною системою подавання з твердопаливним газогенератором. В даних тезах обговорюються перспективи конструювання верхнього ступеня ракети "Шівастра-1". Конструкція містить коаксіально розташовані паливні баки, які живляться згоранням порохового заряду кордиту з швидким початковим прискоренням та вихідною швидкістю продуктів згорання 3 М, що забезпечує тягу 10 кН на висоті 35 км. Це є достатнім для досягнення висоти 100-180 км при корисному вантажі 2 кг. Стінки камери згорання емальовані для захисту від високої температури згорання. Попередні конструкторські розробки виконано з використанням CATIA-V5 та оптимізацією з використанням ANSYS. Отримані результати дають надію на успішні наземні вогневі випробування на стенді, що призведе до створення надійного ракетного двигуна на рідкому паливі.

Ключові слова: ракета, Шівастра, тверде паливо, газогенератор, зондуючі ракети, верхня ступінь.

ВЫТЕСНИТЕЛЬНАЯ СИСТЕМА ПОДАЧИ С ГАЗОГЕНЕРАТОРОМ ДЛЯ ДВИГАТЕЛЯ ВЕРХНЕЙ СТУПЕНИ ГИПЕРЗВУКОВОЙ СУБОРБИТАЛЬНОЙ РАКЕТЫ-НОСИТЕЛЯ ДЛЯ СПУТНИКА CUB-SAT И ЗОНДИРУЮЩИХ РАКЕТ "ШИВАСТРА-1"

Раджеш Н. Мунешвар, Аджай Викрам Сингх

Растущий интерес, низкая стоимость и доступность микроспутников и, как следствие, потребность в эффективных, простых, надежных, а также недорогих носителях привели к быстрой разработке и созданию конструкторских прототипов малого ракетного двигателя на жидком топливе с вытеснительной системой подачи с твердотопливным газогенератором. В данной статье обсуждаются перспективы конструирования верхней ступени ракеты. Конструкция "Шивастра-1" включает коаксиально размещенные топливные баки, которые используют сгорание порохового заряда кордита с быстрым начальным ускорением и выходной скоростью продуктов сгорания 3 М, что обеспечивает тягу 10 кН на высоте 35 км. Этого достаточно для достижения высоты 100-180 км при полезной нагрузке 2 кг. Стенки камеры сгорания эмалированы для защиты от высокой температуры сгорания. Предварительные конструкторские разработки выполнены с использованием CATIA-V5 и оптимизацией с применением ANSYS. Полученные результаты позволяют надеяться на успешные наземные огневые испытания на стенде, что приведет к созданию надежного ракетного двигателя на жидком топливе.

Ключові слова: ракета, Шивастра, твердое топливо, газогенератор, зондирующие ракеты, верхняя ступень.

Rajesh N Muneshwar – 4th year student of Faculty Aircraft and Helicopter Designing and Manufacturing, National Aerospace University named after Zukovsky «KhAI», Xarkov, Ukraine, e-mail: hindaputra@gmail.com , r.n.muneshwar@mail.ru.

Ajay Vikram Singh – Aerospace and Marine Engineer, Vice President of Golden Eye Security System Ghaziabad, New Delhi, India, e-mail: ajay261@hotmail.com.