Integrated modeling and analysis for a LOX/RP1 Gas generator cycle based Rocket engine Combustion Chamber and gas generator Chamber: Focusing on Regenerative cooling of Coaxial shell thrust chamber

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Abstract— An integrated engine design and operation analysis methodology was created and employed to study a 25 kN thrust class Gas generator based LOX/RP-1 (Kerosene) Liquid Rocket engine (LRE) for use in upper stages of a Satellite Launch Vehicles (SLV). Major components of the Gas generator were modeled to produce a simplified dependence of their geometry on engine performance. The paper discusses a technique to determine a feasible operation point for LRE by varying engine operation parameters and geometry for a fixed thrust requirement. The engine design features a Regenerative cooling coaxial shell gas generator for application on Liquid Rocket Engines (LRE). The chamber cross section was considered uniform with respect to axial position along the engine. The analysis is aided by producing trends that determine heat transfer characteristics of the combustion chamber as a function of geometry and principal dimensions. For this purpose relationships obtained by earlier researchers are used to help create a simplified model of the combustion chamber. Characteristic curves are obtained for chamber pressures ranging from 215 to 725 psia for Liquid Oxygen and Rp-1 propellant at a varying oxidizer to fuel ratio (O/F) of 2.0 to 2.4. Characteristic curves are obtained for chamber pressures ranging from 300 to 500 psia for Liquid Oxygen and Rp-1 propellant configuration. Trends are obtained for a constant chamber pressure and on the basis of these trends a feasible operating point is obtained for operation.

I. INTRODUCTION

To date rockets are the only means of access to space and interplanetary missions. Rockets are one of the most amazing endeavors man has ever undertaken for exploration of space. A big part of increasing interest in the field owes to the complexity of the subsystems and their interdependence in development of an operational rocket engine. This has introduced vacancy of constant and ever expanding research in the field. Cryogenic rockets are sophisticated systems developed to obtain higher energy from combustion systems. This is usually translated in high values of specific impulse (~350s Vac) from these rockets. The propellant combination of Liquid Oxygen (LOx) and RP-1 (Kerosene) have proved to be the cheapest propellant for typical LRE missions. Notable examples of engines designed on the combination are Russian RD- series engines and American Merlin rocket engines.

The results and trend presented here are a result of a fully integrated MATLAB program developed to aid in design of a gas generator based LRE system. The program contains routines for engine performance, Injector design, heat transfer analysis and provides manufacturing design parameters of the designed engine. The results obtained were validated against existing data in literature. The program is run in a routine to obtain results for a number of data points from 15 bars to 50 bar combustion pressure at a varying mixing ratio of 2.0 to 2.4.

II. GAS GENERATOR CYCLE

The gas generator cycle ensures continuous reliable operation with reduced complexity compared to other closed rocket engine operation cycles. This makes these engines ideal for the renewed need for access to low earth orbit and beyond for typical space missions. The gas generator cycle shown in the adjacent figure is the simplest schematic of a pump-fed system operating with liquid propellants. The system features a finite number of restarts and is appreciated for their longevity and capability for throttling and potential of delivering higher specific impulse compared to the LRE volume and complexity. The system however involves additional components such as gas generators. The pressure requirements of engines operating on this cycle lie between Expander and staged combustion cycle operated engines.

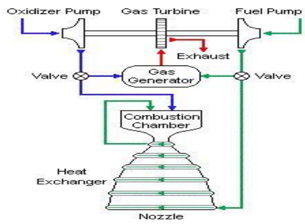

Figure 1: GAS GENERATOR CYCLE

III. LOX- RP1 PROPELLANT ENGINES

The combination of LOX-RP1 has been used to propel rockets for a very long time. The properties of the combination are readily available for use to researchers. The combination provides the cheapest means of propulsion to space with reliability of operation. However the RP1 propellant has a tendency of coking under high temperature. The affect results in producing a layer of solid carbon deposits on the chamber and nozzle walls which further restrict heat flow through the chamber walls. The propellant combination produces a specific impulse of around 280 to 310 s at sea level.

IV. ANALYSIS METHODOLOGY

The design of a rocket for a given thrust requirement and chamber pressure is relatively straightforward. Following methodology is adapted to obtain the design attributes of the engine.

The first step for analysis is to set requirements of the desired LRE. It is carried out by fixing the chamber pressure, mixing ration (O/F) and force required by the engine.

Parameters for LRE baseline design:

An initial guess to system specific impulse is made to aid in the design process and mass flows are obtained through the main engine nozzle by use of the following relation

$$\mathrm{c} = \mathrm{I_s g_o} = \frac{\mathrm{F}}{\dot{\mathrm{m}}}$$

The mass flow of oxidizer and flows are obtained independently by using the following relations.

$$\zeta = \mathrm{m_p}/\mathrm{m_o}$$
$$\zeta = (\mathrm{m_o} - \mathrm{m_f})/\mathrm{m_o} = \mathrm{m_p}/(\mathrm{m_p} + \mathrm{m_f})$$
$$\mathrm{m_o} = \mathrm{m_p} + \mathrm{m_f}$$

Combustion properties of the system are obtained through the use of NASA Combustion Equilibrium Analysis (CEA) software for fixed mixing ratios and exhaust gas exit velocity is obtained as a function of exit pressure.

$$\mathrm{v_2} = \sqrt{\frac{2\mathrm{k}}{\mathrm{k}-1}\mathrm{RT_1}\left[1-\left(\frac{\mathrm{p_2}}{\mathrm{p_1}}\right)^{\frac{(\mathrm{k}-1)}{\mathrm{k}}}\right]}$$
$$= \sqrt{\frac{2\mathrm{k}}{\mathrm{k}-1}\frac{\acute{\mathrm{R}}\mathrm{T_o}}{\mathcal{M}}\left[1-\left(\frac{\mathrm{p_2}}{\mathrm{p_1}}\right)^{\frac{(\mathrm{k}-1)}{\mathrm{k}}}\right]}$$

The expansion ratio of the nozzle is also a function of ambient pressure ratio and its variation is also obtained in the same manner by use of the following relation.

$$\frac{\mathrm{A_t}}{\mathrm{A_x}} = \frac{\mathrm{V_t v_x}}{\mathrm{V_x v_t}} = \left(\frac{\mathrm{k}+1}{2}\right)^{\frac{1}{(\mathrm{k}-1)}}\left(\frac{\mathrm{p_x}}{\mathrm{p_1}}\right)^{\frac{1}{\mathrm{k}}}\sqrt{\frac{\mathrm{k}+1}{\mathrm{k}-1}\left[1-\left(\frac{\mathrm{p_x}}{\mathrm{p_1}}\right)^{\frac{(\mathrm{k}-1)}{\mathrm{k}}}\right]}$$

A design nozzle expansion ratio can be chosen here for a desired nozzle exit pressure. The algorithm is tailored to parse the values of exit velocity and pressure ratios at this reference point to obtain a better estimate of mass flow through the combustion chamber.

$$\mathrm{F} = \dot{\mathrm{m}}\mathrm{v_2} + (\mathrm{p_2} - \mathrm{p_3})\mathrm{A_2}$$

On the basis of this refined value of mass flow the thrust chamber specific impulse is calculated and throat diameters are obtained for a given chamber pressure. This concludes the initial design of the rocket combustion chamber.

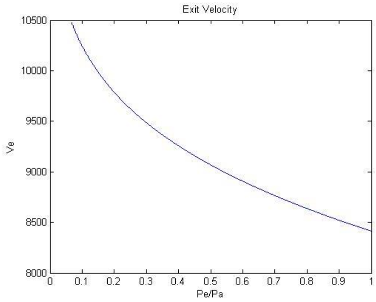

Figure 2: Variation of Exit Velocity with change in Exit pressure

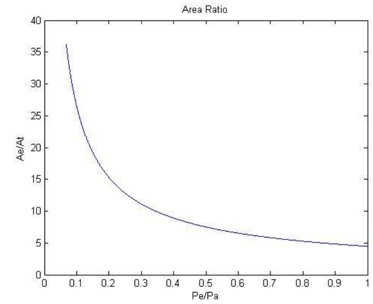

Figure 3: Variation of Area ratio with change in Nozzle exit pressure

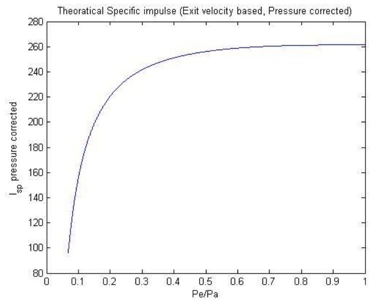

Figure 4: Variation of Specific Impulse with change in Nozzle exit pressure

The above discussion shows how the critical parameters for the proposed LRE are obtained. The design parameters for turbo-pump feed system and gas generator are then obtained through an iterative loop. The total mass flows are calculated as,

$$\dot{m}_o = (\dot{m}_o)_{gg} + (\dot{m}_o)_c$$

$$\dot{m}_f = (\dot{m}_f)_{gg} + (\dot{m}_f)_c$$
$$\dot{m}_c = (\dot{m}_o)_c + (\dot{m}_f)_c$$
$$\dot{m}_{gg} = (\dot{m}_o)_{gg} + (\dot{m}_f)_{gg}$$

An initial estimate of mass flows of gas generator is placed, and based on the value; power required for the pump is obtained. This power is then balanced by the turbine and the required mass flow of gas generator is obtained.

The gas generator features a secondary injection system that quenches the high temperature (~3200 K) combustion flame to turbine feed temperatures (~900 K) the mass balance is carried out and specific heat of the resulting mixture is obtained by using the mass fractions. A simple enthalpy balance yields the following results.

On the basis of turbine technology and materials, a valid operation point for the gas generator

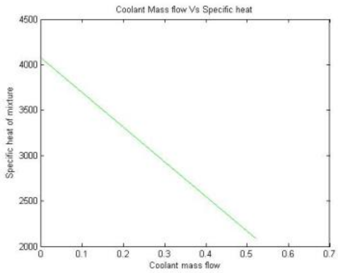

Figure 5: Variation of specific heat of mixture with mass flow of quenching fuel stream

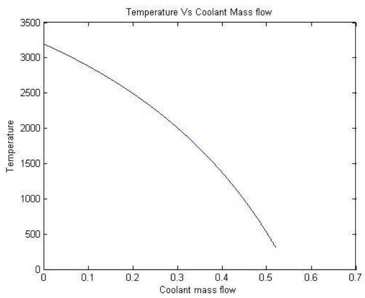

Figure 6: Variation of Temperature of mixture with mass flow of quenching fuel stream

On the basis of turbine technology and materials, a valid operation point for the gas generator is obtained. Once a refined guess of specific heat of the exhaust gas is made the loop is repeated within the Matlab script until a converged value of mass flow through gas generator and total power requirement is obtained for a given pump pressure rise.

Cooling analysis of the gas generator:

The proposed gas generator for the system cooled through regenerative cooling methods. The basic analysis assumes one dimensional energy balance across the cooling jacket of the regenerative combustion chamber. The general equation for energy balance is:

$$\dot{Q} = h_g(T_r - T_{wg})A_g = \frac{k_{CM}}{t_W}(T_{wg} - T_{wc})A_g = h_g(T_{wc} - T_c)A_{Mg}$$

The combustion gas side heat transfer coefficient is found from a correlation developed by Bartz.

$$h_g = \left(\frac{k_g}{D_g}\right) 0.026 Re^{0.8} Pr^{0.4} \left(\frac{\rho_{aw}}{\rho}\right)^{0.8} \left(\frac{\mu_{aw}}{\mu_o}\right)^{0.2}$$

The coolant side heat transfer coefficient is calculated using the following relationship:

$$h_c = \left(\frac{k_c}{D_k}\right) 0.027 Re_c^{0.8} Pr_c^{0.8} \left(\frac{\mu_c}{\mu_{wc}}\right)^{0.14}$$

The heat transfer analysis is performed with the above equations and a result is obtained for a variety of cold side wall temperature values. A feasible operation point is selected considering the choice of materials used in manufacturing the combustion chamber.

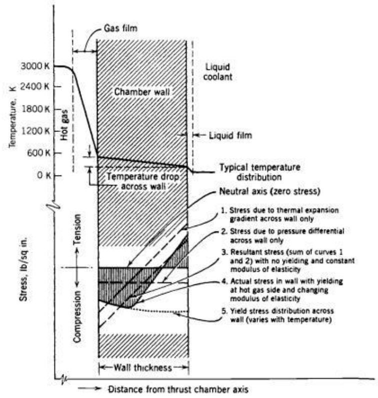

Figure 7: An illustration of the Regenerative cooling case

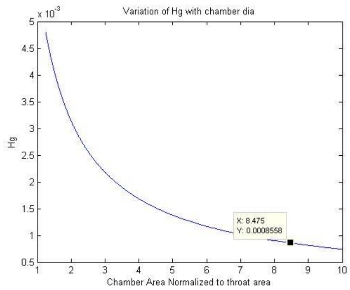

Figure 8: Variation of Hot gas side heat transfer coefficient with changing cross sectional Area of chamber

It is found that for a given chamber cross sectional area there is a critical minimum thickness required for structural integrity and a maximum thickness of chamber wall available for heat transfer after whom the available heat transfer capacity of the coolant falls short of the cooling requirement, as shown in the accompanying figure.

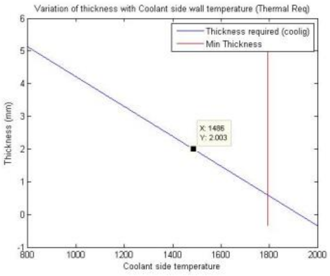

Figure 9: Variation of thickness with coolant side wall temperature

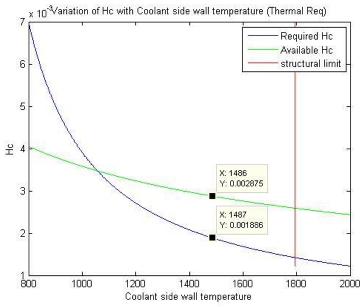

Figure 10: Comparison of available and required heat transfer coefficients for a given coolant side wall temperature

An engineering compromise is made to ensure the proper cooling of the desired gas generator with the maximum possible structural integrity.

Optimization of the LRE system design:

The above mentioned code is adapted to run in an iterative loop in order to provide an insight into optimization of the overall LRE. A set number of O/F ratios and chamber pressures are chosen to obtain LRE performance parameters. A result of such trends is shown in the following figures

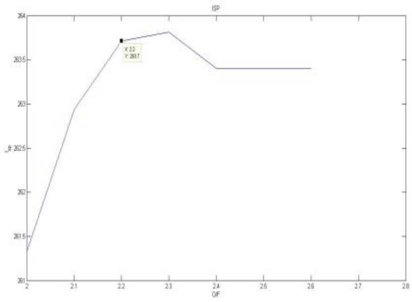

Figure 11: Variation of specific impulse with changing O/F ratio. The figure shows a clear optimum between 2.2 and 2.3

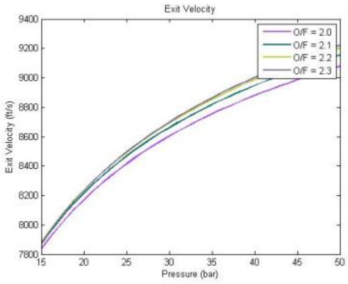

Figure 12: Variation of exit velocity with increasing pressure (bars)

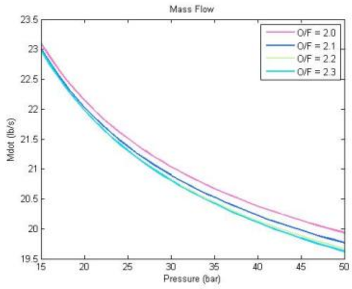

Figure 13: Variation of mass flows with increasing pressure

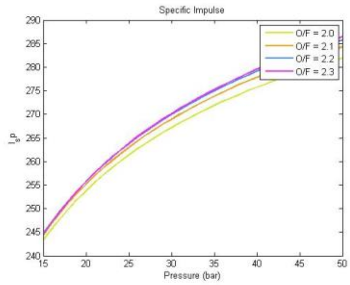

Figure 14: Variation of specific impulse with increasing pressure

On the basis of results obtained a study was carried out at different combustion pressures and expansion ratio in order to predict performance of the LRE system in vacuum of space.

Specific impulses at a fixed expansion ratio are first obtained and then these results are compared with the specific impulse obtained for a fixed exit area. It is found that although the increase in chamber pressure increases the specific impulse, it however results in decreasing produced thrust. For a given total impulse requirement this would mean more run time for designed engine. The results are presented in a tabular form in the following table.

Pressure	Exit Dia @Ae/At = 40	Ground ISP	Mass flow	F @ Ae/At=40 (kN)	Isp @ Ae/At=40	Force for exit dia @ Pc = 25, Ae/At = 40 (kN)	ISP for exit dia @ Pc = 25, Ae/At = 40
25	574.7905	263.7	9.667	33.58	354.254		
50	391.508	285.82	8.919	31.28	357.645	31.517	360.35
97.5	272.627	303.707	8.394	29.91	363.36	30.38	369.13

V. GAS GENERATOR DESIGN AND NUMERICAL ANALYSIS

The above methodology provides the design data required for a simplified gas generator. A three dimensional model of the system and is shown in the accompanying figure.

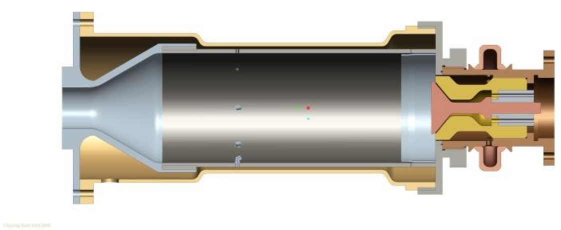

Figure 15: 3D model of the designed Gas Generator

A simplified analysis of the designed gas generator system was carried out in ANSYS numerical solver to gain confidence in the produced results. It was found that the system agrees appreciably with the one dimensional analysis results produced by the developed code.

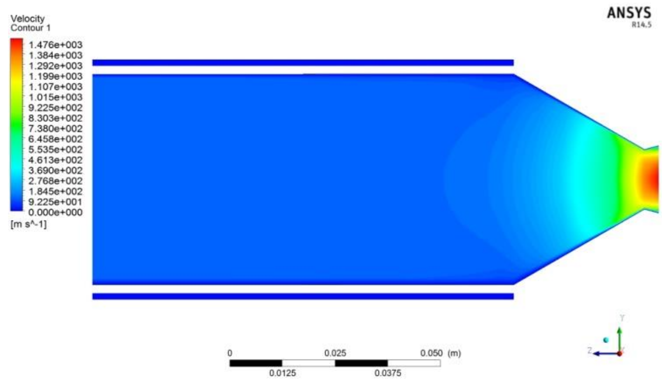

Figure 16: Velocity throughout the gas generator, showing both the hot gas core flow and the cold fuel annular flow

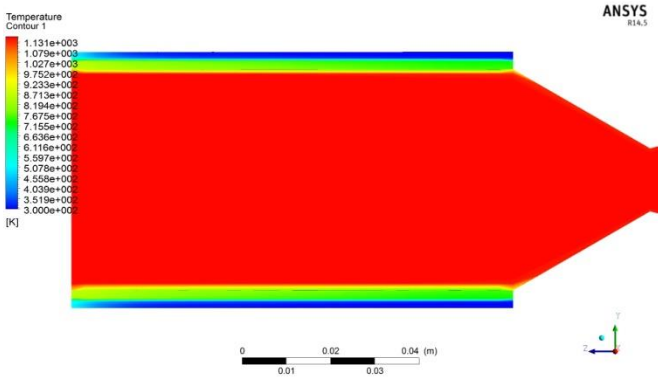

Figure 17: Temperature contours showing temperature distributions across the hot core, chamber wall and cooling fuel flow

VI. CONCLUSION

A design methodology for obtaining the design of a gas generator based LRE engine's gas generator for a proposed space mission was performed. The study takes advantage of pertinent data available in open literature to derive a fully integrated gas generator model including combustion chemistry; turbo pump feed system power requirements, regenerative coaxial cooling jacket and an optimized baseline rocket engine design.

An integrated model of the problem was coded in Matlab and result trends were obtained to observe how the system behaves to the design parameters. On the basis of these trends an operation point was chosen to enable a better performance in achieving mission success. The thermal one dimensional steady state analyses were also carried out to explore the cooling characteristics of the system and a numerical analysis was carried out to gain confidence in the obtained solutions. Additionally, the model developed could further facilitate research on different propellant combination and different expansion ratios for a variety of probable LRE missions.

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