Development of Cryogenic Engine for GSLV MkIII: Technological Challenges

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Abstract. Cryogenic engine capable of delivering 200 kN thrust is being developed for the first time in the country by ISRO for powering the upper stage of GSLV Mk-III, the next generation launch vehicle of ISRO capable of launching four tonne class satellites to Geosynchronous Transfer Orbit (GTO). Development of this engine started a decade ago when various sub-systems development and testing were taken up. Starting with injector element development, the design, realization and testing of the major sub-systems viz the gas generator, turbopumps, start-up system and thrust chamber have been successfully done in a phased manner before conducting a series of developmental tests in the integrated engine mode. Apart from the major sub-systems, many critical components like the igniter, control components etc were independently developed and qualified. During the development program many challenges were faced in almost all areas of propulsion engineering. Systems engineering of the engine was another key challenge in the realization. This paper gives an outlook on various technological challenges faced in the key areas related to the engine development, insight to the solutions and measures taken to overcome the challenges.

1. Introduction

The upper stage of GSLV MkIII the next generation launch vehicle of ISRO is powered by a cryogenic engine called CE-20. This engine produces a nominal thrust of 200 kN, but has an operating thrust range between 180 kN to 220 kN and can be set to any fixed values between them. The engine is first of its kind that works in gas generator cycle and indigenously developed. The combustion chamber burns liquid hydrogen and liquid oxygen at 6 MPa with 5.05 engine mixture ratio. The engine has a thrust-to-weight ratio of 34.7 and a specific impulse of 443 seconds in vacuum. ISRO successfully tested the sea level version of the engine for a cumulative duration of about 1900 s spread over ten hot tests which includes flight duration hot test of 635s on April 28, 2015 and endurance hot test for duration of 800 seconds.

CE-20 engine works on “Gas Generator Cycle” which has flexibility for independent development of each sub-system before the integrated engine test, thus minimising uncertainty in the final developmental phase with reduced development time. The high thrust cryogenic engine is one of the most powerful upper stage cryogenic engines in the world. Starting with injector element development for configuration of injector head, the design, realization and testing of the major sub-systems viz the gas generator, turbo pumps, start-up system and thrust chamber, have been successfully completed in a phased manner before conducting a series of developmental tests in the integrated engine mode. Apart from the major sub-systems, many critical components like the igniter, control components etc were independently developed and qualified.

The following schematic illustrates the working of the engine.
Liquid Oxygen at a temperature of 78 K and pressure of 0.21 MPa is stored in the tank and the liquid Oxygen pump driven by the respective turbine raises its pressure to the required level. High pressure liquid Oxygen from pump outlet gets divided into 3 streams. The major portion (91.3%) is injected in to the thrust chamber. Approximately 1.5% of liquid oxygen is metered by a liquid pressure regulator and cavitating venturi and injected into the gas generator. The remaining liquid oxygen is recirculated back to the pump inlet through a regulator meant for mixture ratio control.

Liquid Hydrogen at a temperature of 21 K and pressure of 0.3 MPa is stored in the tank and is raised to the required pressure by liquid hydrogen pump driven by another turbine. From the pump outlet, liquid hydrogen gets split into two streams. The first stream which is 91.2% of the total flow is fed into the thrust chamber coolant channel. Out of this, the major portion (88.7%) gets injected into the thrust chamber after passing through regeneratively cooled channels cooling the thrust chamber. A small portion is used for pressurisation of liquid hydrogen tank and stage reaction control system. The remaining quantity is used for cooling the nozzle extension. High temperature hydrogen coming out of the nozzle divergent coolant channels is expanded through two dump nozzles. The second stream of liquid hydrogen from pump outlet (8.8%) is metered by an orifice and cavitating venturi and fed into a gas generator, through the gas generator coolant channels. Liquid oxygen and liquid hydrogen injected into the gas generator chamber is ignited by an igniter and hydrogen rich hot gas is generated. This hot gas from gas generator drives the hydrogen and oxygen turbines in series, and is disposed of through turbine exhaust gas nozzle, thereby developing a small thrust. Before the vehicle lifts off, the propellant feed circuits are chilled with gaseous helium and the feed line temperatures are brought down to around 100 K. After liftoff, the pyro components used for preserving the engine internal cavities with Helium gas are sequentially operated and the various ports are communicated to the
atmosphere. Once the vehicle crosses the dense atmosphere, the propellant feed circuits are further chilled with the respective propellants and the required temperatures at pump inlets are achieved.

After the separation of lower stage, cryogenic engine is started by a pre-programmed engine start sequence. Prior to engine starting, the regulator is kept in the start position. The thrust chamber is ignited first by admitting \( \text{H}_2 \), firing the igniter and admitting liquid Oxygen in a sequential mode. After thrust chamber ignition, the turbopumps are started by admitting ambient \( \text{H}_2 \) stored in a gas bottle to the turbines. Then liquid hydrogen is admitted to gas generator and as the pump outlet pressure builds up to the levels required for gas generator ignition, the gas generator igniter firing and liquid Oxygen injection valve opening to gas generator are carried out in a sequential manner. After gas generator ignition, the engine operates in boot-strap mode and reaches the nominal operating regime. The regulator will then be moved to the nominal position. After engine performance check, the closed loop regulator system will be engaged to maintain the required mixture ratio in the thrust chamber.

After vehicle attains the required velocity or in the event of liquid Oxygen depletion, the engine is stopped by cutting off the liquid Oxygen supply to gas generator, liquid hydrogen supply to gas generator, liquid Oxygen supply to thrust chamber and liquid hydrogen supply to thrust chamber in a sequential manner.

2. Technological Challenges

From the description of working of the engine in the preceding, there were a series of challenges involved in the development of the engine. They can be broadly categorized into the following:

- Design [1] and realisation of the sub systems viz the thrust chamber, the gas generator, the liquid hydrogen and liquid oxygen turbopumps.
- System engineering of the engine with given envelope constraints and mass budget in the launch vehicle.
- Ignition of the thrust chamber at low chamber pressure in the tank head mode (available pressure in the thrust chamber will be very small) and achieving sustained and stable combustion.
- Boot strap mode starting using stored gaseous hydrogen. Gas generator ignition without hard start.
- Passive thrust control system [2] using pressure regulator and cavitating venturi.
- Mixture Ratio Control using re-circulation flow
- Development of flight sequence including chilling requirements of feed circuits
- Testing and qualification.

The subsequent sections will give certain salient features of the methodologies adopted to approach the challenges mentioned above:

As far as the thrust chamber development is concerned, all the important factors like atomization, vaporization, reaction, mixing, thermal loads, nozzle performance and stability of the engine are considered for the design [3]. Based on the detailed thermal analysis, the nozzle cooling method selected is regenerative cooling up to area ratio 10 and dump cooling from area ratio 10 to 100. In addition to this 8 % film cooling is also provided along the wall for avoiding hot spots. The design of the cooling system for the thrust chamber is arrived at after a series of parametric studies on the geometry of the coolant passage, number of coolant channels and the direction of coolant flow, with an objective to optimize the conflicting requirements of wall temperatures and pressure drops. To take care of the extreme thermal loads in the combustion chamber, throat and the nozzle, the regeneratively cooled portion of the chamber up to area ratio 10 is made out of high conductivity copper alloy and is cooled by hydrogen. Helical coolant passages are milled on the outer surface of the copper shell and
are closed out by a stainless steel outer jacket by vacuum brazing. For the dump cooled nozzle extension, both inner and outer shells are made out of stainless steel material.

Another challenge for the design of thrust chamber and the gas generator was the injector head design to result in stable combustion in the combustion chamber with combustion efficiency [4] greater than 98%. Though initial phases of development, problems were faced in this regard, it was overcome by modifying the injector element configuration and injector dome volume. Co-axial injector elements were used in both thrust chamber and gas generator.

For turbopumps, apart from the rotordynamics of the system, the main challenge was in the development of sealing system and axial thrust balancing mechanism for rotors. Bellow seal, helium purge seal and hot gas seal provide the sealing between the pump and turbine. Since the liquid oxygen pump side fluid is an oxidizer and turbine side fluid is hydrogen rich hot gas from gas generator, any mixing of two is extremely dangerous. Hence this sealing system between pump and turbine is highly critical.

During the initial design of GSLV MK III mission, studies indicated that a variation in CE-20 engine thrust to a level of 5% from the rated value does not have any bearing on the mission performance. In view of this, the engine is configured with passive thrust control system which can
maintain the required thrust with in a band of ± 3%. The cavitating venturies feed constant propellant flow rates to gas generator irrespective of variations in outlet pressure and provides constant power to the turbines thereby maintaining the engine thrust at the required level. The LOX cavitating venturi inlet pressure is maintained constant by a LOX pressure regulator provided upstream of the cavitating venturi. This pressure regulator is given a regulated GHe reference pressure for high regulation accuracy. To take care of the performance variation of LH2 pump, a control orifice is provided before the LH2 cavitating venturi which was sized using the cold flow test results of LH2 turbo pump to achieve the required venturi inlet pressure. The cavitating venturies could be sized to get the required thrust level (180 kN, 200 kN or 220 kN) depending on the mission requirement.

The engine is designed to operate at a Mixture Ratio of 5.05. Analysis indicates that for each 1% variation in mixture ratio, an additional propellant loading of 70 kg is required to meet the mission requirement. Considering the accuracy of engine tuning and variations in propellant inlet conditions, deviation in mixture ratio to the extent of 7% (max) is possible during flight with a passive mode of mixture ratio control. This can result in an additional payload penalty of 100 to 125 kg as compared to active mixture ratio control system. In order to achieve an optimum stage performance with minimum propellant outage, the engine is designed with a closed loop Mixture Ratio Control (MRC) system which regulates the Mixture Ratio with an accuracy of ± 1.5%. The MRC system comprises of the MRC regulator, regulator drive, LOX & LH2 flow meters, LOX & LH2 temperature sensors and control electronics. Using the volumetric flow rates and temperatures of LOX and LH2, control system computes the consumption mixture ratio, deviation from the requirement and control command required to correct the deviation in mixture ratio with the help of mixture ratio control algorithm. Based on the control command, the regulator drive actuates the mixture ratio control regulator mounted in the engine LOX by-pass feed line. Actuation of the regulator varies the LOX flow rate to the engine by adjusting the by-pass flow rate to get the required mixture ratio.

Selection of materials and realization strategies which involve all types of processing methods like precision machining, heat treatment, 5-axis milling and vacuum brazing for the thrust chamber, diffusion bonding for the impellers etc were challenges by itself. Team ISRO has successfully faced all the fabrication challenges over a period of time.

Figure 7: Integrated Engine Configuration Model

Configuration design of CE20 engine is carried out by considering the overall envelope of the system, its weight and functional aspects. This includes the location of system’s centre of gravity. Judicious
packaging design is applied to minimize the number of interconnecting hydraulic, pneumatic and electrical lines with their fittings, connectors, and joints. Welded joints are used as much as possible. Problems introduced by vibration, high temperature & pressures, leakage and space restrictions were thus handled. In general, a modular packaging approach is selected for engine systems. It provides a compact package for handling, transportation and installation in the vehicle. For steering, the complete engine package is gimbaled from a gimbal point that attaches directly to the thrust chamber dome. The thrust chamber serves as the principal structural member of the engine system. The minor components such as control valves, gas generator, igniters and interconnecting components are made as the integral part of major component assembly. All major components are attached to the main thrust chamber. Interconnecting components are defined as fluid carrying conduits lines or ducts along with integral elements such as bellows, flexible joints, flanges and seals, tube fittings and engine gimbal mount. The interconnecting components and lines are designed to be leak tight under extreme operating conditions of the engine. Support system for plumb lines is finalized based on frequency response analysis. Structural adequacy of the finalized configuration is ensured based on thermo structural analysis.

After exhaustive testing of the sub-systems, the next challenge was to develop the flight sequence for the engine. Here because of the tank head based ignition in vacuum, a nozzle closure is configured to create a back pressure enabling sustained thrust camber ignition. In ground sequence development a step by step approach of testing was followed starting with cold flow tests to see the control components response, thrust chamber ignition trial, thrust chamber ignition followed by start-up gas bottle operation and LH2 entry to gas generator to see the build-up characteristics and finally the above operations followed by gas generator ignition. The engine was tuned to meet the flight requirement criteria by setting the reference pressure regulator and by adjusting the by-pass orifice between the turbines. Ignition sequence is fixed based on the following conditions. Hydrogen injection temperature and mixture ratio at the time of ignition chosen to avoid non-ignition, or hard start and maintain low ignition peak to avoid back flow or low frequency instability.

Testing of this engine was another key area of challenge as the feed line length to the engine is higher compared to flight conditions where the lines are relatively shorter. Hence, oscillations in the feed lines at the time of valve opening had to be managed by delaying the sequence of operations. Handling of H\textsubscript{2} accumulation prior to engine ignition at the nozzle exit is another challenging task. This is managed by using pilot burners at the nozzle exit. Back flow of H\textsubscript{2} to the test article is prevented using suitable thermal protection system. A series of sea-level engine (Area ratio 10) tests are successfully completed in two hardware of the engine at various operating regimes of chamber pressure, inlet conditions and mixture ratio. During tests, the cavitation margins for the pumps were also kept in mind and the regimes of operation of the engine were judiciously chosen for demonstration in sea-level. Now, team ISRO is heading towards commencement of high altitude testing that involves testing of the engine in vacuum conditions as in flight. This will be a major milestone as far the development of this cryogenic engine is concerned.

3. Conclusion

This is the first time that a cryogenic engine that operates in gas generator cycle capable of delivering a thrust of 20T has been developed by ISRO. The technological challenges faced during the various phases of development were briefly highlighted in the paper and the systems approaches adopted to overcome those challenges were also highlighted. The cryogenic engine is going to be flown in LVM3-D1 flight of GSV MkIII.
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